NASA-TM-108721

FLIGHT EXPERIMENTS TECHNICAL INTERCHANGE MEETING

October 5-9, 1992
Monterey, CA.

MEETING PROCEEDINGS

SPACE STRUCTURES
PROPULSION
SPACE POWER SYSTEMS
SPACE ENVIRONMENTS AND EFFECTS
SPACE OPERATIONS

Co-Chaired by:
Dr. Judith H. Ambrus
NASA Headquarters
Office of Aeronautics and Space Technology

Major John C. Frazier
Air Force Systems Command
Phillips Laboratory Space Experiments Division

SESSION 4:

SPACE STRUCTURES

Co-Chaired by:
Mr. Jerry Newsom, NASA Langley
Research Center
Capt. Maurice Martin, Phillips Laboratory

NASA Marshall Space Flight Center

Low-Cost Active Structural Control Space Experiment (LASC)

DOE/Sandia National Laboratories

Rush Robinett

Department 9811

NASA Marshall Space Flight Center

Angelia P. Bukley

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- The DOE Lab Director's Conference identified the need for the DOE National Laboratories to actively and aggressively pursue ways to apply DOE technology to problems of National need.

- Space structures are key elements of DoD and NASA space systems and a space technology area in which DOE can have a significant impact.

- LASC is a joint agency space technology experiment.
  (DoD Phillips, NASA Marshall, DOE Sandia)

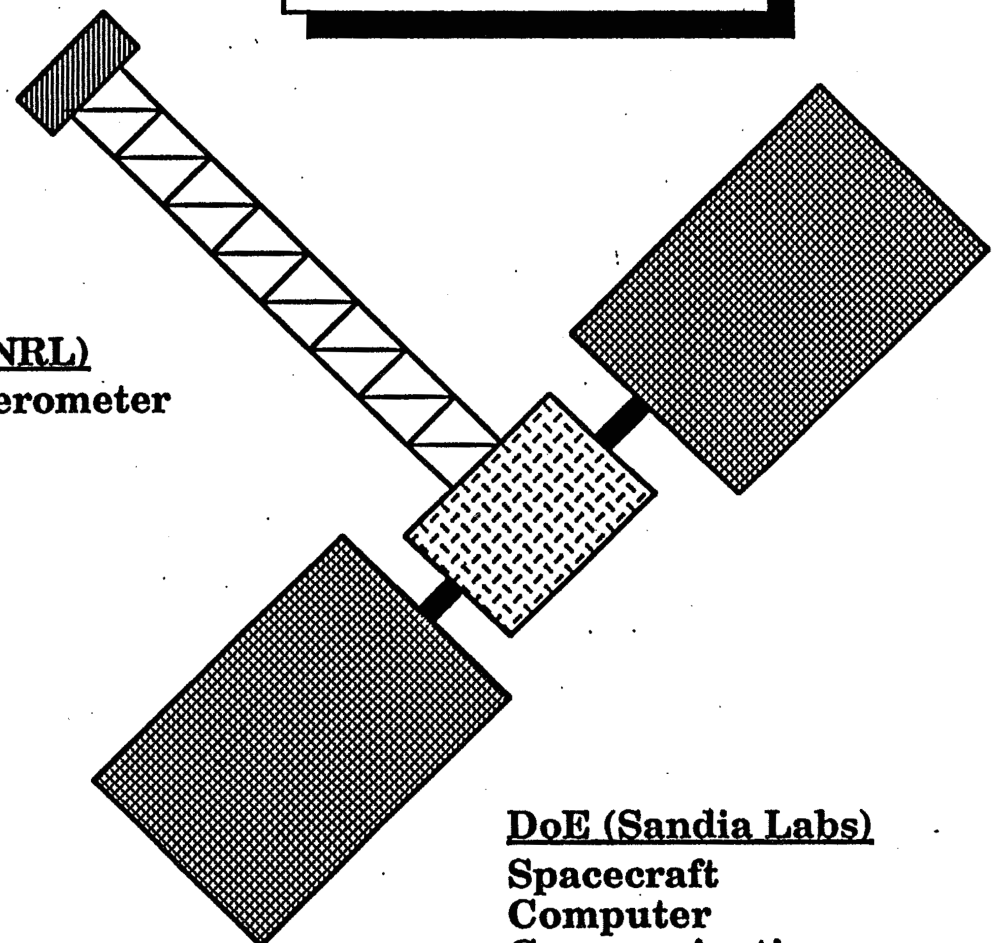
- Total cost is $35 Million spread over a 3 year project life.

- We are seeking concurrence among the three participating agencies (DoD, NASA, DOE).

LASC
Task Breakdown

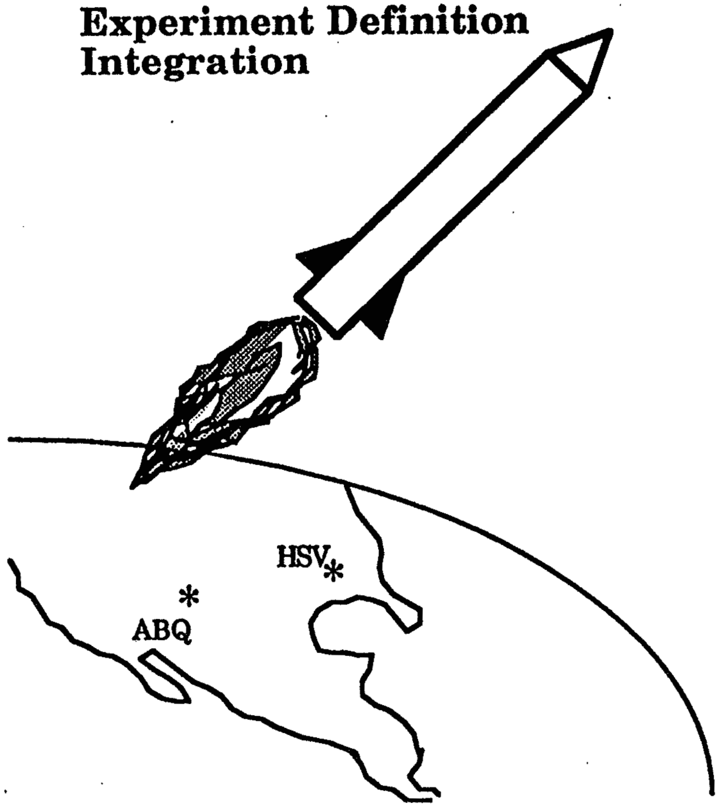
NASA (MSFC)
Experiment Definition
Payload/Sensors & Actuators
Controllers/System ID
Ground Station
Ground Testing

Phase IV Guest Investigator Testbed

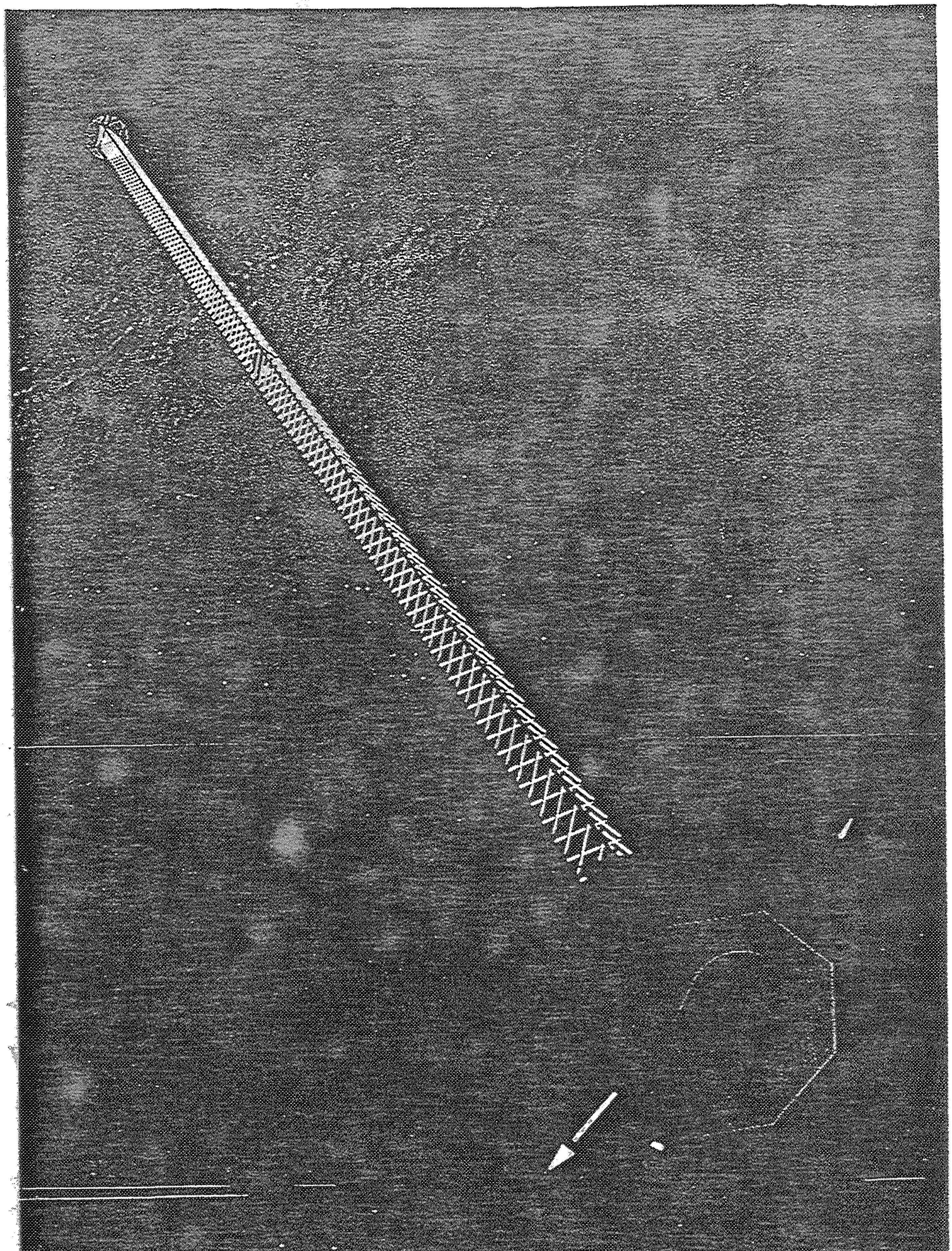


DoD (USAF)
Launch Vehicle
Controllers
Experiment Definition
Integration

DoD (NRL)
Interferometer



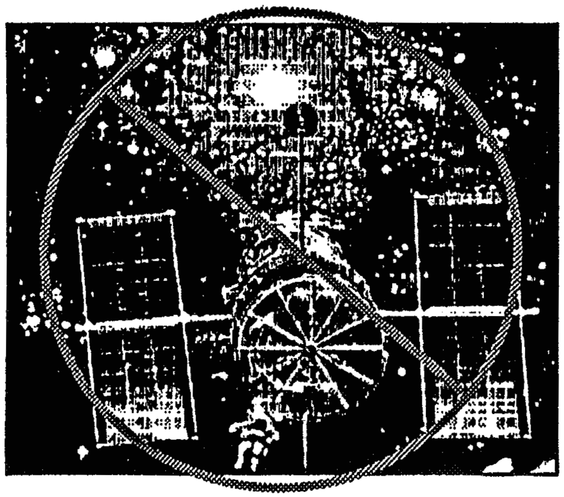
DoE (Sandia Labs)
Spacecraft
Computer
Communications
Controllers/System ID
Ground Station
Experiment Definition



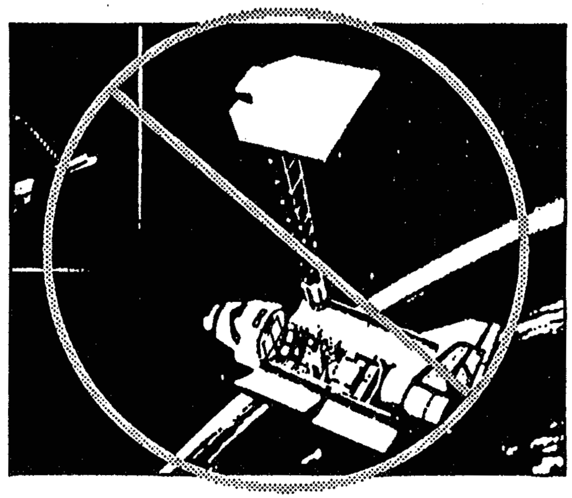
Critical Technology: Control of Large Flexible Structures In-Orbit

Necessary for:

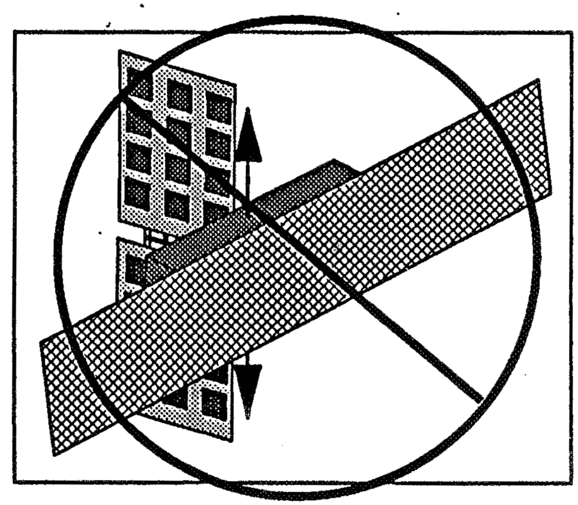
- SEI
- Deep Space Exploration
- Solar System Observation
- EOS
- Space Nuclear Power Systems
- Falcon Laser
- Solar Arrays
- SOLAR COLLECTORS



Hubble



NASA
(CASES)



SARSAT

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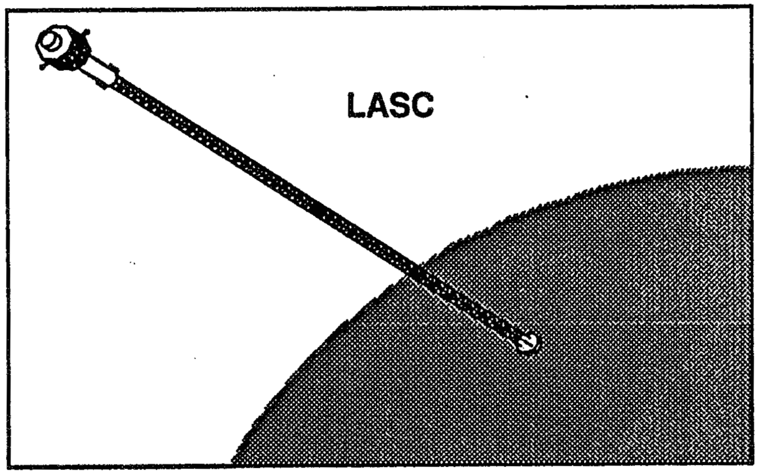
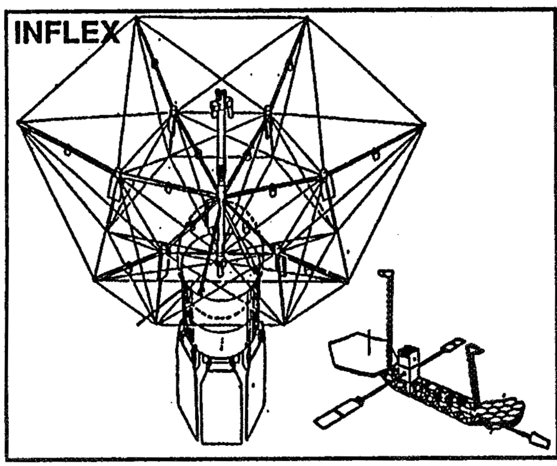
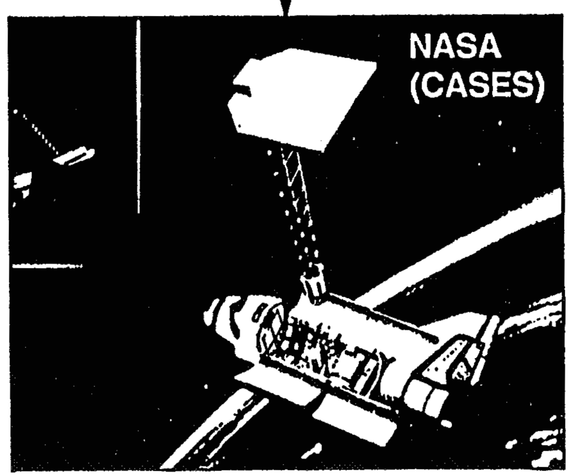
First Step of Technology Development

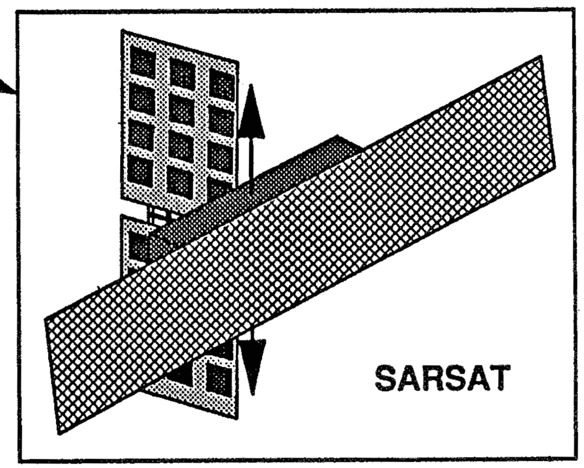


→ DOE

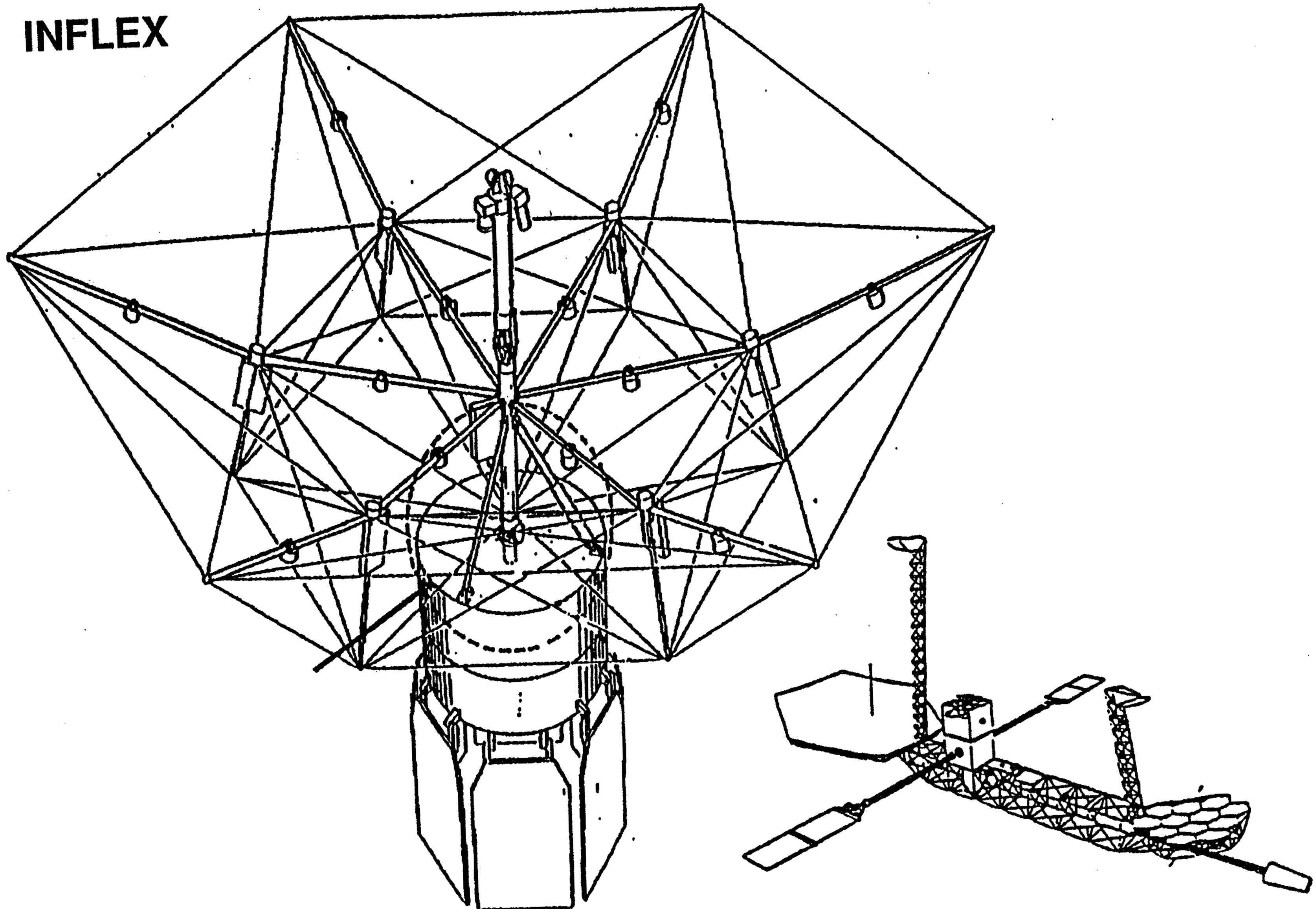
- SAR Technology
- Nuclear Electric Propulsion
- Falcon Laser
- Space-based Relay Mirrors
- Space Energy Solar Arrays







INFLEX



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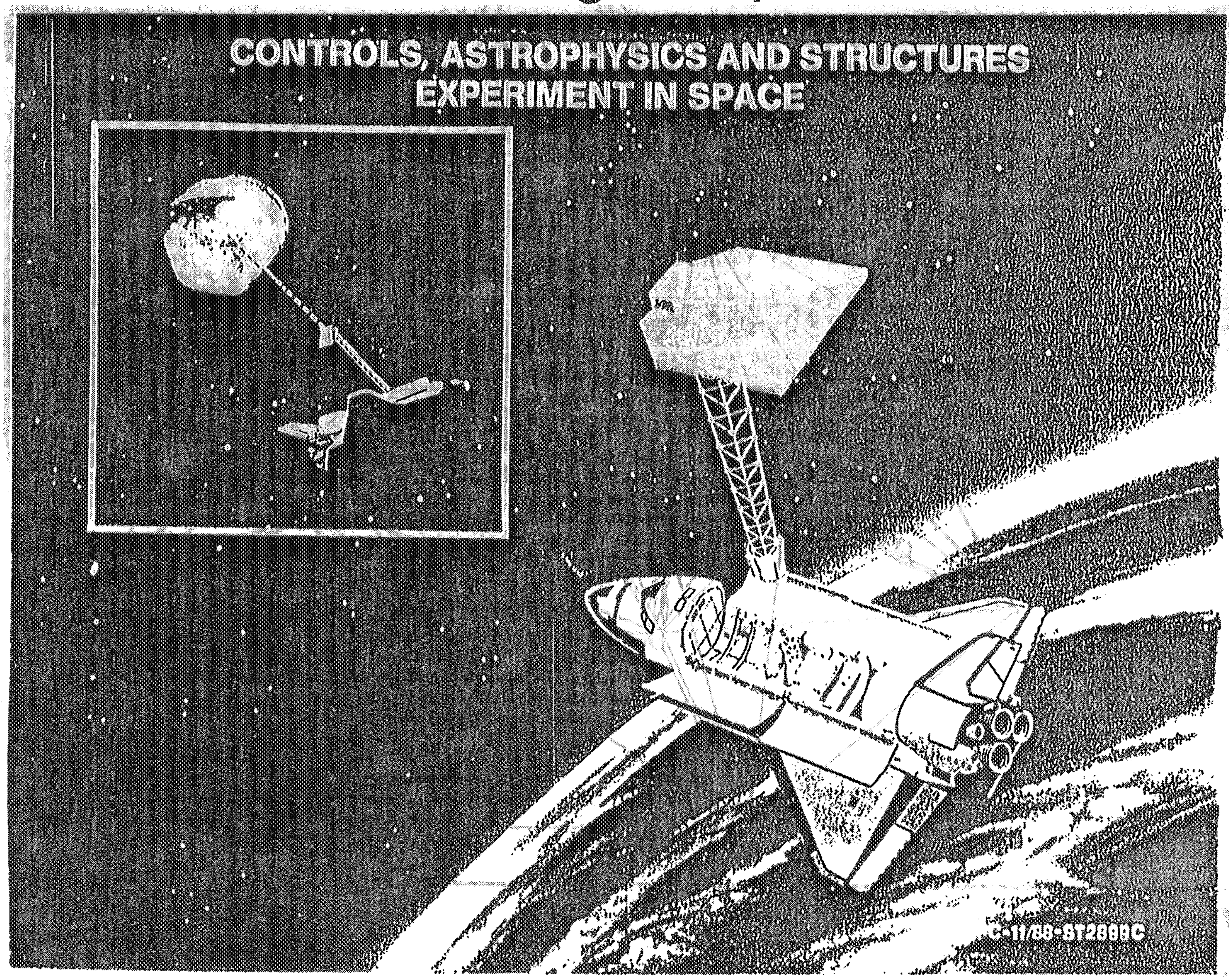
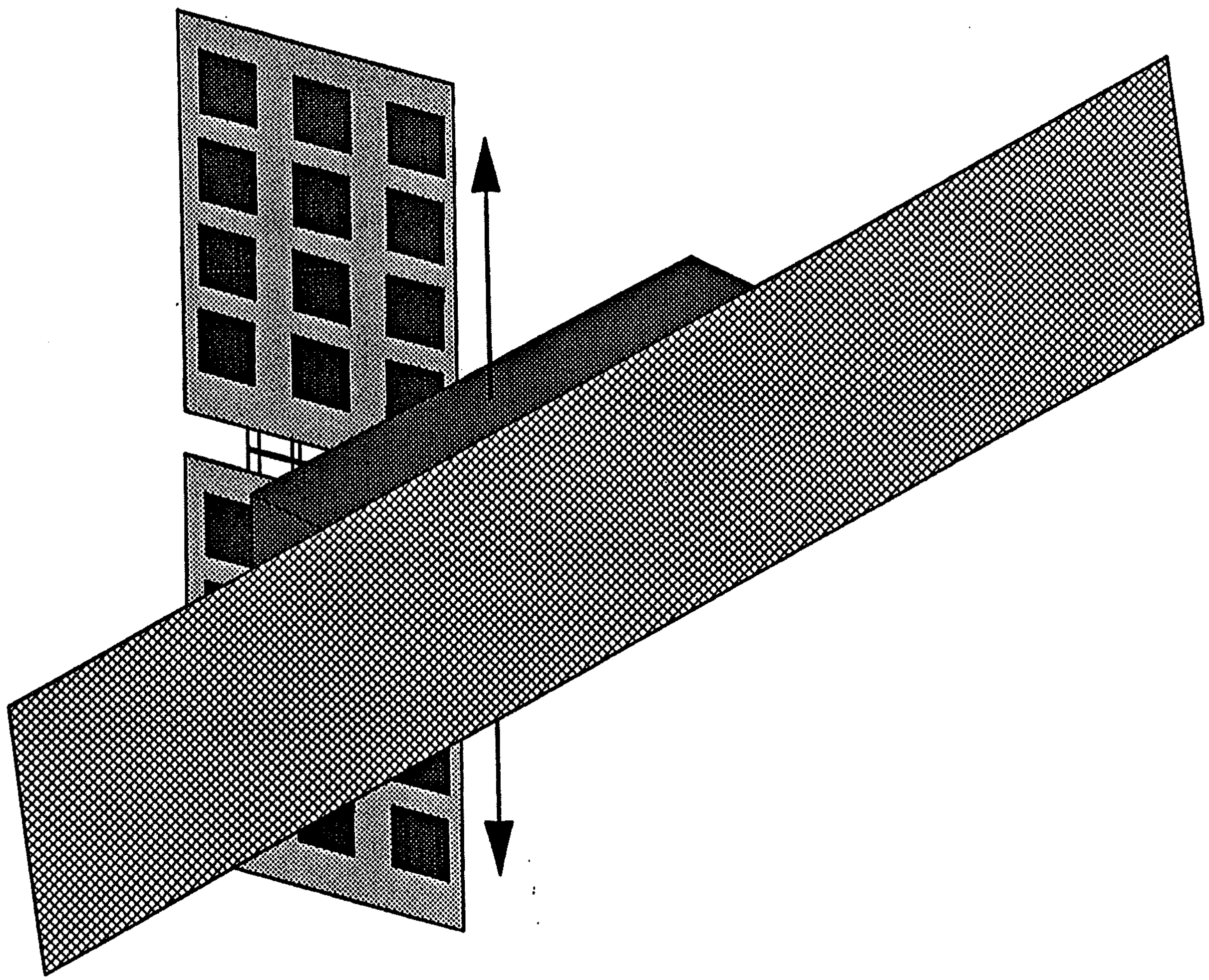
CASES Flight Experiment



SARSAT



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Fundamental Issue: Control of Flexible Structures In-Orbit

a). Help determine the relevance of ground test environments to space environments

b). Technology development of active structural control

c). Technology development of structural system identification

d). Model verification/validation

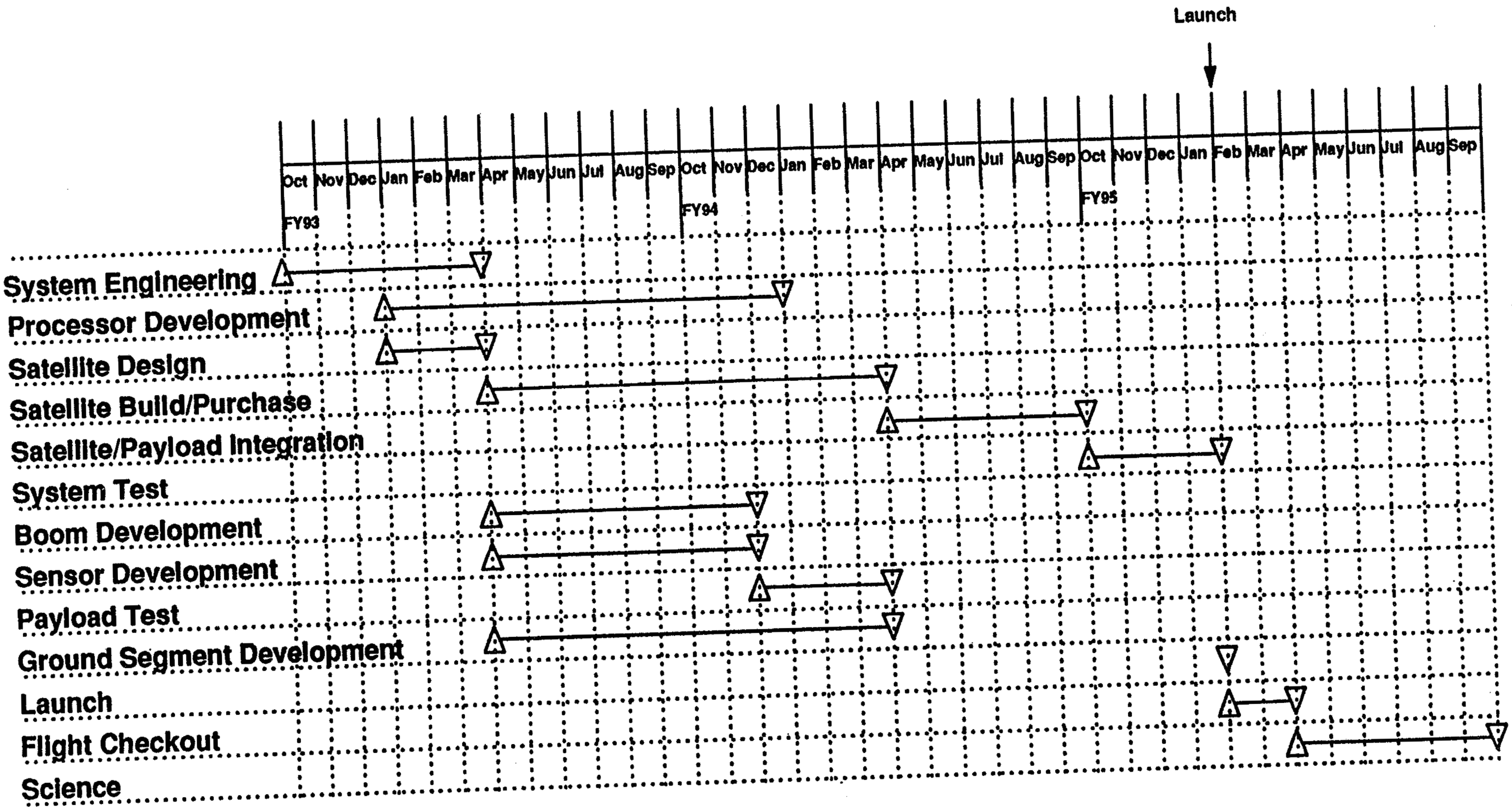
Objective: To provide a Low-Cost, Low Earth Orbit Active Structural Control Space-Based Testbed.

Why: To provide the first opportunity for a dedicated series of tests for the community that will address the CSI issues of flexible structure control in-orbit.

How: By teaming with government labs, universities, and industry in order to combine funding from several government institutions interested in this technology.

LASC Mission Objectives

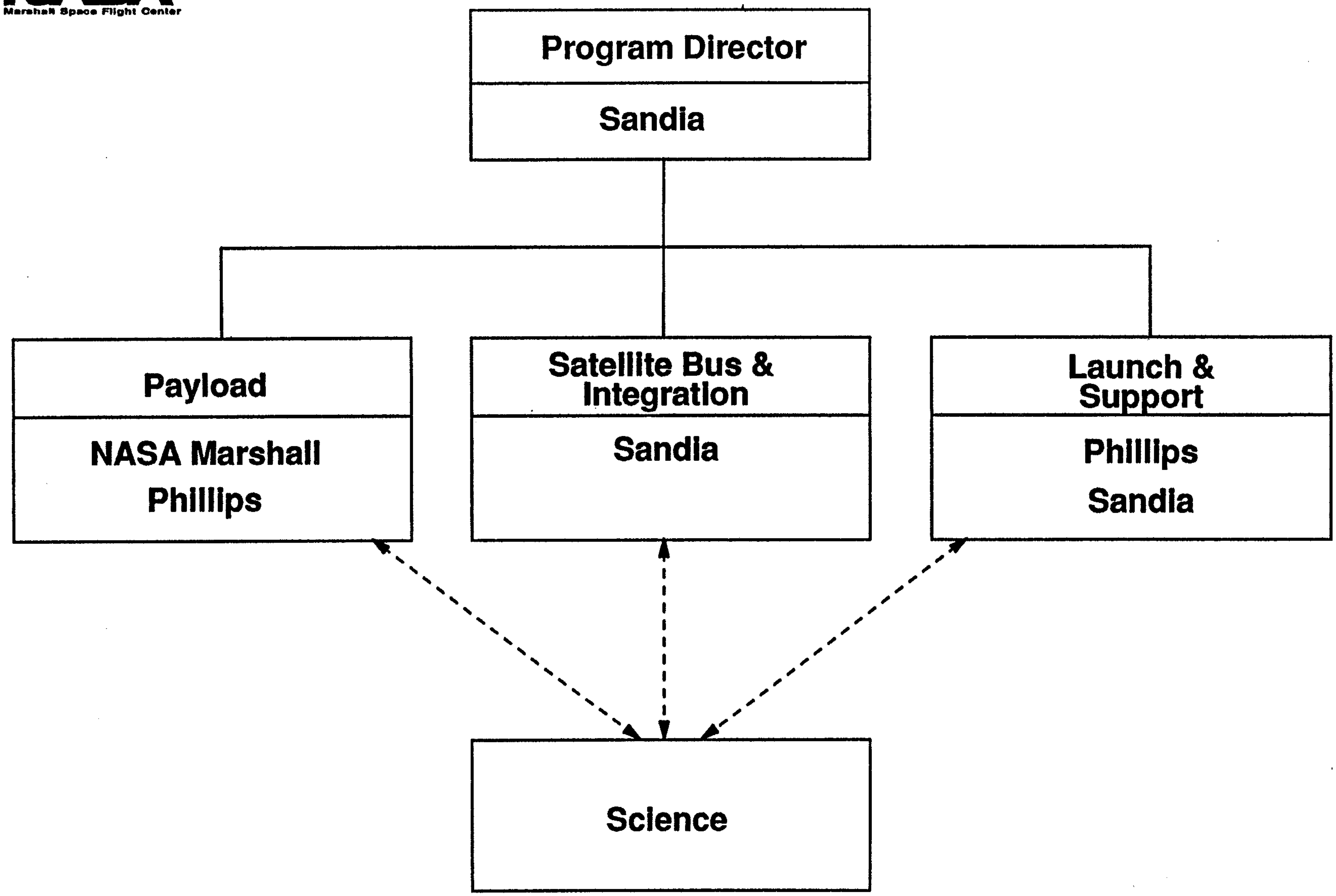
- On-orbit modal testing and system identification for comparison to similar data obtained via ground testing and simulation.
- Conduct controls experiments in the areas of vibration suppression, pointing, slewing, disturbance rejection, and deployment & retraction stabilization to the degree possible.
- Employ active/passive damping elements within the structure to assess their effectiveness.
- Examine variable structure configurations control.
- Provide an on-orbit Guest Investigator laboratory.
- Investigate tendon control.


Launch
Oct Nov Dec Jan Feb Mar Apr May Jun Jul Aug Sep
FY93
Oct Nov Dec Jan Feb Mar Apr May Jun Jul Aug Sep
FY94
Oct Nov Dec Jan Feb Mar Apr May Jun Jul Aug Sep
FY95
System Engineering
Processor Development
Satellite Design
Satellite Build/Purchase
Satellite/Payload Integration
System Test
Boom Development
Sensor Development
Payload Test
Ground Segment Development
Launch
Flight Checkout
Science

Sandia National Laboratories

LASC

NASA
Marshall Space Flight Center



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IN-STEP INFLATABLE ANTENNA EXPERIMENT

G. Veal
L'Garde, Inc.
Tustin, California

R. E. Freeland
Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California

NASA/DOD
Flight Experiments
Technical Interchange Meeting
October 5-9, 1992/Monterey, California

IN-STEP INFLATABLE ANTENNA EXPERIMENT

PRESENTATION CONTENTS

- POTENTIAL SPACE ANTENNA APPLICATIONS
- EXPERIMENT OBJECTIVES
- EXPERIMENT TECHNICAL APPROACH
- EXPERIMENT SCENARIO
- SPARTAN SERVICES
- EXPERIMENT ORBITAL CONFIGURATION
- EXPERIMENT CANISTER STRUCTURE
- SURFACE MEASUREMENT SYSTEM CONFIGURATION
- ORBITAL FUNCTIONAL SEQUENCES
- SUMMARY

JPL

IN-STEP INFLATABLE ANTENNA EXPERIMENT

POTENTIAL SPACE ANTENNA APPLICATIONS

APPLICATION	APERTURE SIZE RANGE (M)	RF RANGE (GHz)	INFLATABLE ANTENNA APPLICATION POTENTIAL*
MOBILE COMMUNICATIONS	10 — 20	1.5	VERY HIGH
MOBILE COMMUNICATIONS	4 — 8	20 — 30	MODERATE
EARTH OBSERVATION RADIOMETRY	20 — 40	6 — 60	MODERATE
ACTIVE MICROWAVE SENSING	0.4 × 2 & 4 × 16	1 — 94	MODERATE
OVLBI	20 — 25	0.3 — 90	MODERATE
DOD SPACE BASED RADAR	20 — 30	1.5 — 2.5	VERY HIGH

* APPLICATION CRITERIA
- LOW COST
- LOW WEIGHT
- HIGH RELIABILITY
- HIGH PRECISION
- PACKAGING EFFICIENCY
- DIMENSIONAL STABILITY
- DYNAMICS CHARACTERISTICS
- CONCEPT GROWTH POTENTIAL

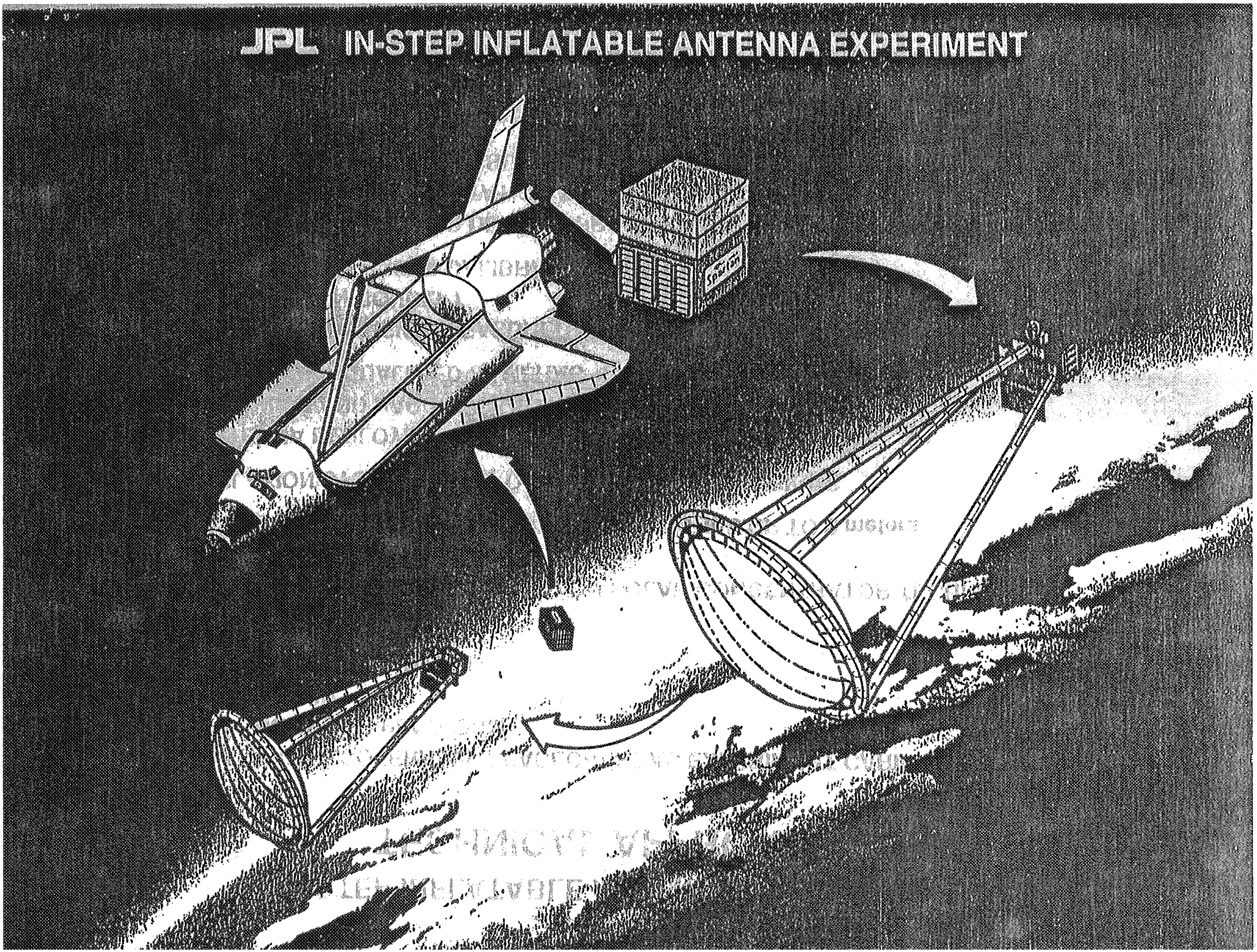
IN-STEP INFLATABLE ANTENNA EXPERIMENT

EXPERIMENT OBJECTIVES

- VALIDATE THE DEPLOYMENT OF A 14 METER INFLATABLE PARABOLIC REFLECTOR STRUCTURE IN A ZERO GRAVITY ENVIRONMENT
  - INFLATABLE ELEMENT DEPLOYMENT SEQUENCE
  - DEPLOYMENT RATES
- MEASURE THE REFLECTOR SURFACE ACCURACY UNDER ORBITAL MECHANICAL AND THERMAL LOADING CONDITIONS
  - FIVE SUN ANGLES
  - THREE REFLECTOR/CANOPY PRESSURES
- INVESTIGATE REFLECTOR STRUCTURE DAMPING CHARACTERISTICS UNDER ORBITAL OPERATIONAL CONDITIONS
  - EXCITE FUNDAMENTAL NODES
  - MEASURE AMPLITUDE DECAY

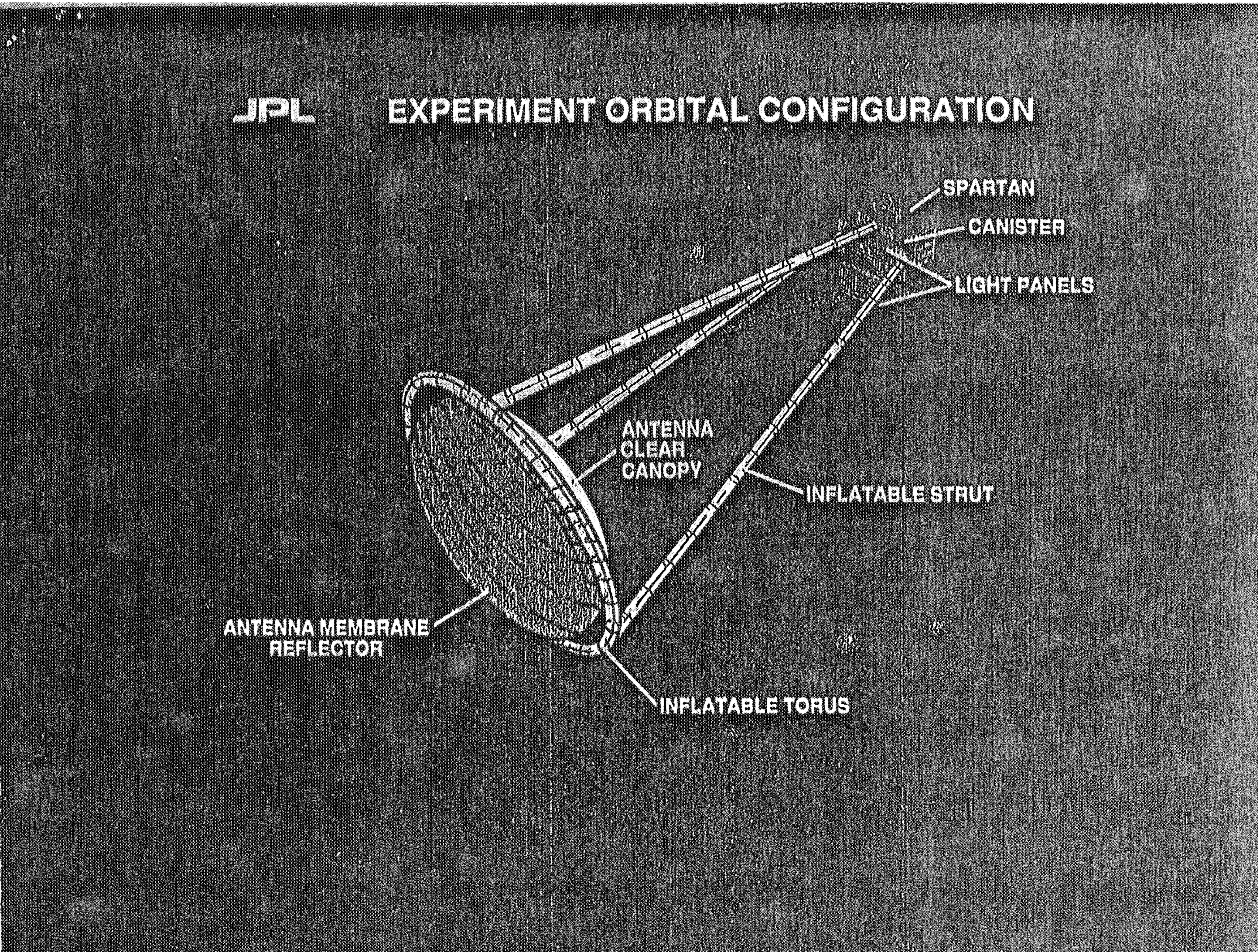
IN-STEP INFLATABLE ANTENNA EXPERIMENT
TECHNICAL APPROACH

- SPARTAN RECOVERABLE SPACECRAFT AS EXPERIMENT CARRIER
  - MOUNTING PLATFORM
  - POWER
  - ATTITUDE CONTROL
  - DATA RECORDING
- REFLECTOR STRUCTURE BASED ON SOLAR CONCENTRATOR TECHNOLOGY
  - CONFIGURATION IDENTICAL
  - CONCEPT DEVELOPMENT OF STRUCTURES UP TO 9 meters
- INFLATION SYSTEM BASED ON FLIGHT PROVEN DESIGNS
- ANTENNA DEPLOYMENT MONITORED WITH VIDEO CAMERAS
  - WIDE AND NARROW ANGLE CAMERAS
  - FLIGHT QUALIFIED CAMERAS
- SURFACE PRECISION MEASURED WITH DIGITAL IMAGING RADIOMETER
  - PROVEN CONCEPT
  - GROUND BASED CALIBRATION
- STRUCTURAL DAMPING DERIVED FROM MOTION DECAY PLOTS
  - STRUCTURAL EXCITATION PROVIDED BY SPARTAN CONTROL SYSTEM
  - MOTION DECAY MEASURED WITH TRANSDUCERS

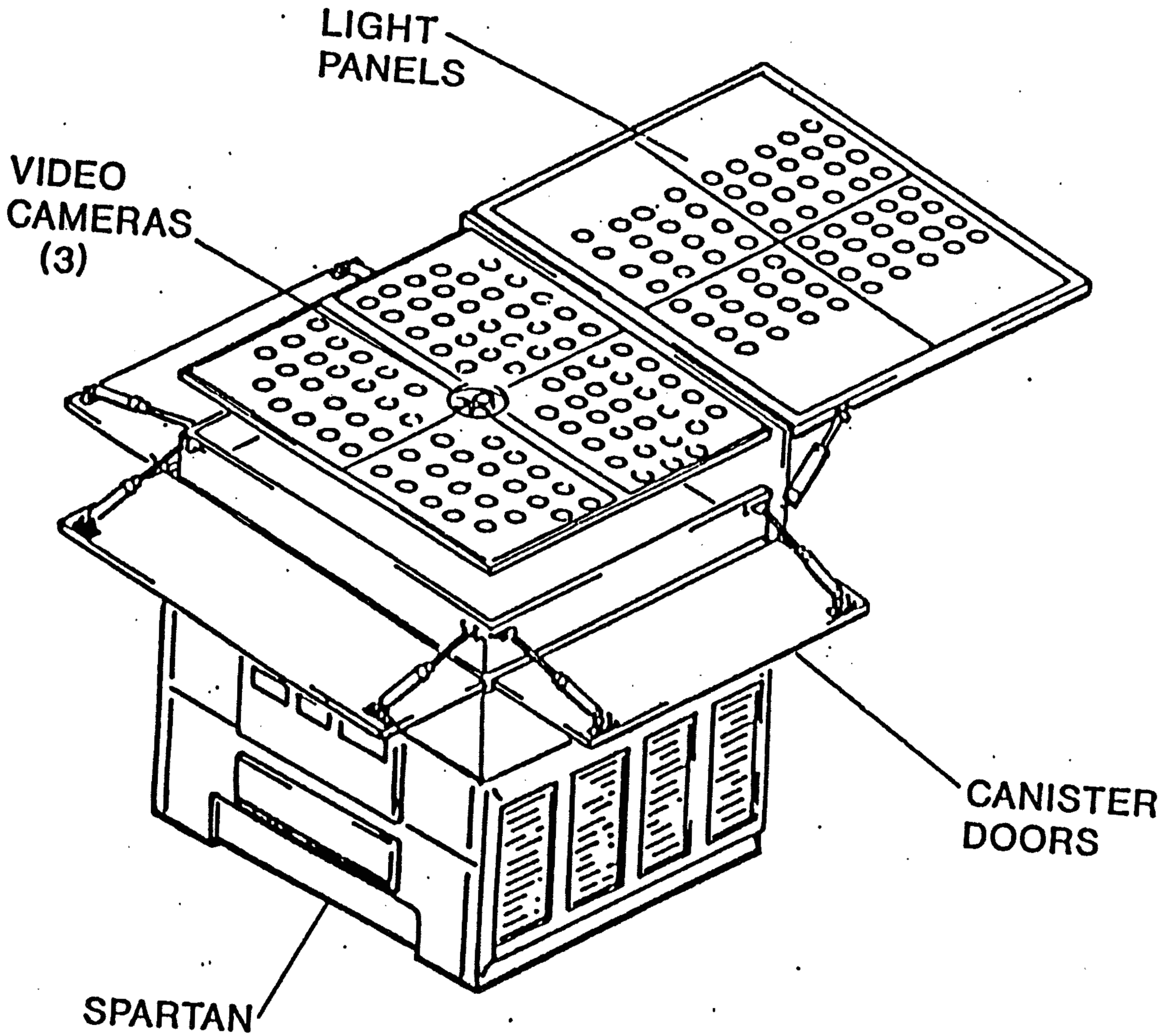

JPL IN-STEP INFLATABLE ANTENNA EXPERIMENT

IN-STEP INFLATABLE ANTENNA EXPERIMENT
SPARTAN SERVICES

- EXPERIMENT CARRIER
- EXPERIMENT INTERFACE WITH STS
- THERMAL CONTROL (PRIOR TO DEPLOYMENT)
- ATTITUDE CONTROL
- ELECTRICAL POWER
- DATA RECORDING
- ORBIT POSITION AND ATTITUDE VS. TIME DATA
- EXPERIMENT INITIATION SIGNAL (TWO FAULT TOLERANT)
- EXPERIMENT/SPARTAN SEPARATION SYSTEM

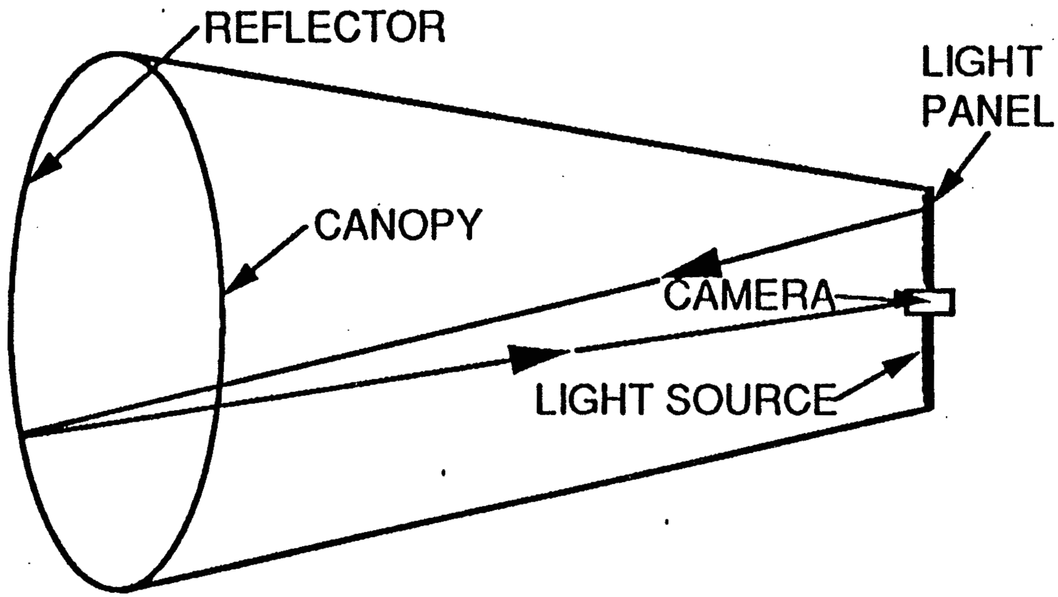

JPL
EXPERIMENT ORBITAL CONFIGURATION
SPARTAN
CANISTER
LIGHT PANELS
ANTENNA
CLEAR
CANOPY
INFLATABLE STRUT
ANTENNA MEMBRANE
REFLECTOR
INFLATABLE TORUS

IN-STEP INFLATABLE ANTENNA EXPERIMENT
CANISTER AND SPARTAN

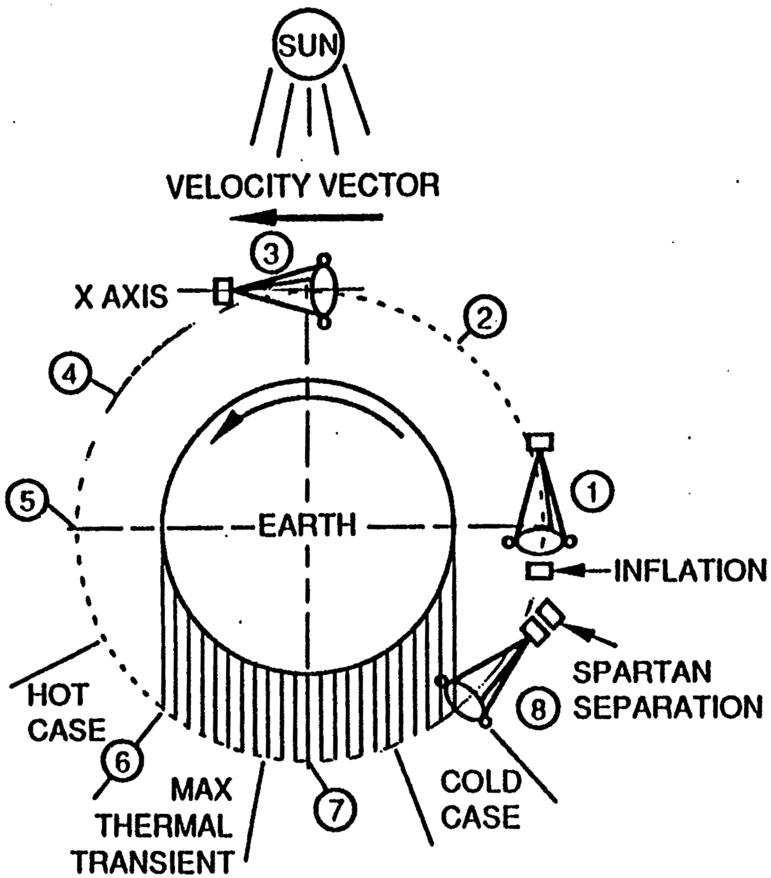


IN-STEP INFLATABLE ANTENNA EXPERIMENT

SURFACE MEASUREMENT SYSTEM CONFIGURATION



IN-STEP INFLATABLE ANTENNA EXPERIMENT
ORBITAL FUNCTIONAL SEQUENCES



ORBITAL FUNCTIONS

- EXPERIMENT INITIATION COMMAND FROM SPARTAN
- CANISTER DOOR DEPLOYMENT
- REFLECTOR STRUCTURE DEPLOYMENT BY INFLATION
- VIDEO COVERAGE OF REFLECTOR DEPLOYMENT
- SURFACE MEASUREMENT AS FUNCTION OF
  - SUN ANGLE: ORBITAL POSITIONS 1-5
  - CANOPY PRESSURE: ORBITAL POSITION 6-8
- ANTENNA STRUCTURE EXCITATION PROVIDED BY SPARTAN
- MEASUREMENT OF STRUCTURE AMPLITUDE DECAY
- ANTENNA/SPARTAN SEPARATION

IN-STEP INFLATABLE ANTENNA EXPERIMENT
SUMMARY

- PROGRAM STATUS
  - PRELIMINARY DESIGN COMPLETE
  - CONCEPTUAL DESIGN REVIEW ACCOMPLISHED
  - CARRIER INTERFACE ESTABLISHED
  - PHASE 0 SAFETY REVIEW IN PROCESS
  - PHASE C/D PROGRAM PLANNING COMPLETE
- EXPERIMENT RESULTS
  - DEPLOYMENT RELIABILITY VALIDATED BY EXPERIMENT
  - LOW WEIGHT AND VOLUME DEMONSTRATED BY FABRICATION OF LARGE SIZE STRUCTURE
  - LOW STRUCTURE COST VERIFIED BY LOW COST EXPERIMENT
  - USER POTENTIAL DETERMINED BY RESULTS OF SURFACE MEASUREMENT
  - ESTABLISH CONCEPT TECHNOLOGY DATA BASE
  - PROJECTIONS OF PERFORMANCE FOR DIFFERENT APPLICATIONS

NASA/DoD FLIGHT EXPERIMENTS TECHNICAL INTERCHANGE MEETING

NASA IN-STEP / MDMSC JITTER SUPPRESSION EXPERIMENT (JITTER)

5 October 1992

Edward V. White, Experiment Manager (314) 232-1479

McDonnell Douglas Missile Systems Company

BACKGROUND

- Many present and future systems would benefit from vibration suppression
  - Precision pointing
  - Precision dimensional stability
  - Micro-gravity
- Benefits are increased performance and reliability
  - Uncertain dynamics
  - Uncertain or unexpected disturbances
  - Increasingly severe disturbance environments
- Current users consider vibration suppression technology immature
  - Schedule and cost risk unknowns
  - Lack of in-space demonstrations
- Use of this technology on high value/high priority systems is unlikely without in-space testing and demonstrations

EXPERIMENT OBJECTIVES / GOALS

OBJECTIVES: Develop and demonstrate in-space performance of both passive and active damping systems for suppression of micro-amplitude vibration

- on an actual application structure
- operate despite uncertain dynamics and uncertain disturbance characteristics

Correlate ground and in-space performance

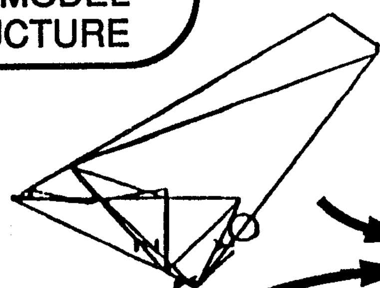
- Performance metric is vibration attenuation

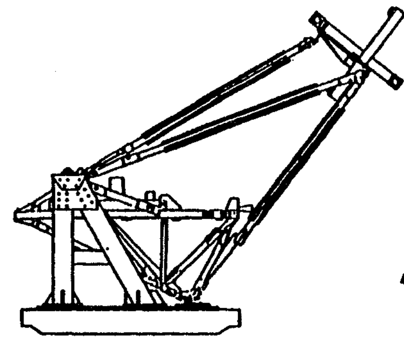
GOALS: Achieve vibration suppression equivalent to:

5% passive damping in selected modes
15% active damping in selected modes
(Baseline structure intrinsic damping is approx. 0.5%)

JITTER EXPERIMENT APPROACH

START WITH TEST VERIFIED FINITE ELEMENT MODEL AND STRUCTURE





MODIFY TO DAMPED EXPERIMENT CONFIGURATION

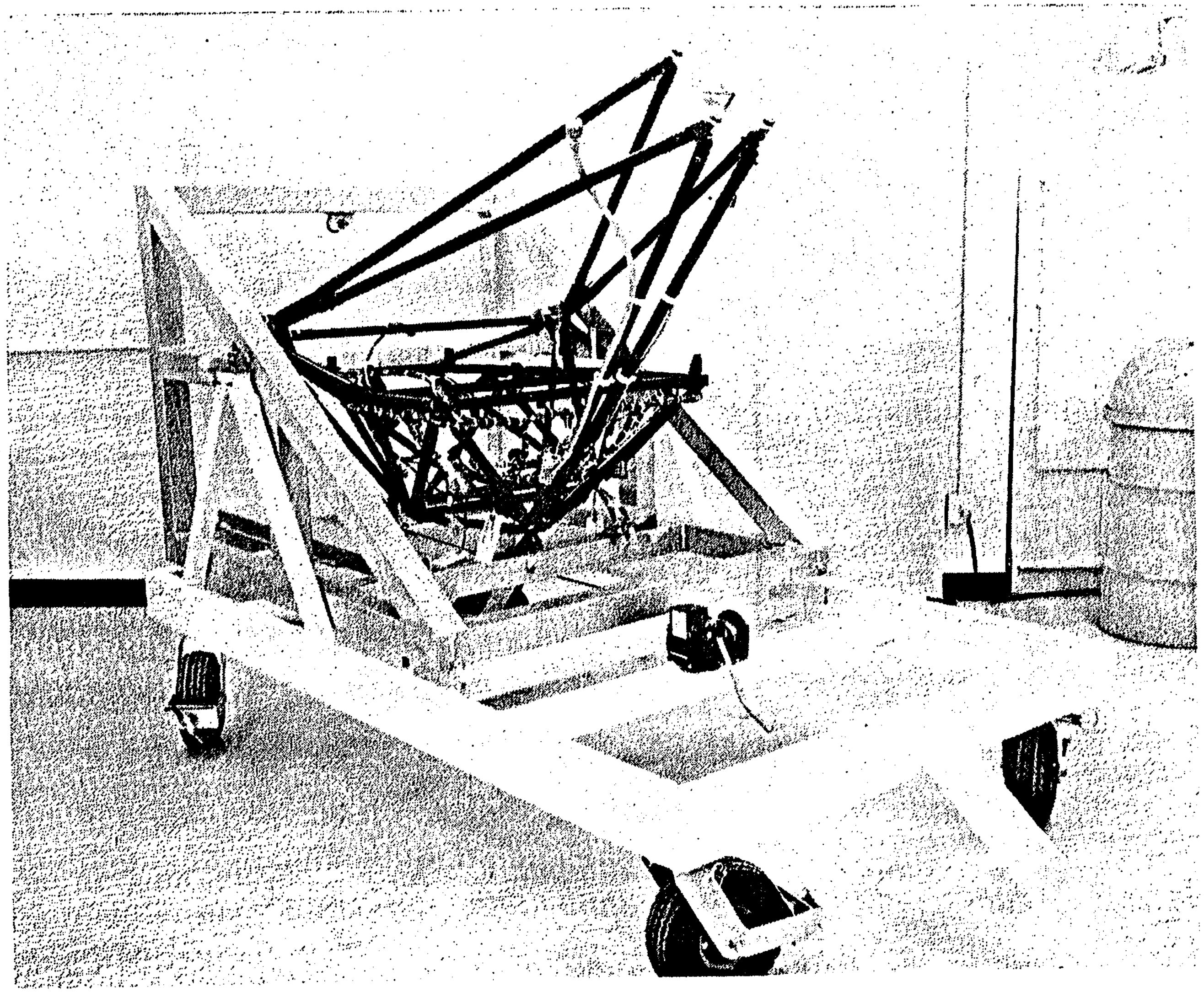
DEVELOP PASSIVE AND ACTIVE DAMPING SYSTEMS

GROUND TESTING: DAMPING SYSTEM PERFORMANCE AND FLIGHT CERTIFICATION

UPDATE GROUND BASED PREDICTIONS OF ON-ORBIT DAMPING PERFORMANCE

DETERMINE ON-ORBIT DAMPING PERFORMANCE AS MEASURED BY RESPONSE ATTENUATION

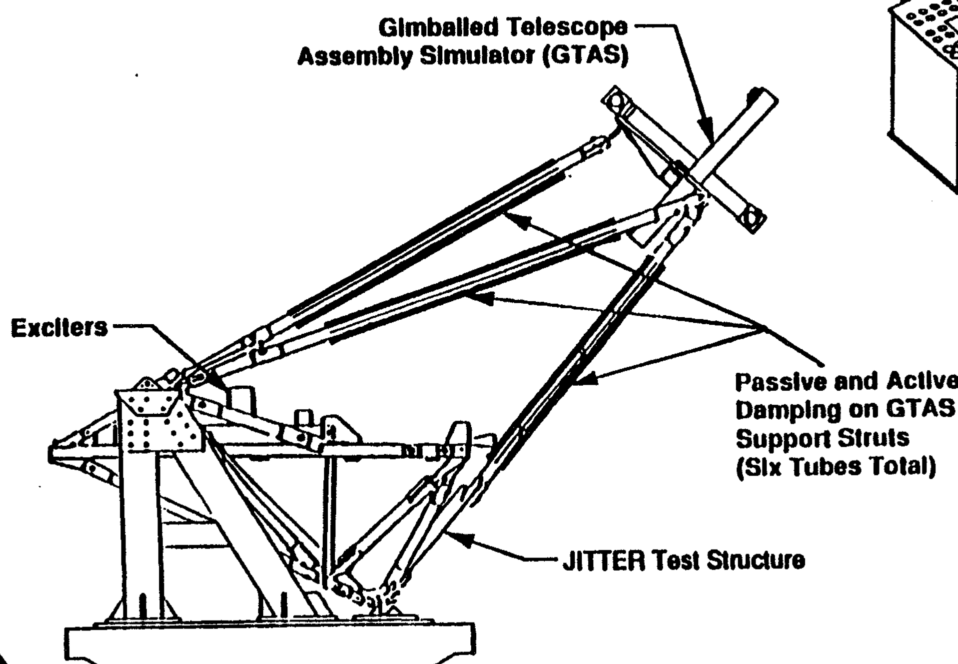
EXPERIMENT TAKES ADVANTAGE OF AN EXISTING PRECISION SPACE STRUCTURE

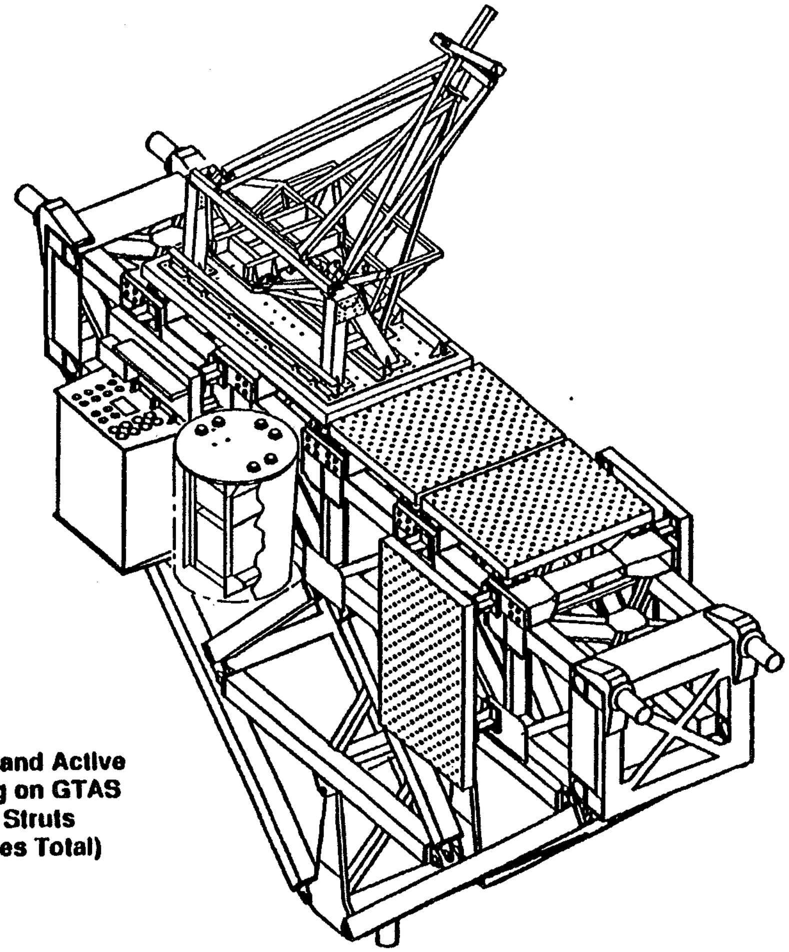


McDonnell Douglas Missile Systems Company

EXPERIMENT CONFIGURATION

- Uses Hitchhiker-M Carrier
- Accelerometers measure vibration suppression
- Sine, random and Shuttle background excitation





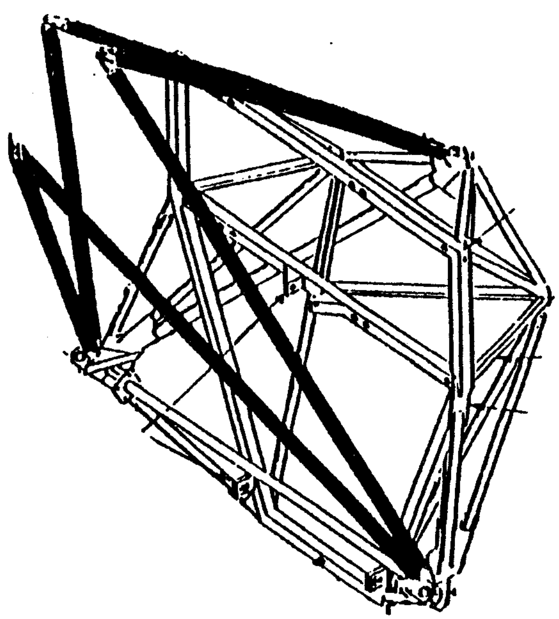
JITTER DAMPING SYSTEMS

Passive: Constrained layer viscoelastic material (stave configuration)

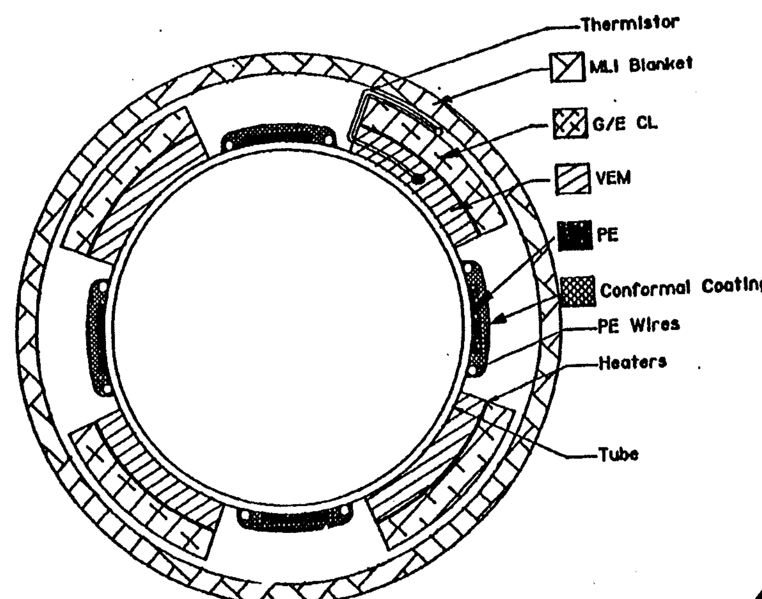
Active: Piezo-ceramic sensors and actuators with digital controller controlling both axial and bending strain

Frequency Range: 20 Hz to 200 Hz (covers 45 experiment modes)

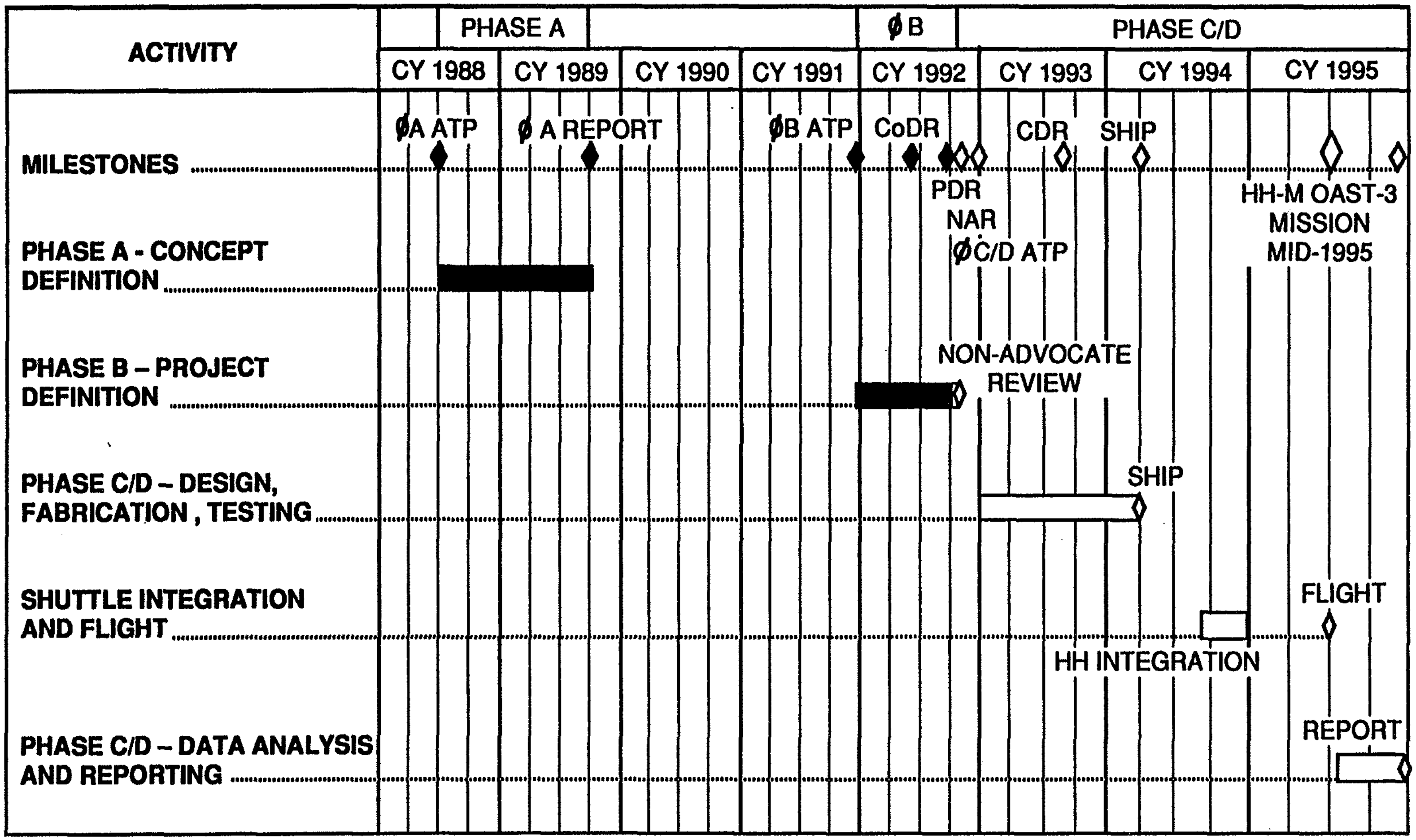
Both damping systems applied to six major struts



Strut Cross-Section



JITTER SUPPRESSION EXPERIMENT PROGRAM SCHEDULE



SPECIFIC TECHNOLOGY BENEFITS

- Uses an actual application structure to include complexities inherent in real systems
  - Representative size and structural complexity
  - Broad frequency range of interest results in uncertain dynamics
    - High modal density
    - Higher order mode shapes
    - Modal coupling with uncertain carrier vehicle modes
    - Effects induced by One-g strain levels
- Demonstrates performance of damping systems tolerant to uncertain and/or unexpected disturbances
- Development of effective passive and active damping designs compatible with Shuttle payload requirements
- Provides data for undamped structure, passive damping alone, active damping alone, and passive and active damping together
- Demonstrates effectiveness of passive and active damping systems against Shuttle environments (typical of manned vehicle platforms)

Description of the Joint Damping Experiment (JDX)

Flight Experiments Technical Interchange Meeting

Principal Investigators:
Steven L. Folkman
Frank J. Redd
Mechanical and Aerospace Engineering Department
Utah State University

Presented by:
Steven L. Folkman

October 5, 1992

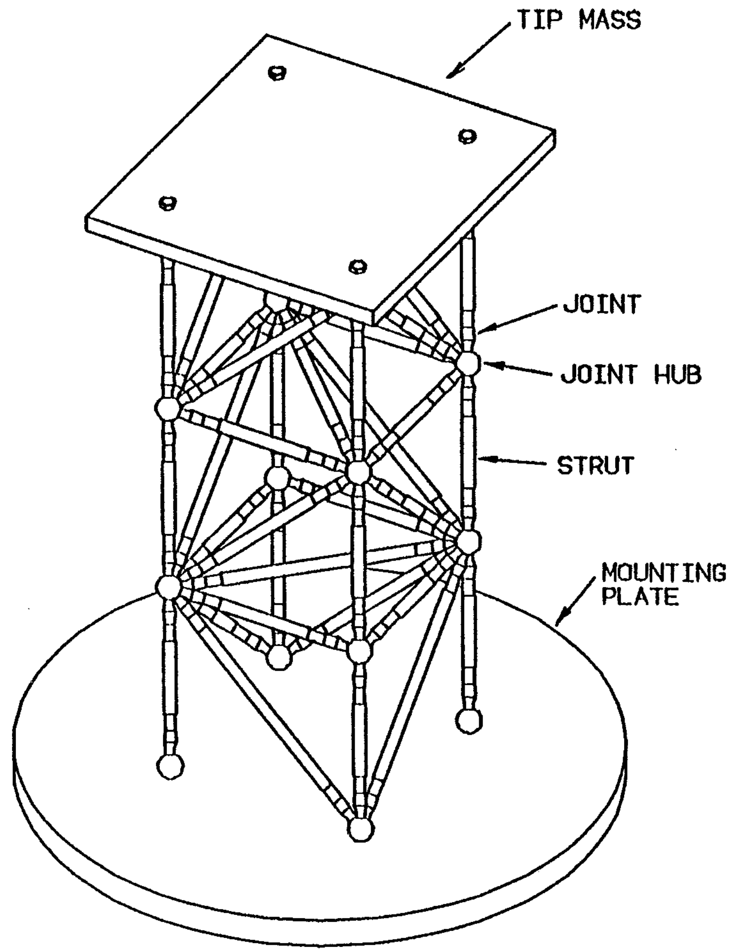
Overall Objective

Develop a small-scale shuttle flight experiment which allows researchers to: 1) characterize the influence of gravity and joint gaps on structural damping and dynamic behavior of a small-scale truss model, and 2) evaluate the applicability of low-g aircraft test results for predicting on-orbit behavior.

JDX Description

The experiment consists of a three-bay truss and associated hardware for truss excitation and measurement of oscillations.

- **The experiment dimensions fit inside of a 5 cubic foot GAS canister.**
- **Cantilever truss with a tip mass to reduce the resonant frequency.**
- **Canister can be evacuated to eliminate air damping.**
- **Truss excitation in two bending modes and a torsional mode.**
- **Truss tip supported during launch and reentry.**



Project Objectives

1. Student oriented project.

    - Graduate and undergraduate students will perform most of the design, analysis, and testing effort under the direction of the principle investigators.

    - JDX is to be relatively simple and inexpensive.

    - Fly as a Complex Autonomous Payload (CAP) in a sealed GAS canister to simplify integration problems and safety concerns and maximize flight opportunities.

    - JDX will provide a meaningful experience for students and an opportunity to extend the understanding of damping mechanisms in joints.

2. Construct a small truss with joints which provide gravity dependent damping.
    - Past tests show that a truss with pinned-joints can produce gravity dependent damping.
    - Damping from tight joints is generally not gravity dependent.

Project Objectives (continued)

3. Develop a database of damping behavior for various gravity environments and various joint pin gaps.

   - Ground-based testing to measure damping with 1-g loads.
     - A good characterization of the truss dynamics can be achieved.
     - Verification of gravity dependent damping achieved by testing the truss in different orientations.

   - Fly in aircraft tests for short duration low-g tests.
     - JDX must be cantilevered during testing - aircraft vibrations will be significant.
     - Short time period.
     - Space flight needed to verify low-g aircraft tests.

   - Fly as CAP Payload to measure damping in micro-gravity.
     - CAP Payload should provide relatively low cost and simple integration.
     - Test during orbiter free drift mode for a micro-gravity environment.

4. Correlating ground-based and low-g aircraft test results with on-orbit test results.

   - Can ground tests simulate zero-g tests.

   - Can low-g aircraft tests simulate zero-g tests.

Project Objectives (continued)

5. Refining analytical models of gravity-dependent damping mechanisms based on test results.

   - Relate measured damping with damping predicted from strut hysteresis tests, expected material damping, and simple friction and impact damping models.

   - Compare measured time histories with results of transient, non-linear finite element modeling techniques.

     - The recorded data should be a time history which can be readily be simulated using a transient computer model.
     - The transient decay of a single mode is desired.
     - A simple "twang" excitation method will produce the desired excitation.
     - The only motion recorded in flight would be the tip mass to reduce data storage.

Technology Need

Proposed space structures could often benefit from accurate prediction of structural damping and a better understanding of joint dynamics.

- Damping from the support structure is generally small.
- Passive damping sources generally are preferred.
- Joints will be a source of damping.
- Joints with gaps make dynamic behavior harder to predict.

Predicting damping in large space structures can be difficult.

- Difficult or impossible to test full scale structures on the ground.
- Ground test results of components may be affected by:
  - gravity
  - air
  - temperature
  - scale
- Analytical methods of predicting damping need improvement.
- Ground tests have shown that gravity effects joint damping.

A database of in-orbit and on-ground tests would be helpful:

- Providing qualitative information an important design variables.
- Assisting in improving analytical models of joint damping.

Current Understanding of Joint or Connection Damping

Pinned or bolted structures typically have more damping than welded structures.

Damping is typically amplitude and frequency dependent.

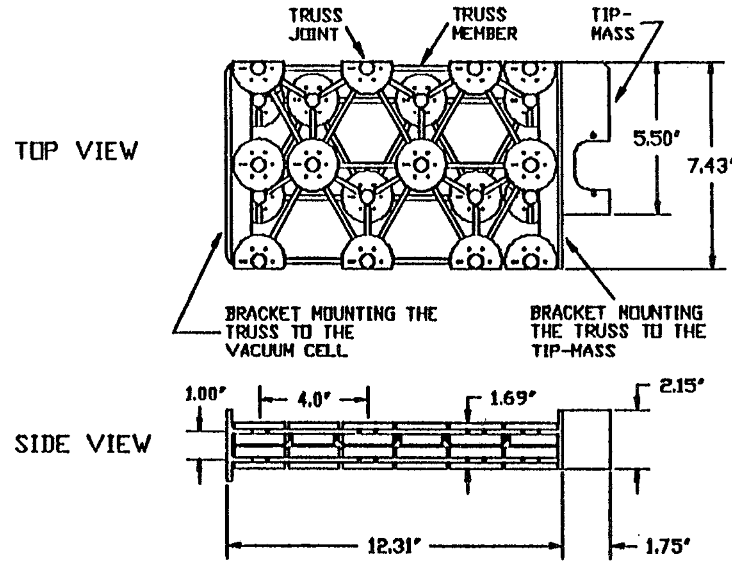
Common Mechanisms of Passive Damping in Joints or Connections:

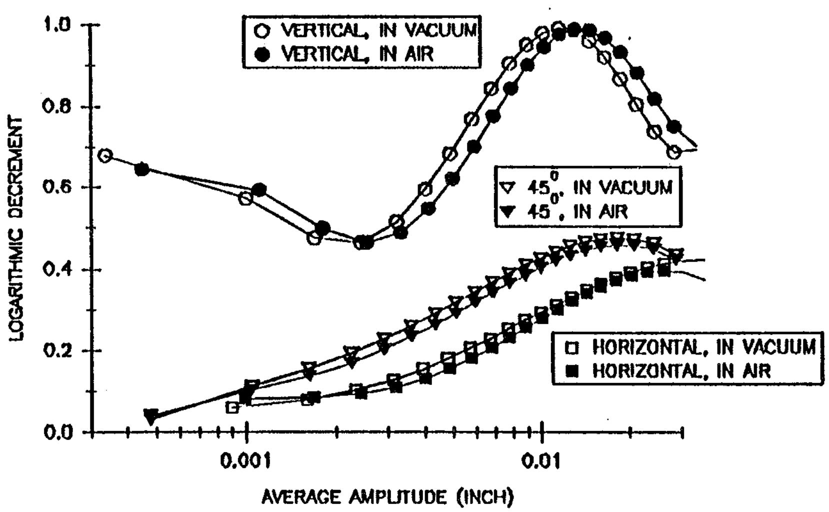
- Air Damping (not present in space)
- Material Damping ($\zeta<0.001$ for most metals at room temperature)
- Coulomb Friction:
  - Macroslip:
    - Can be a large source of damping.
    - Dependent on friction coefficients and joint loads.
    - Damping contributions can be inferred from joint pull tests.
    - Analytical models are available.
  - Microslip:
    - Damping is less than Macroslip damping.
    - Difficult to predict.
- Impact Damping:
  - Implies a gap is present.
  - Generally believed to be more important at higher frequencies (>1 Hz.).
  - Difficult to predict but characterized by the coefficient of restitution.
  - Difficult to separate from Coulomb Friction damping.

Previous Work Done at USU - Prior to Phase A

An experiment has been constructed to measure damping of a tetrahedral truss with pinned joints.

- **Developed on a very small budget.**
- **Demonstrated gravity dependent damping**





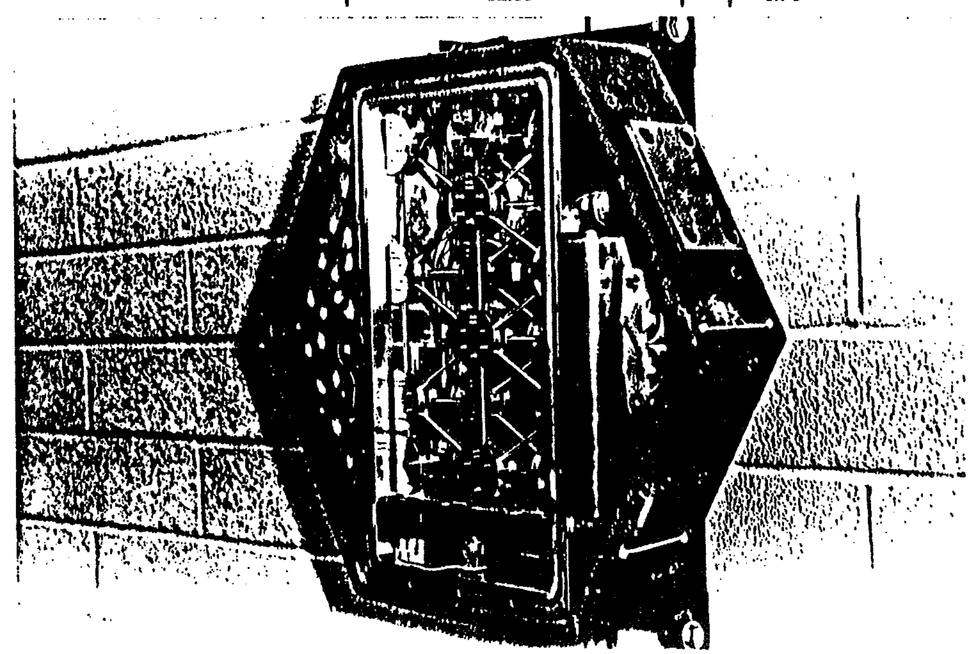
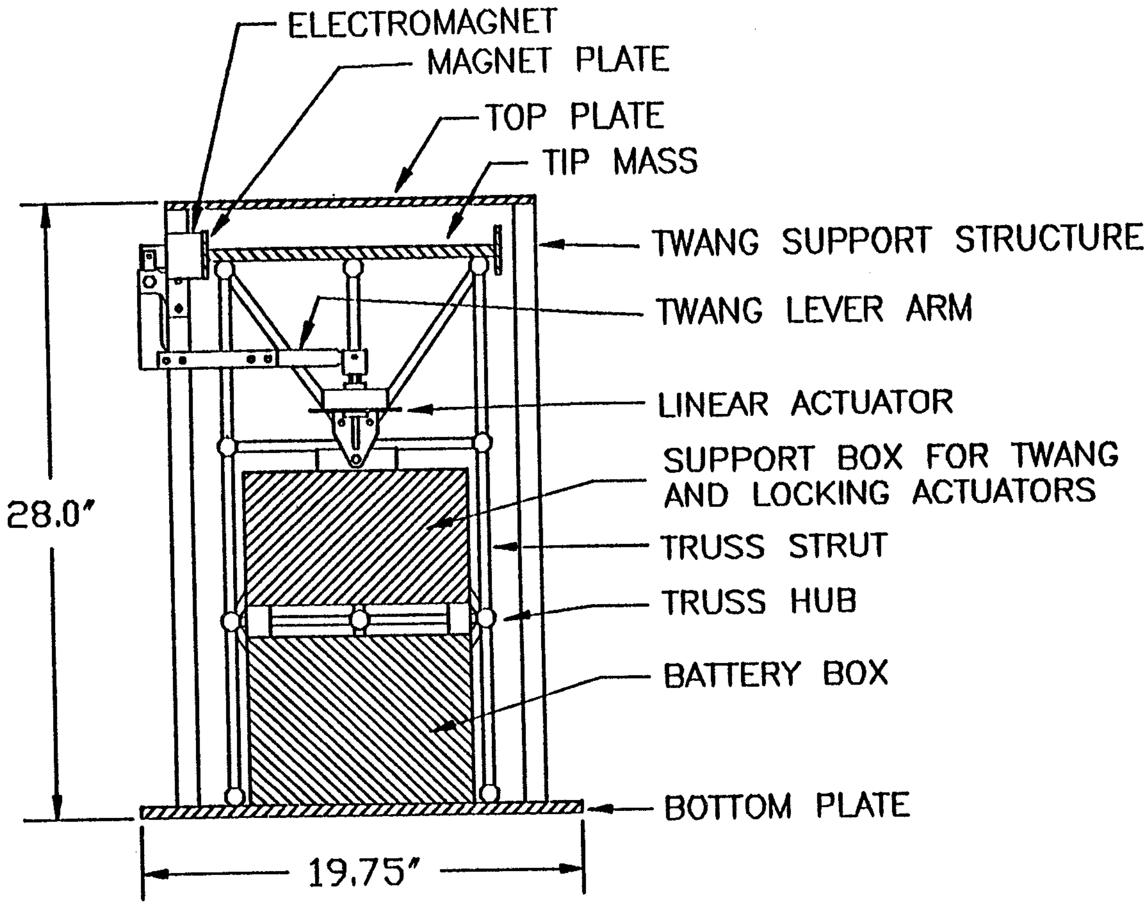
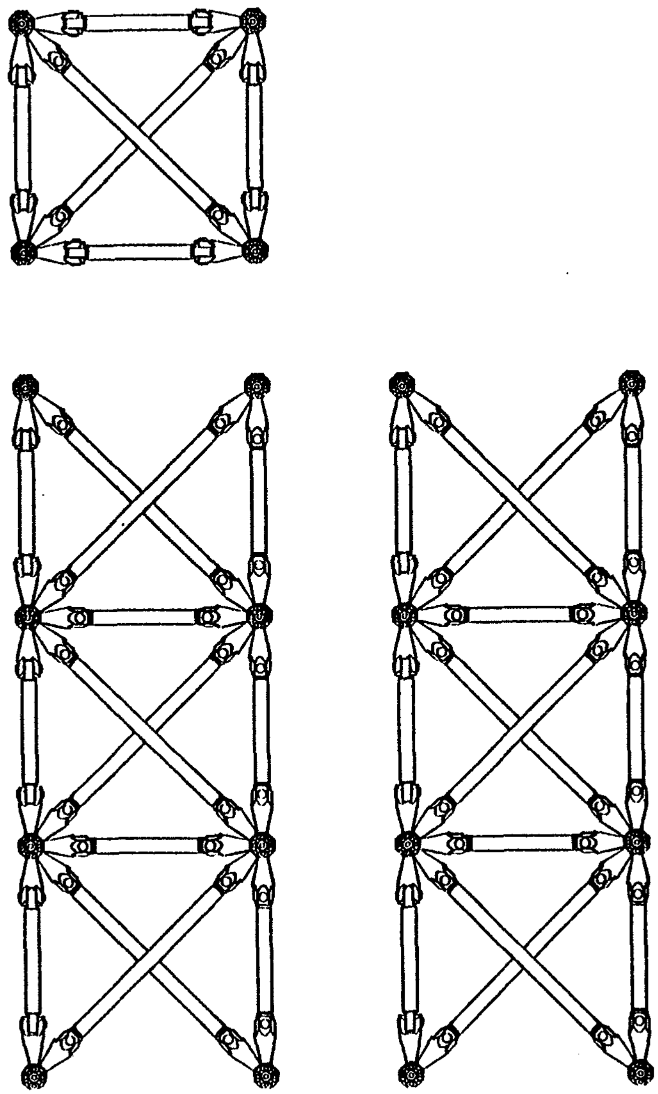


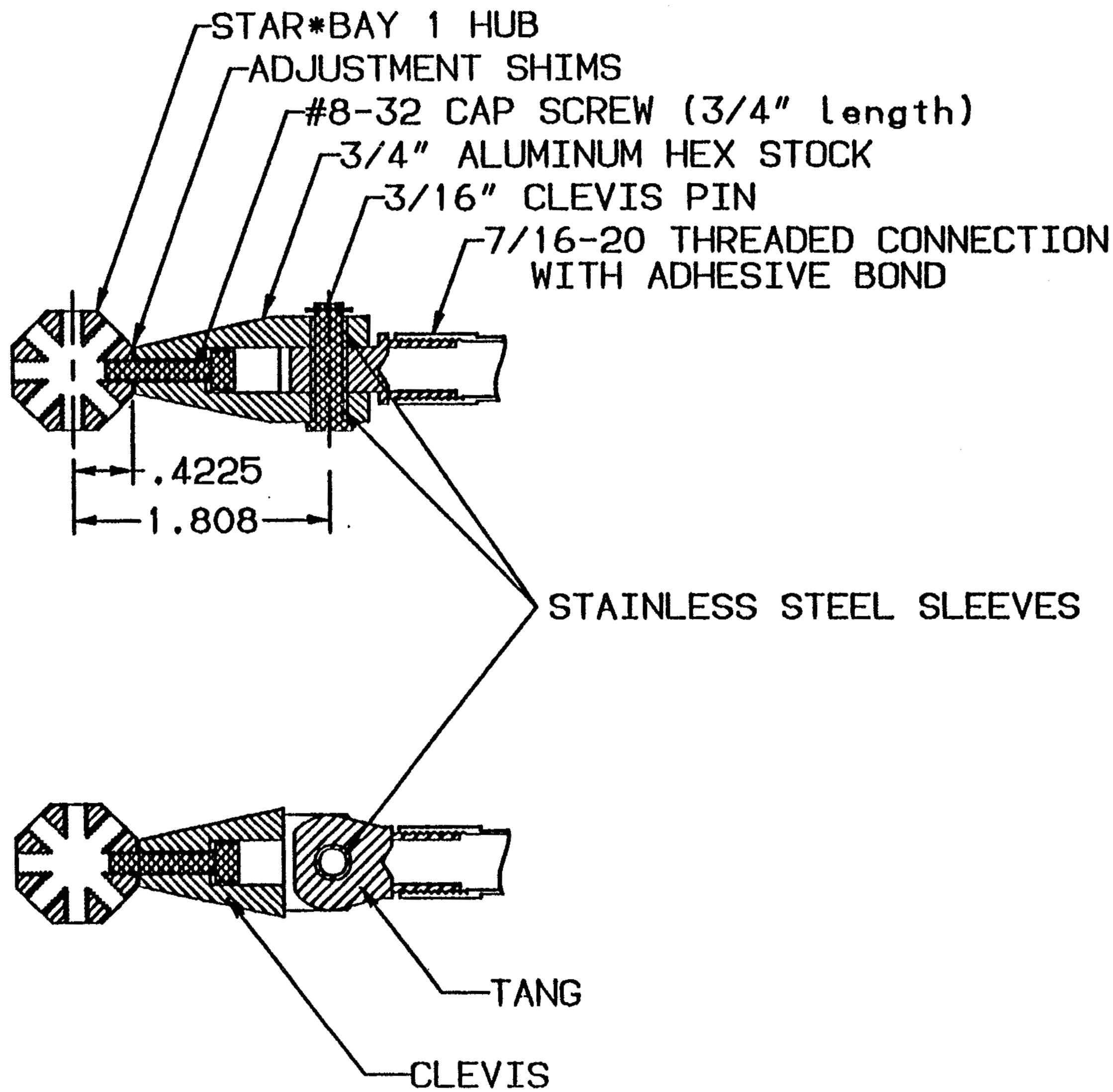
Illustration of the Experiment Layout



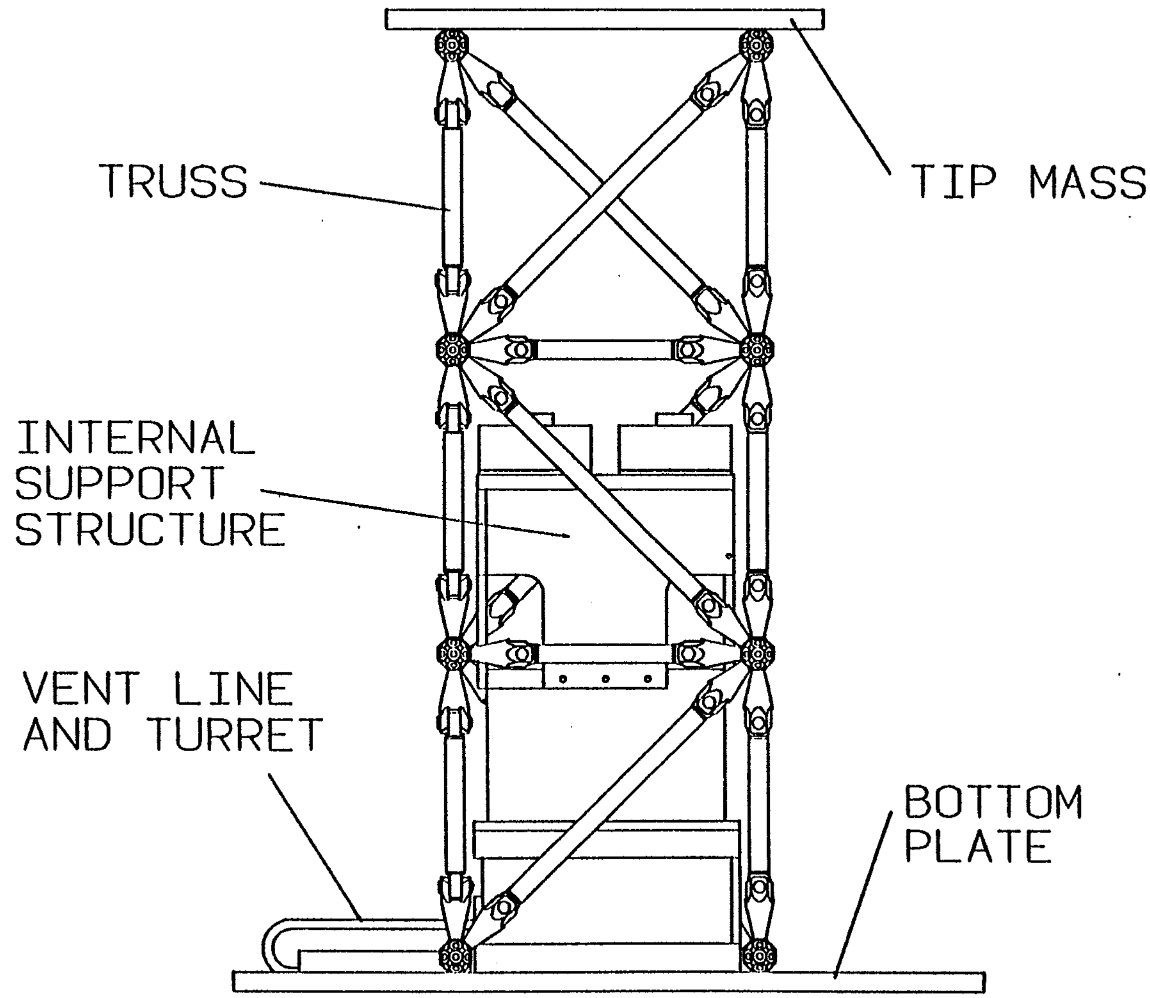
Strut Arrangement



Joint Design



<u>Truss, Battery Box, and Bottom Plate</u>

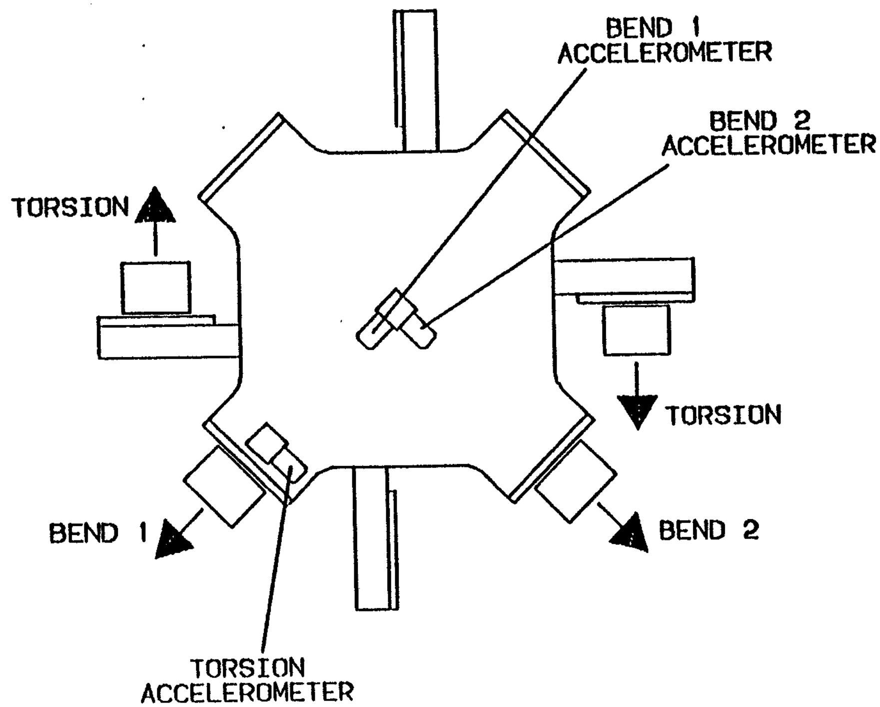


Twang Method of Excitation

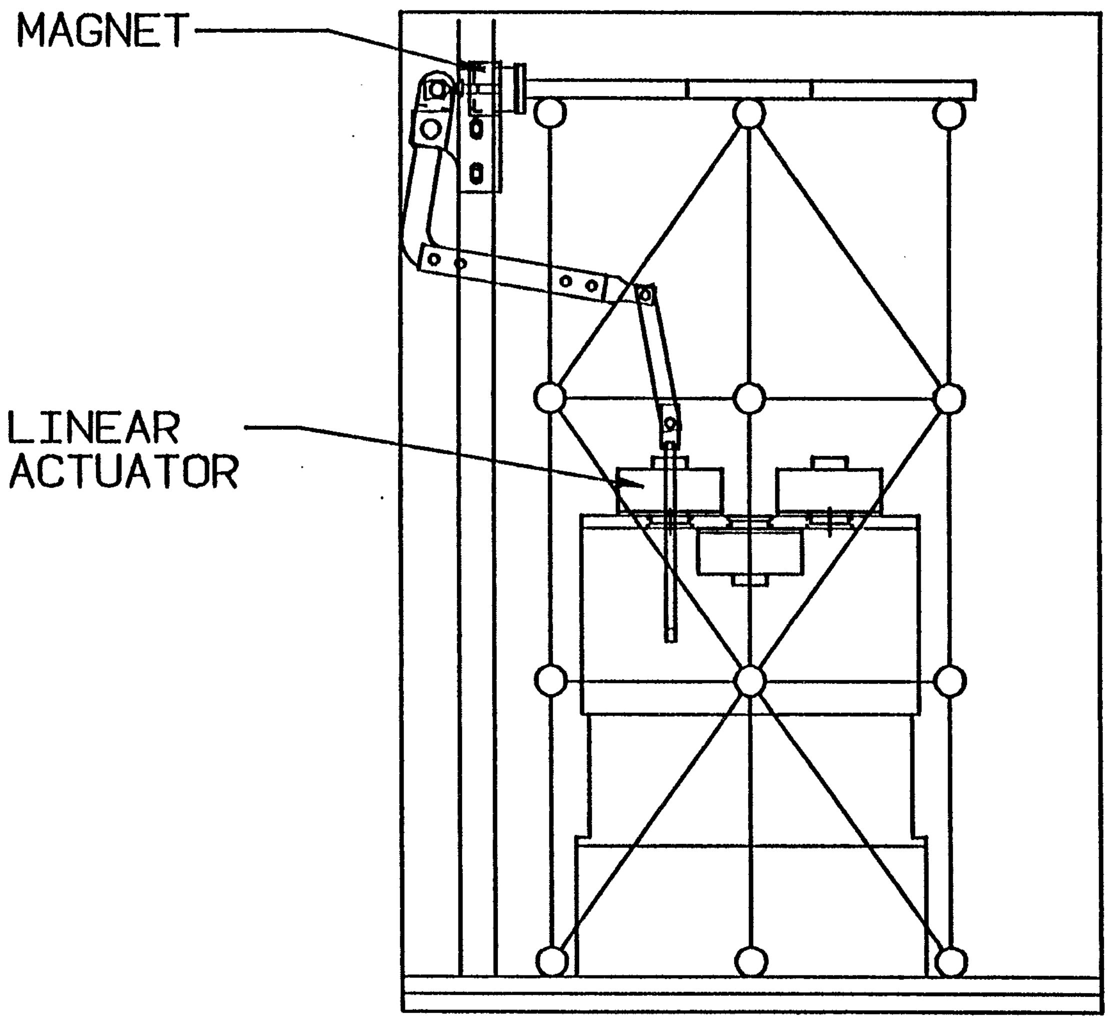
The twang excitation method is accomplished by a linear actuator/lever arm/electromagnet assembly.

- A magnet is moved into contact with a magnet plate on the truss tip mass.
- The electromagnet is then energized and pulls the truss from its neutral position.
- The power is removed from the electromagnet, the truss is released, and the decay of oscillations is recorded.
- Two bending modes and a torsional mode excitation provided.

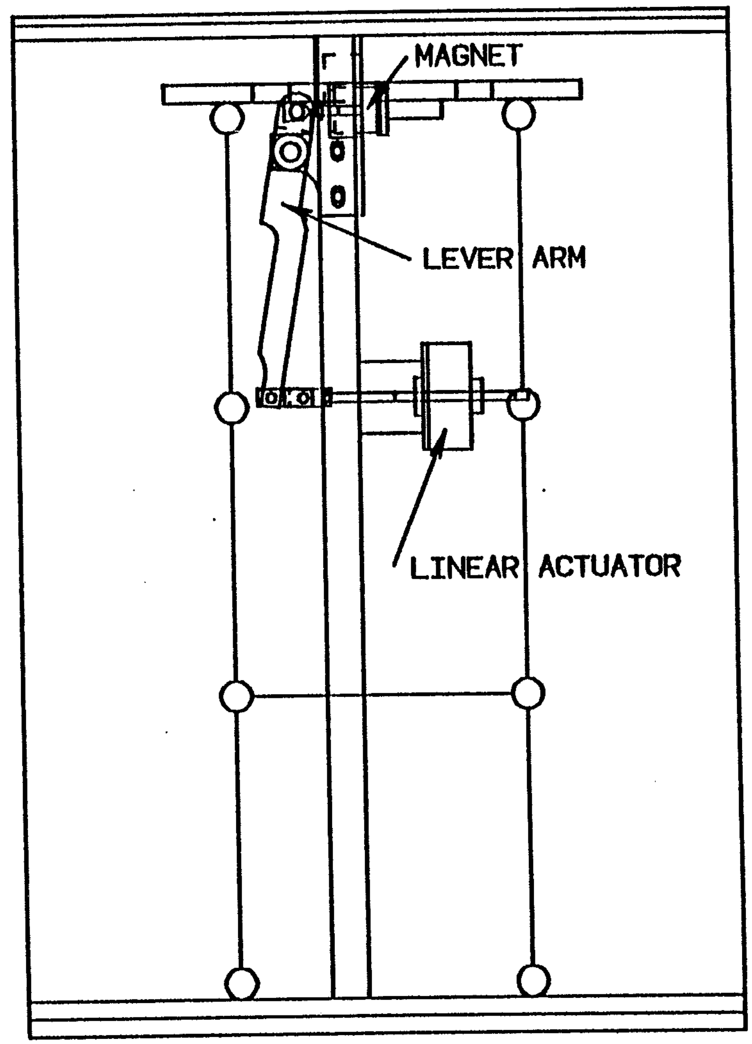
Top View of the Truss Tip Mass



Bend Excitation Side View

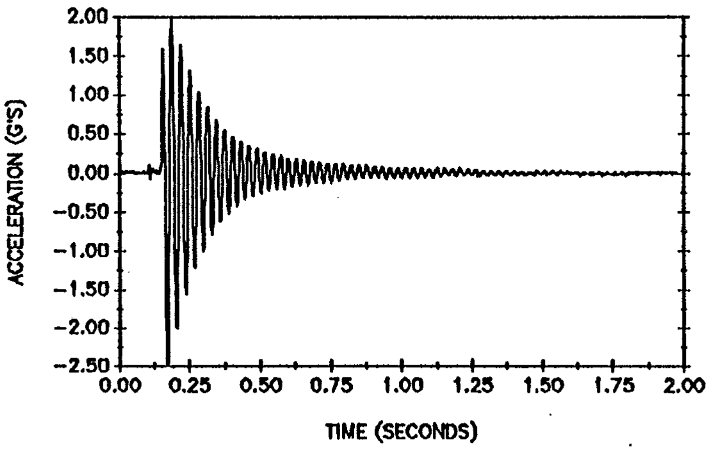


Torsion Excitation Assembly

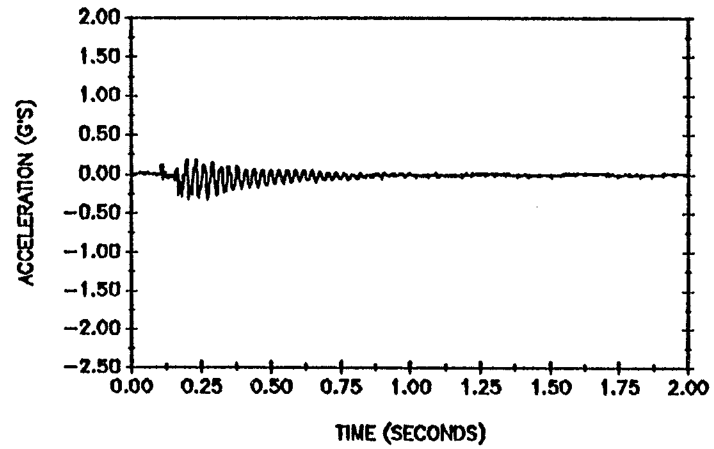


Ground testing of the Experiment Prototype

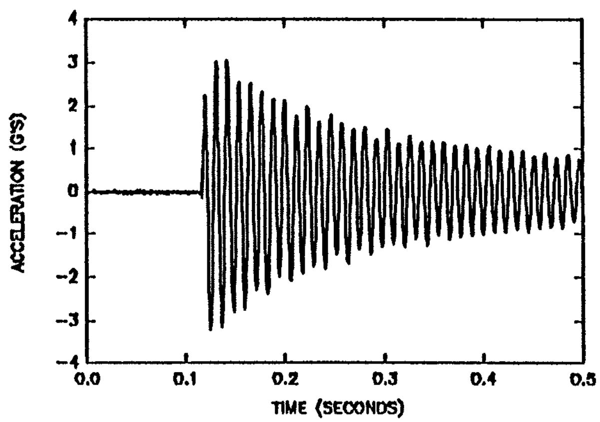
Twang tests of the truss were conducted.



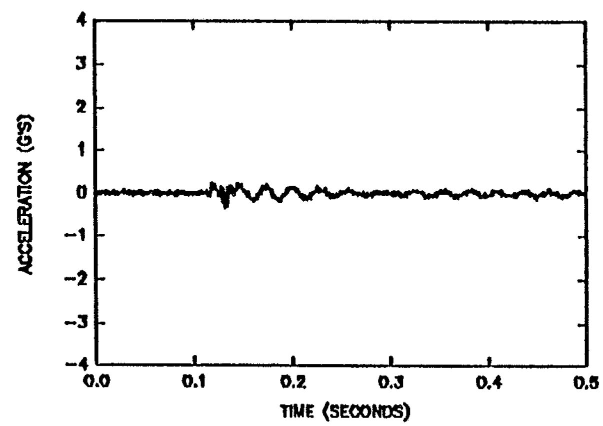
Acceleration in the direction of the first bending mode.



Acceleration in the opposite direction of the first bending mode.



Acceleration time history for a torsional mode twang test.



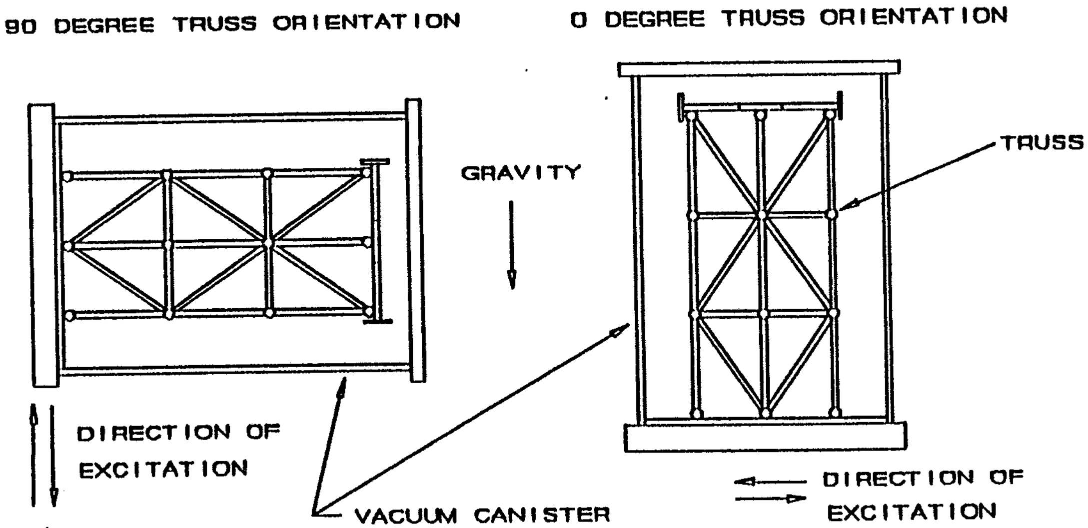
Acceleration at the tip-mass center during a torsional mode twang test.

Ground testing of the Experiment Prototype

Twang, Random Vibration, and Sine Sweep Tests of the truss.

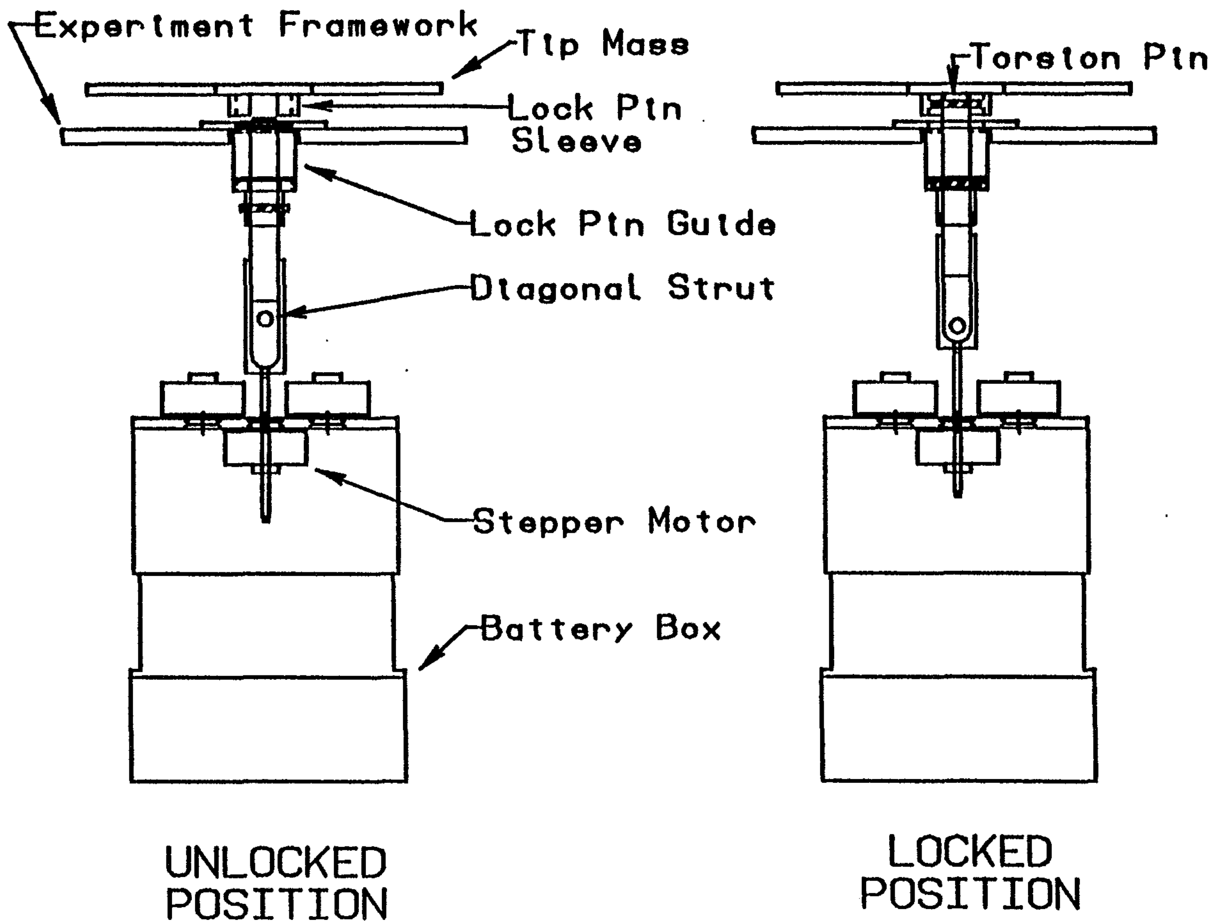
- **Experiment mounted inside a can:**
  - **Allows testing in vacuum.**
  - **Provides stiff mounting for base excitation.**

- **Tests conducted at different orientations to examine gravity dependence of damping.**



Tip Mass Locking Mechanism

- Lock mechanism provides support during launch and reentry to minimize joint wear.
- The truss design can withstand launch and reentry design loads in case the locking mechanism fails to operate.



Experiment Controller/Data Acquisition System

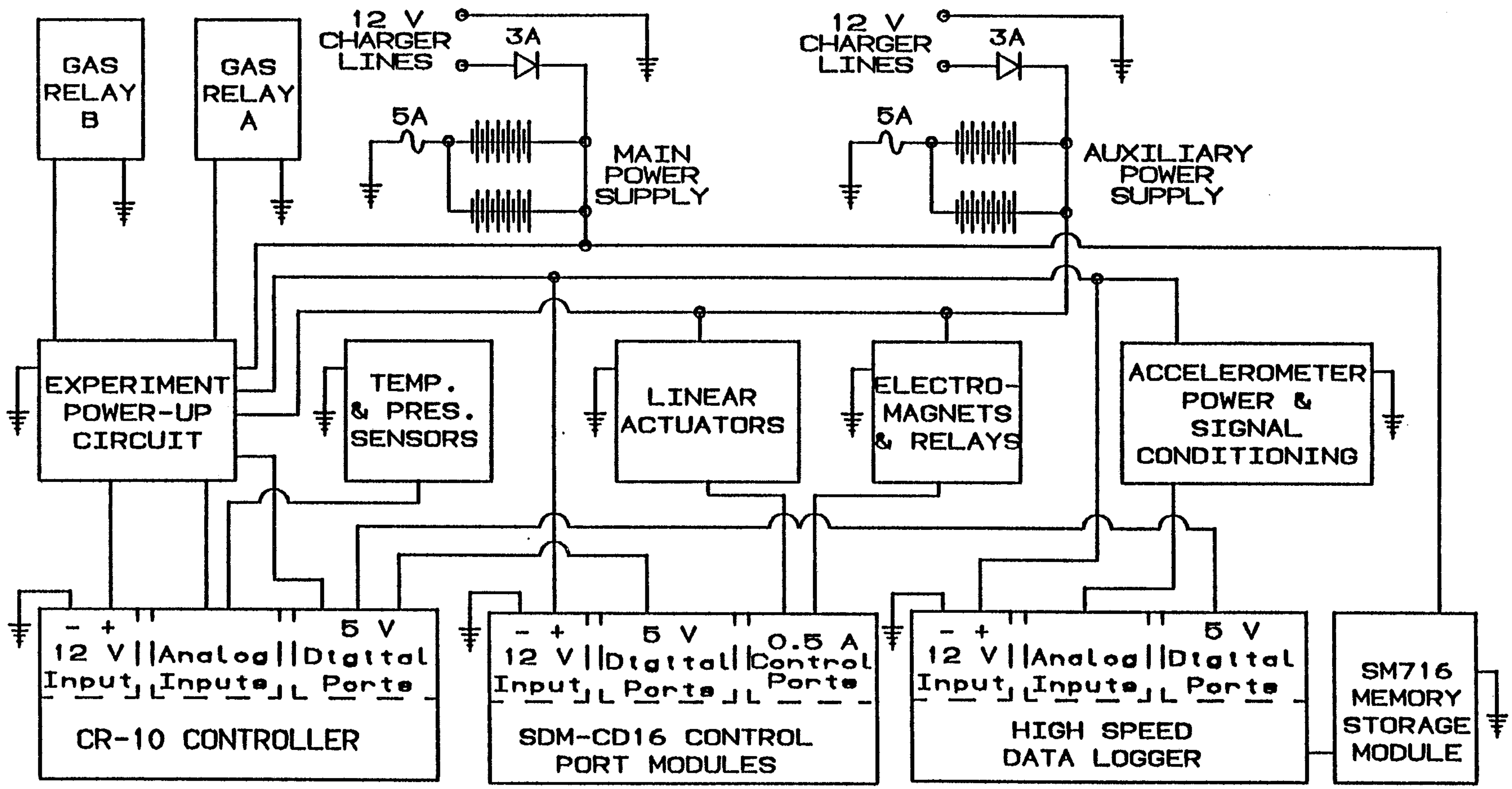
Campbell Scientific CR-10 controller/datalogger

- Will be used to control actuators and magnets and monitor temperature and pressure.
- Low power consumption (0.5 mA quiescent, 35 mA during measurements @ 12 V)
- 64K EEPROM for program and data.
- Loads program from EEPROM on Power-up.
- Easily programmed.
- Uses a Campbell Scientific Control Port Module (SDM-CD16) for control of 32, 0.5 A circuits.
- Powers-up a High Speed Data Logger for twang testing.

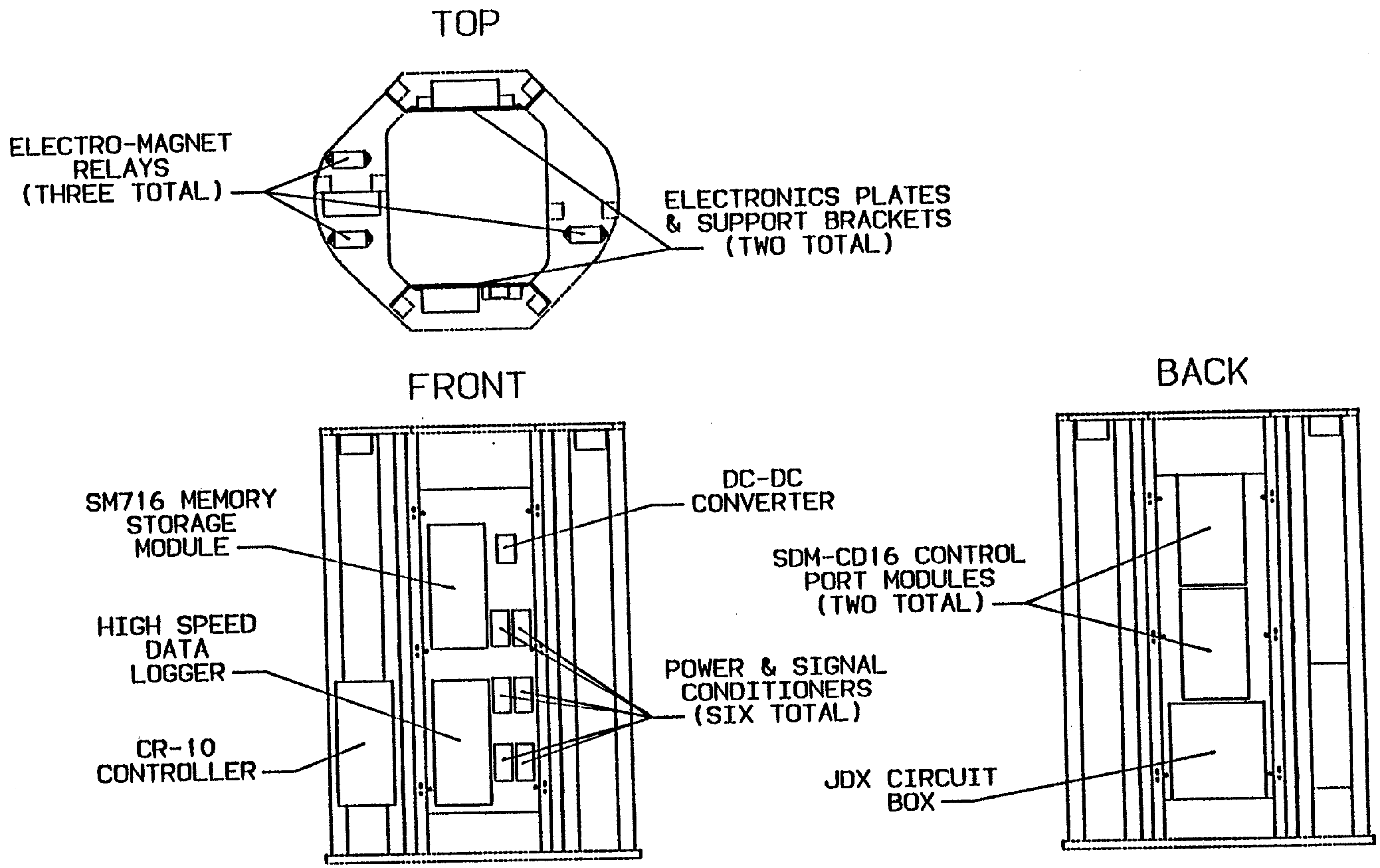
Campbell Scientific High Speed Data Logger

- 16 bit A/D and 100,000 sample/sec capacity.
- Power consumption: 0.13 Amp @ 12 V
- 512 K EEPROM for program and memory and 512 K RAM
- Storage of 358K values using a Campbell Scientific Memory Module (SM716)

JDX Top Level Wiring Diagram



Mounting of Electrical Components



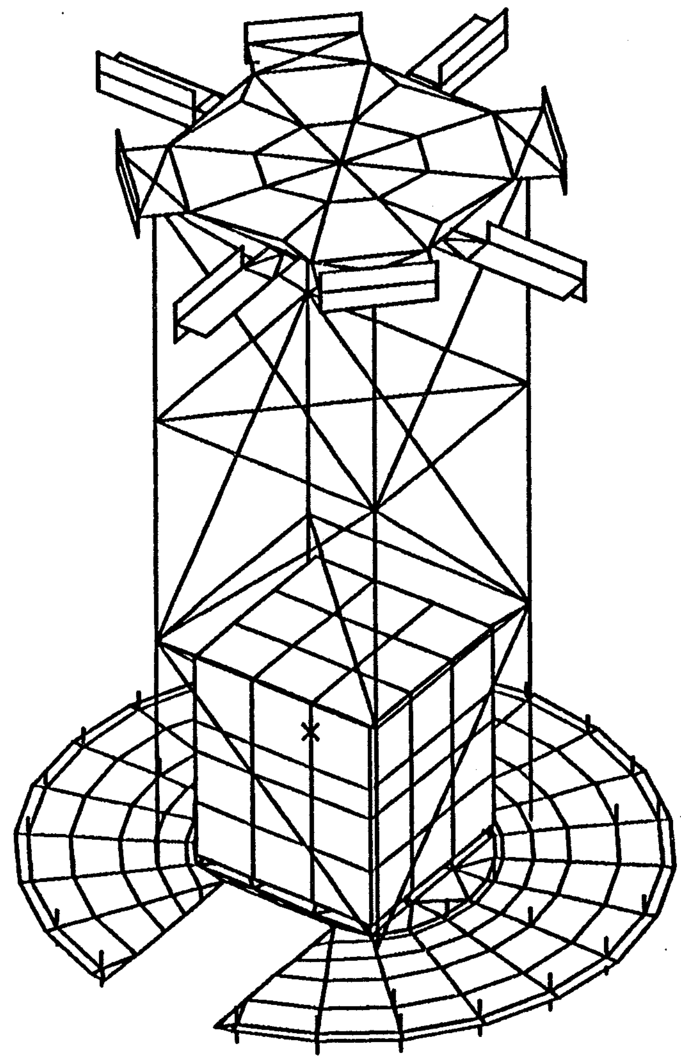
Structural Analyses of the Truss and Base Plate

Linear Static Analysis

- Design Accelerations: ±11.0 g's in X and Y and Z.
- Safety Factor against yield > 2

Predicted Resonant Frequencies (ignoring joint gaps)

Mode Number	Natural Freq. (Hz.)	Mode Description
1	46.8	Bending mode
2	51.3	Bending mode
3	110.1	Torsional mode



JDX Mission Operational Plan

Experiment Activation:

- Powered up at 50,000 feet by the baroswitch attached to APC relays.
- Unlock the truss during the first hour before significant cooling of the experiment occurs.
- The controller will monitor its built-in clock, the GAS relay switches A and B, and the battery box temperature.

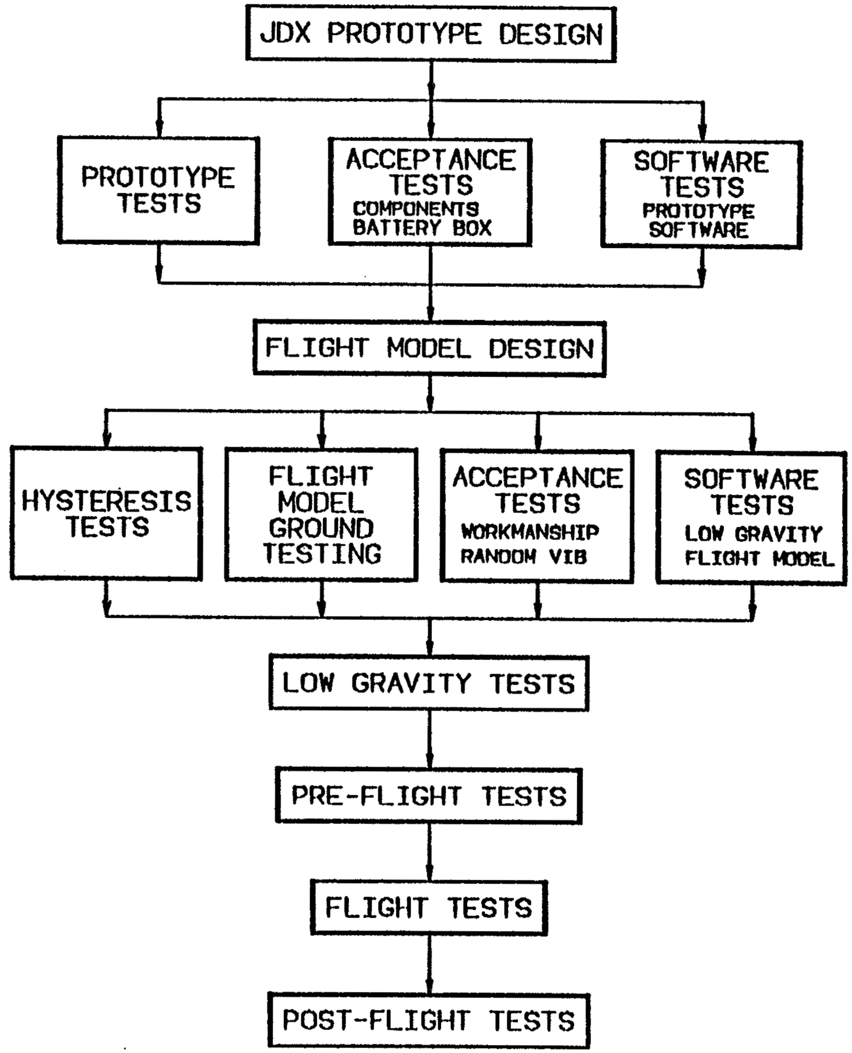
Experiment Execution:

- JDX will begin the twang test sequence when the first of the three following events occur:
  (1) Relay B is manually activated by the crew indicating a period of orbiter free drift,
  (2) the temperature of the experiment drops below a lower limit value (TBD), or
  (3) 18 hours has passed since closure of the APC baroswitch.
- Begin testing by:
  (1) Move all electromagnets to their preset stop positions,
  (2) Perform approximately 10 twang tests for each mode shape,
  (3) Record experiment temperature and air pressure during the tests, and
  (4) Lock the truss by activating a linear actuator.

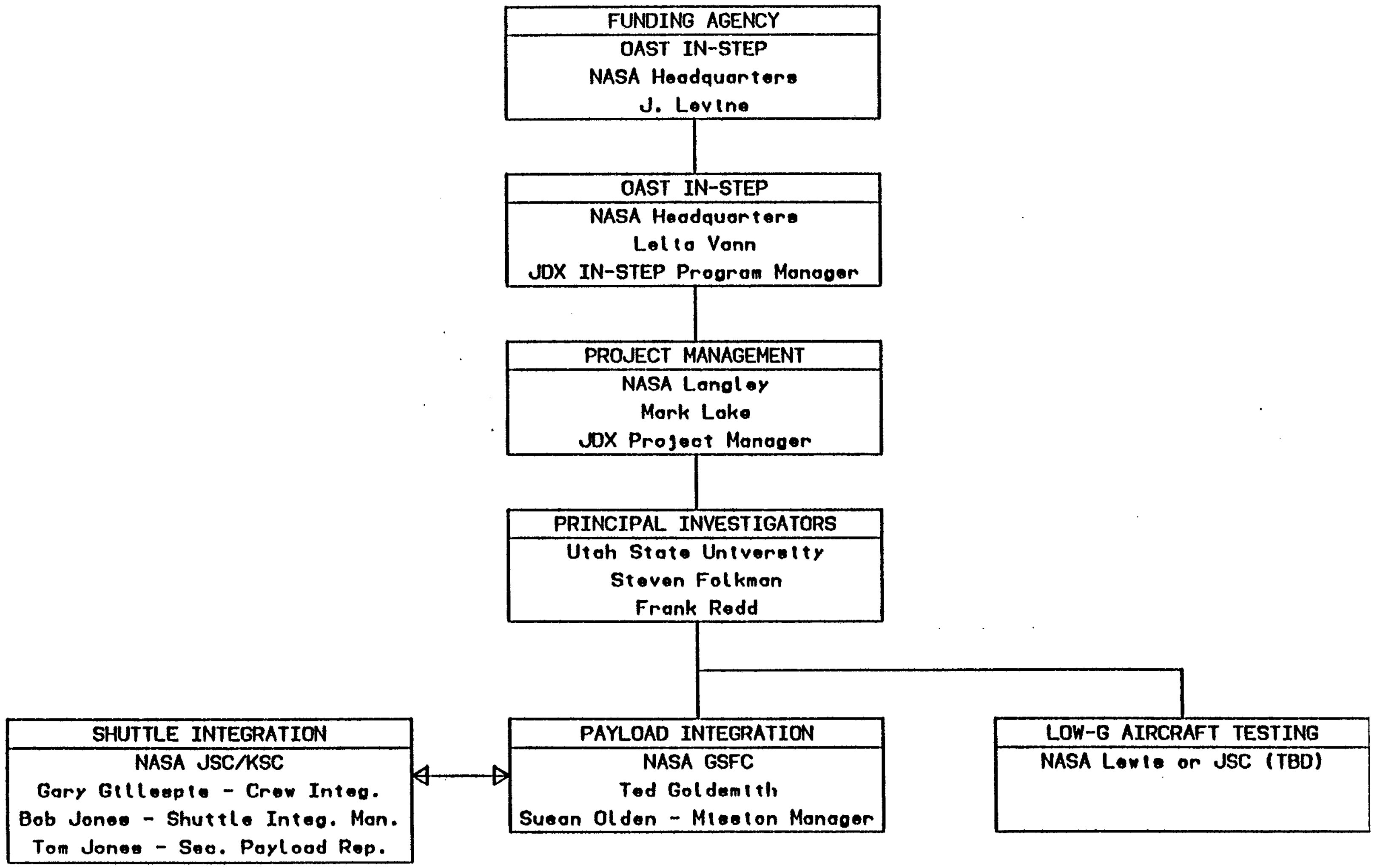
Experiment Deactivation:

- One hour after crew activation of Relay B, the crew will set Relay B to latent.
- Lock the truss (if it is not already locked).
- Shut down all experiment activities except the monitoring of experiment temperatures.
- Prior to the end of the mission, the crew will set Relay A to latent, thus powering down JDX.
- In the event of unsuccessful deactivation of Relay A, baroswitch opening at 50,000 feet during orbiter entry will power down the controller.

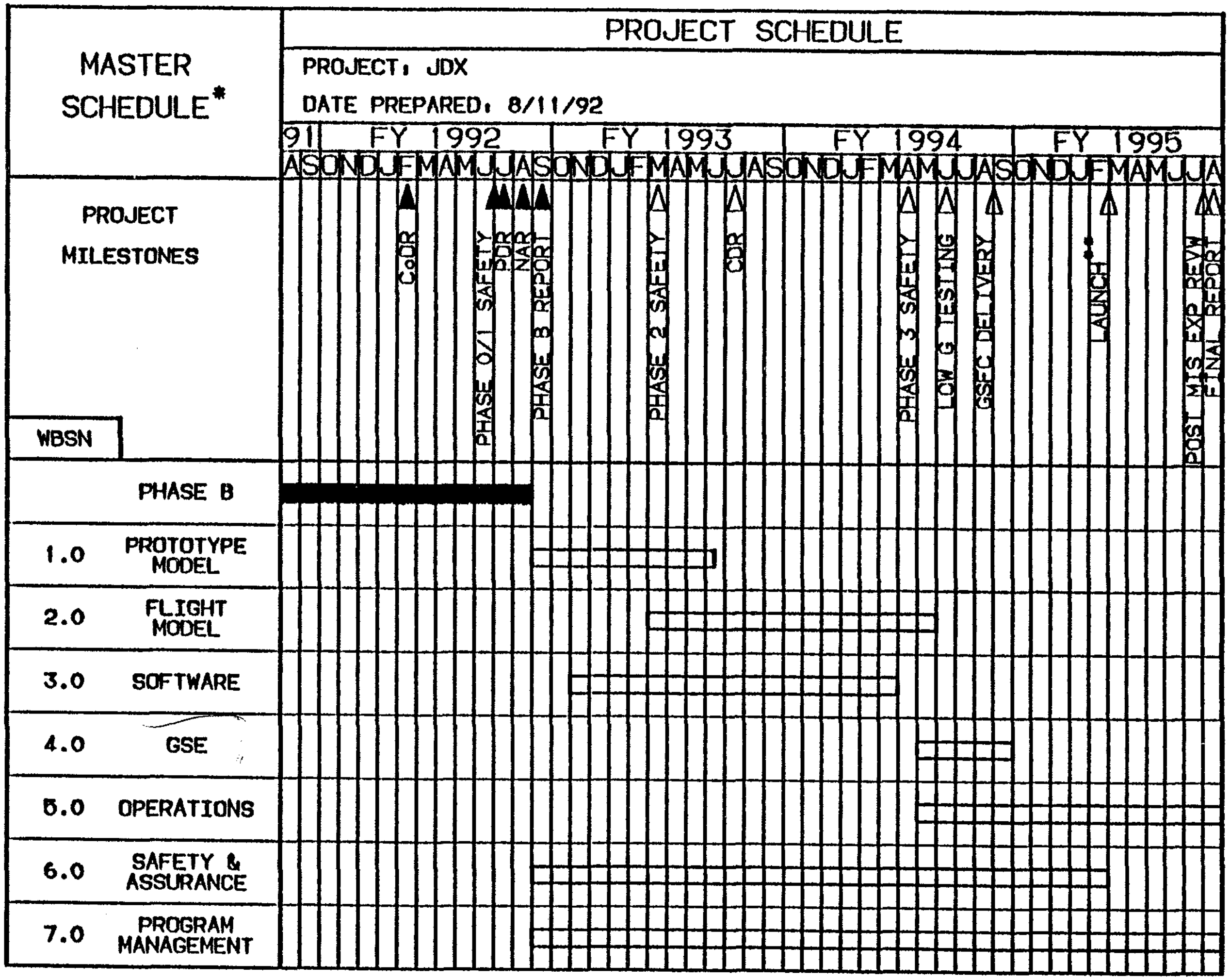
JDX Testing Flow Chart for Phase C/D



JDX Phase C/D Organization



JDX Schedule



* PHASE C/D IS ASSUMED TO BEGIN SEPTEMBER 1, 1992

** PER CONTRACT SPECIFICATION, A SIX MONTH TIME PERIOD BETWEEN HARDWARE DELIVERY TO GSFC AND LAUNCH ASSUMED. NORMAL PAYLOAD PROCESSING WOULD GIVE A LAUNCH DATE OF JANUARY 1, 1996

PHILLIPS LABORATORY

DIRECTORATE OF SPACE AND MISSILES TECHNOLOGY

ADAPTIVE STRUCTURES FLIGHT EXPERIMENTS

CAPT MAURICE MARTIN

NASA/DOD Flight Experiments
Technical Interchange Meeting
Monterey, CA
October 5-9, 1992

ADAPTIVE STRUCTURES FLIGHT EXPERIMENTS

1. ADVANCED CONTROLS TECHNOLOGY EXPERIMENT (ACTEX)

2. ADVANCED CONTROLS TECHNOLOGY EXPERIMENT II (ACTEX-II)

3. STRV-1B CRYOCOOLER VIBRATION SUPPRESSION FLIGHT EXPERIMENT

4. PRECISION OPTICAL BENCH (PROBE)

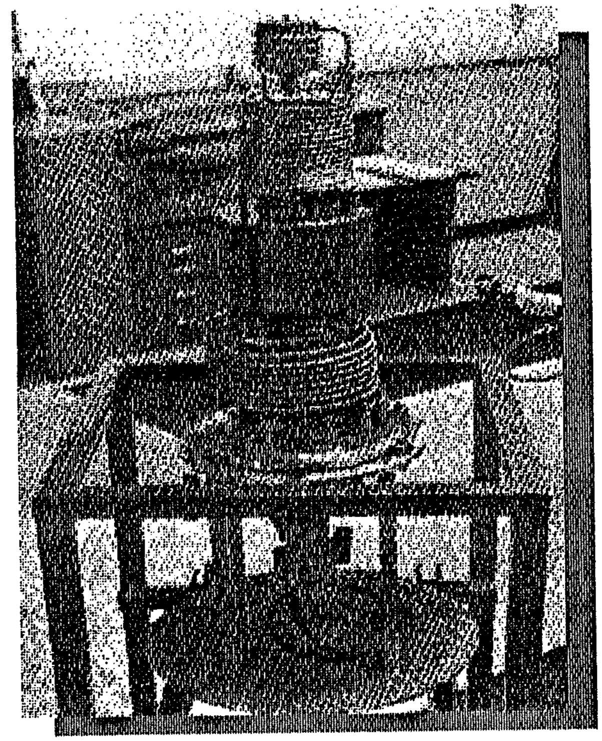
5. OTHER SDIO FLIGHT PROGRAMS
   - MODULAR CONTROL PATCH
   - ADVANCED COMPOSITE STRUCTURAL COMPONENTS FOR CLEMENTINE
   - TECHSAT ALL-COMPOSITE SPACECRAFT

6. INEXPENSIVE STRUCTURES AND MATERIALS FLIGHT EXPERIMENT (INFLEX)

A0802.02

Enhanced Resolution Using Active Vibration Suppression....

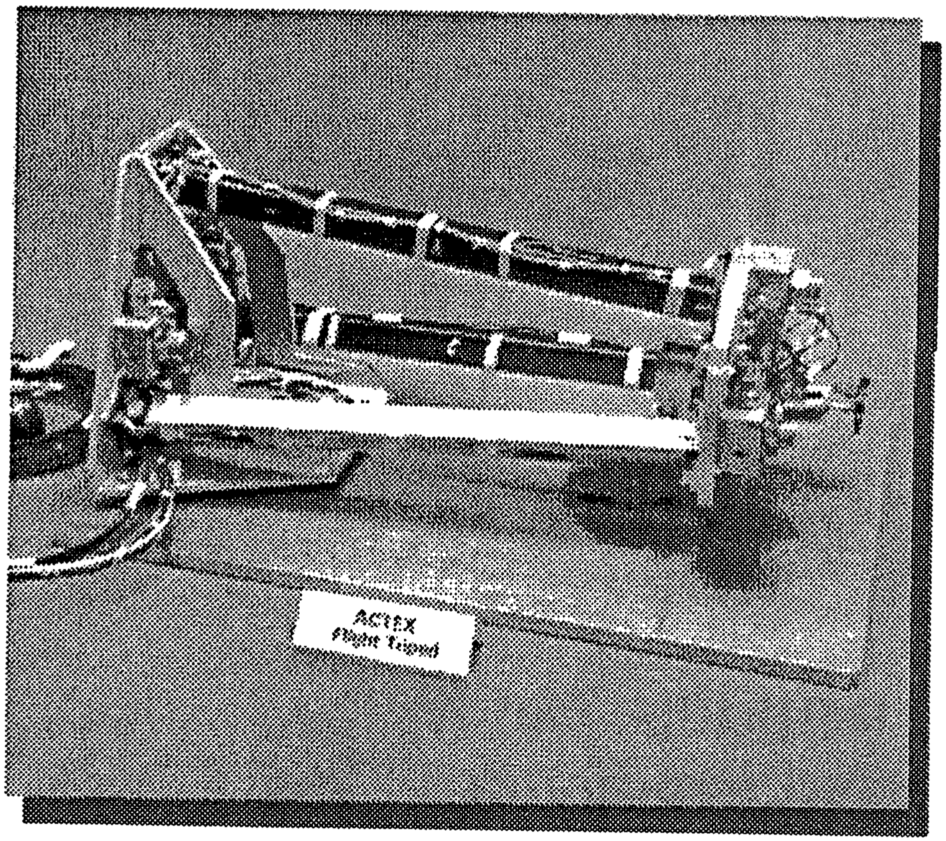
PIXEL SMEARING DUE TO JITTER



ENHANCED IMAGE USING ACTIVE VIBRATION SUPPRESSION

B1182.01b

Advanced Controls Technology Experiment (ACTEX)



OBJECTIVE

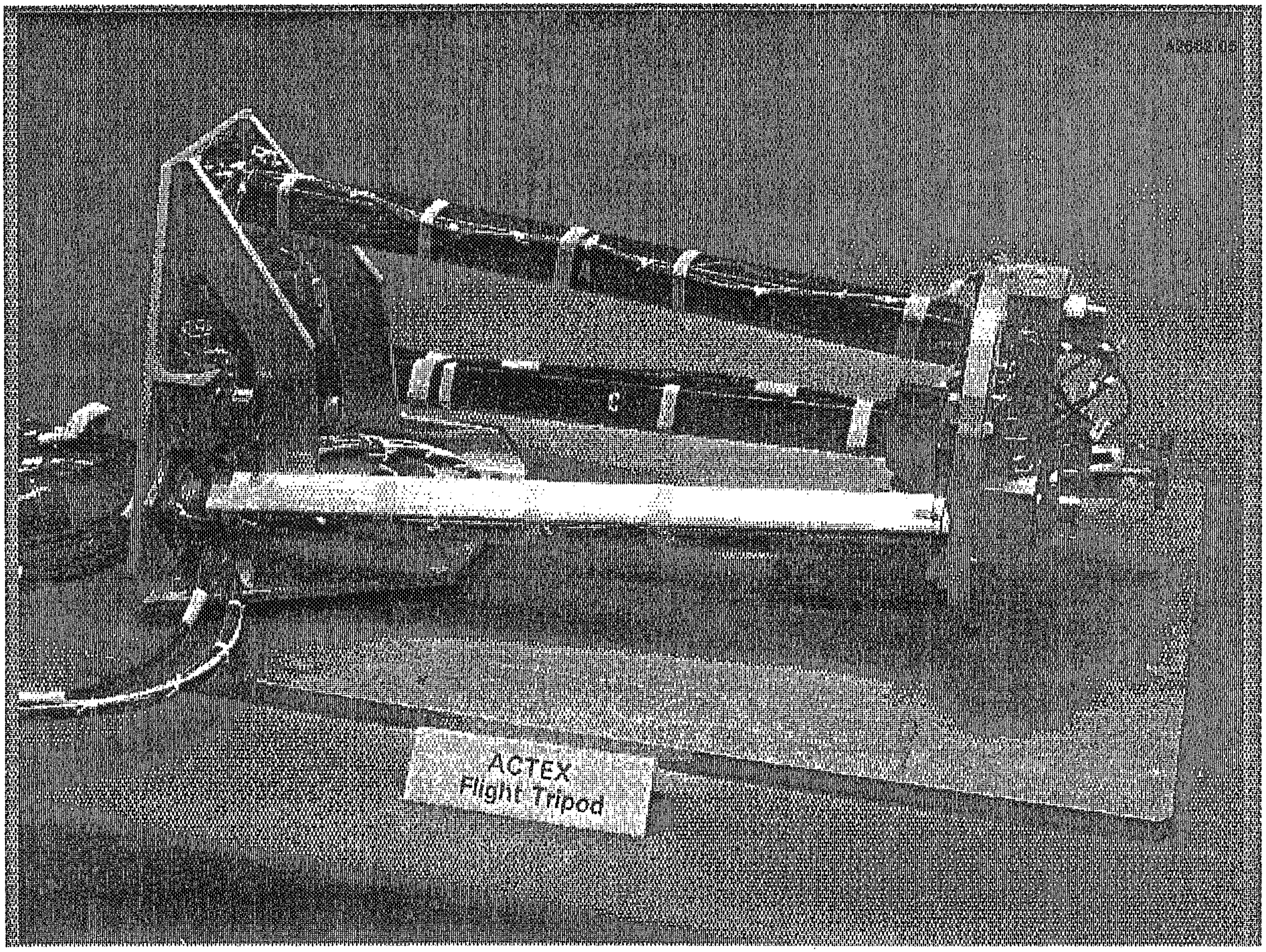
On-Orbit Demonstration of Embedded Piezoceramic Sensors and Actuators for Active/Passive Vibration Suppression

DESCRIPTION

- 1 ft x 1 ft x 2 ft Tripod Structure
- Piezos for Active Control Layered in 1 inch Advanced Composite Tubes
- Passive Damping Using Piezos with Resistor Shunt
- On-Orbit System ID/Structural Characterization
- Dynamic Change Mechanism with On-Orbit Adaptive Control
- Launch Restraint Using Nitinol Non-Pyrotechnic Release Device

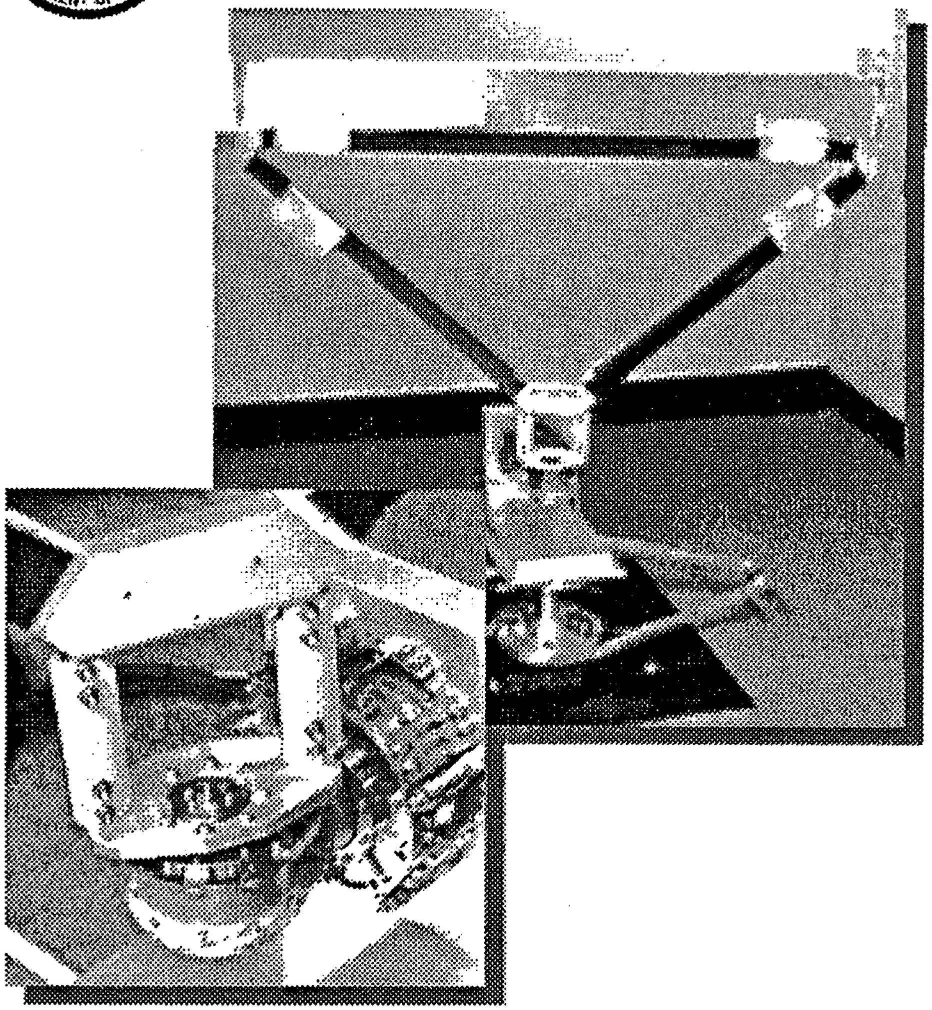
ACTEX PROGRAM STATUS

- PROGRAM FULLY FUNDED BY SDIO
- TRW HAS COMPLETED EXPERIMENT FABRICATION
- EXPERIMENT DELIVERED TO NAVAL RESEARCH LABORATORY IN AUGUST 1992 FOR SPACECRAFT INTEGRATION
- LAUNCH ANTICIPATED IN 1994


ACTEX
Flight Tripod

B1182.01a

Advanced Controls Technology Experiment II (ACTEX-II)



OBJECTIVE

System Application of Piezoceramic Sensors and Actuators to Damp Solar Array Vibrations

DESCRIPTION

- Solar Array Yoke with Embedded Piezoceramic Sensors and Actuators
- 6 ft x 2.5 ft Simulated Solar
  - Deployable Aluminum Framework
  - Modal Frequencies of 0.5-10 Hz
- Vibration Suppression Using Digital Control Electronics
- Electronics Miniaturized Into Multichip Module Mounted on Yoke
- Advanced Solar Array Drive Motor with Viscoelastic Damped Interface
- On-Orbit System ID/Structural Characterization

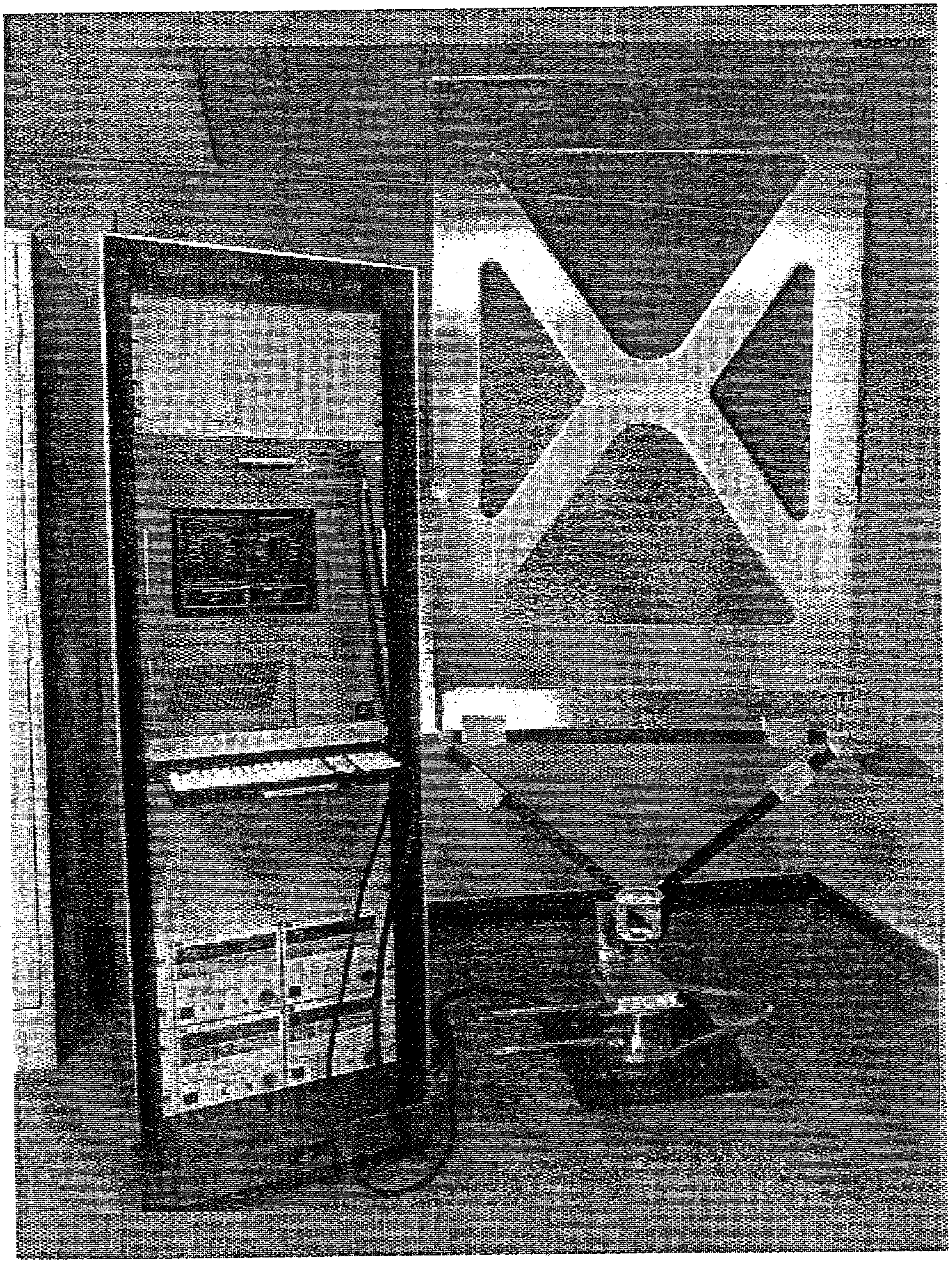
ACTEX-II PROGRAM STATUS

- PROGRAM FULLY FUNDED BY SDIO

- TRW IS FINALIZING FLIGHT HARDWARE DESIGN

- EXPERIMENT TO BE DELIVERED IN EARLY 1994 FOR INTEGRATION ON STEP-3 SPACECRAFT

- LAUNCH ANTICIPATED IN EARLY 1995

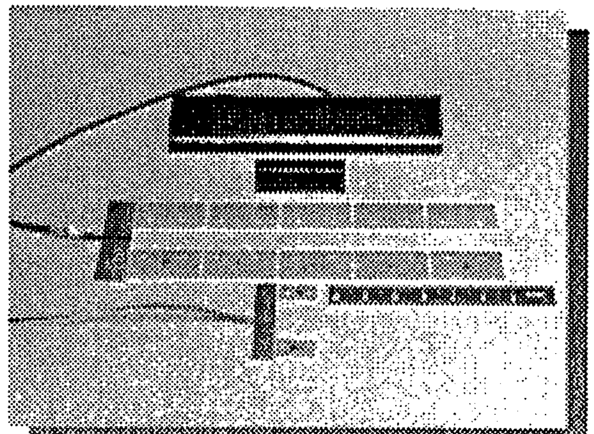


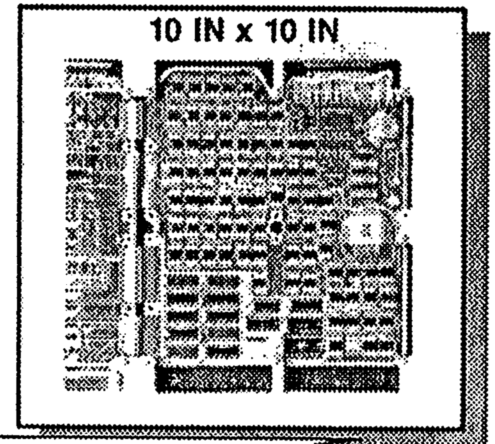
A1271.04

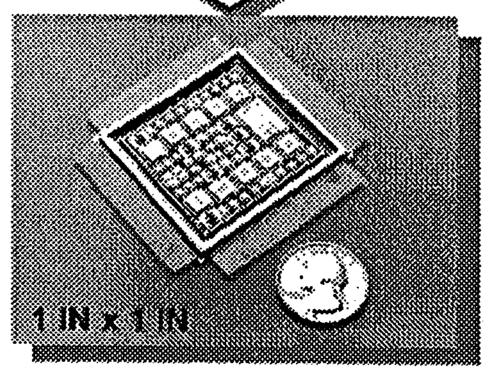
Modular Control Patch

OBJECTIVE

Develop a Miniaturized, Modular Vibration Suppression System Having Sensing, Actuation, and Control/Power Conditioning Components Integrated into a Self-Contained Package



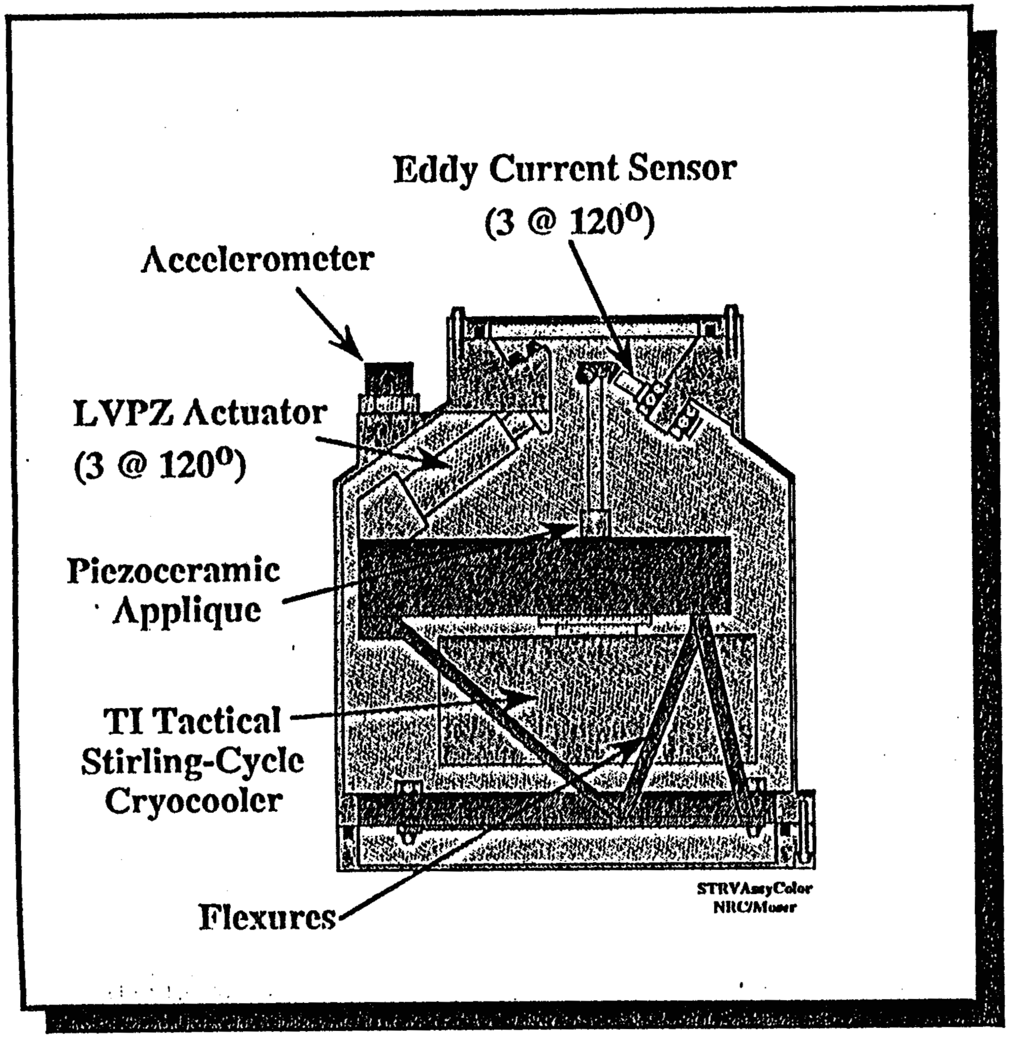




PAYOFF

Miniaturized, Lightweight, Retrofitable Vibration Suppression System

STRV-1b Cryocooler Vibration Suppression Experiment



OBJECTIVE

Vibration Suppression of Cryocooler Cold Finger Using Active Control Technologies

DESCRIPTION

- Stirling-Cycle Cryocooler Traceable to SDI Class Systems
- Piezo Stack Actuators for for 3-Dimensional Control of Cryocooler
- Actuation Using Piezo Applique Bonded to Base of Cold Finger
- Eddy Current Transducer to Measure Cold Finger Tip Motion
- Integrated Digital and Analog Control Electronics

STRV-1B PROGRAM STATUS

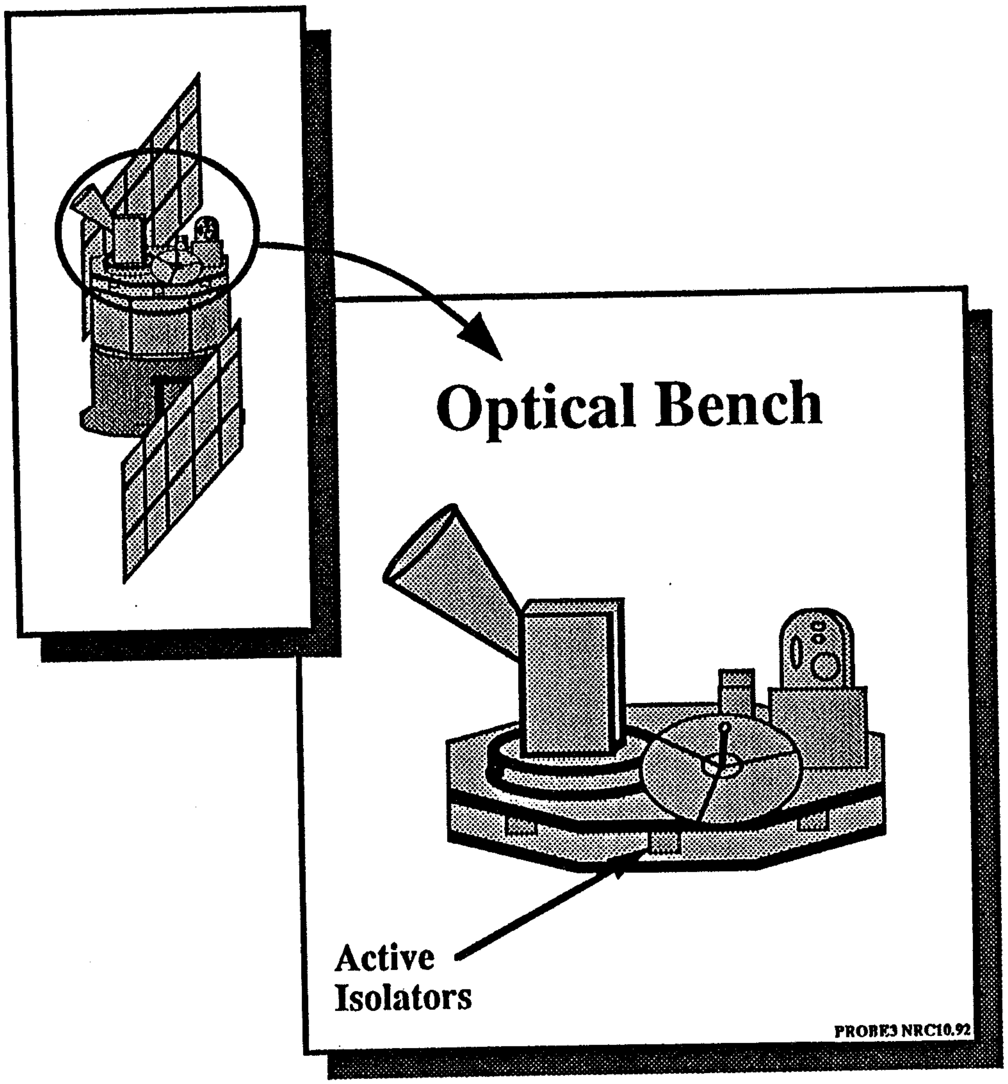
- PROGRAM FULLY FUNDED BY SDIO

- EXPERIMENT FABRICATION IN PROGRESS AT THE JET PROPULSION LABORATORY

- EXPERIMENT TO BE DELIVERED TO THE ROYAL AEROSPACE ESTABLISHMENT EARLY 1993 FOR SPACECRAFT INTEGRATION

- ARIANE LAUNCH ANTICIPATED IN MID 1994

PROBE
PRECISION OPTICAL BENCH EXPERIMENT



OBJECTIVE

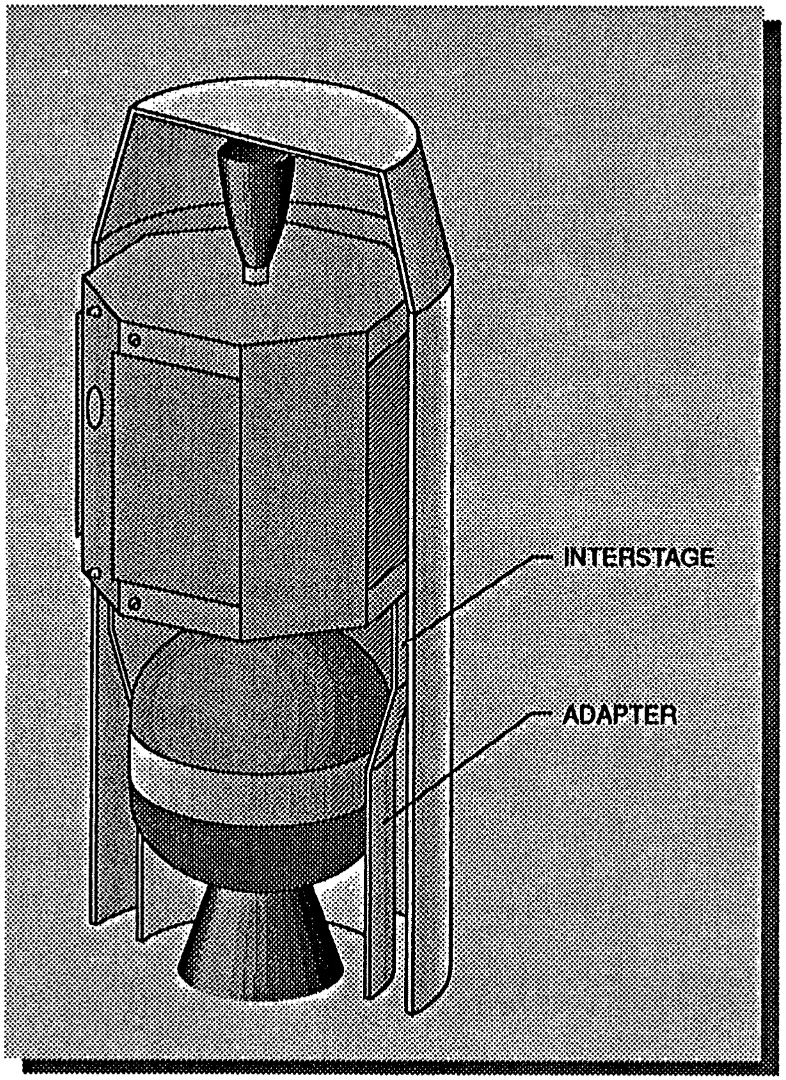
Integration of Active/Passive Control Technologies to Create a Vibration Isolated Optical Bench

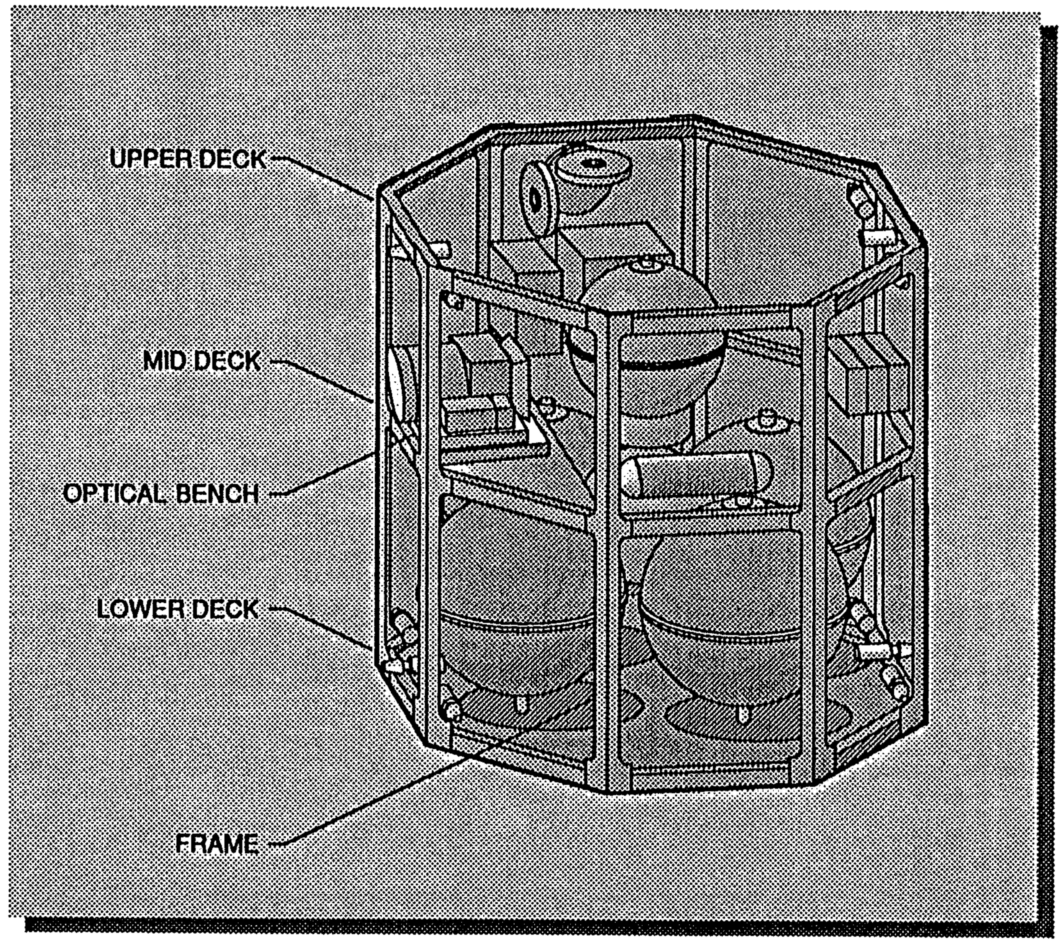
DESCRIPTION

- Advanced Composite Platform with Passive Damping Treatment
- Vibration Isolation of Platform Using Active Control Components
  - Vibration from Spacecraft Bus
  - Disturbances on Platform (Slewing Sensors, Cryocoolers, etc...)
- Active/Passive Vibration Suppression at Optical Sensors
- Correlation of Vibration Suppression to Sensor Performance

A1892.01

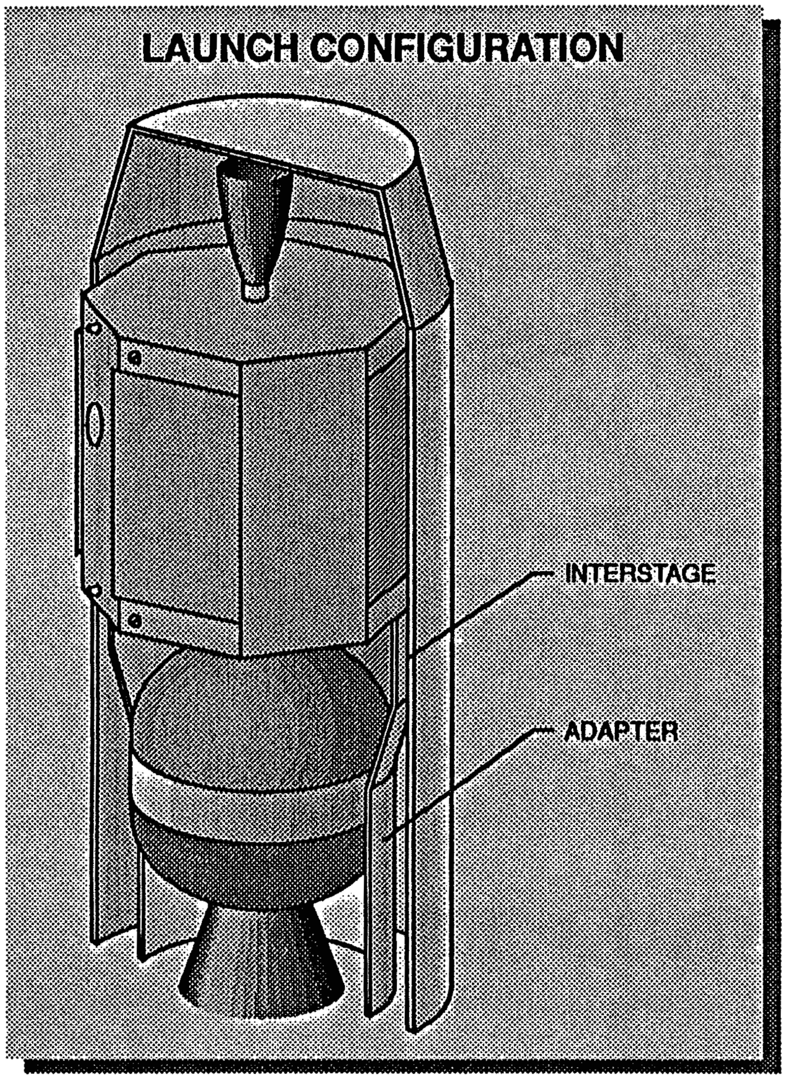
Clementine Candidate Structural Components

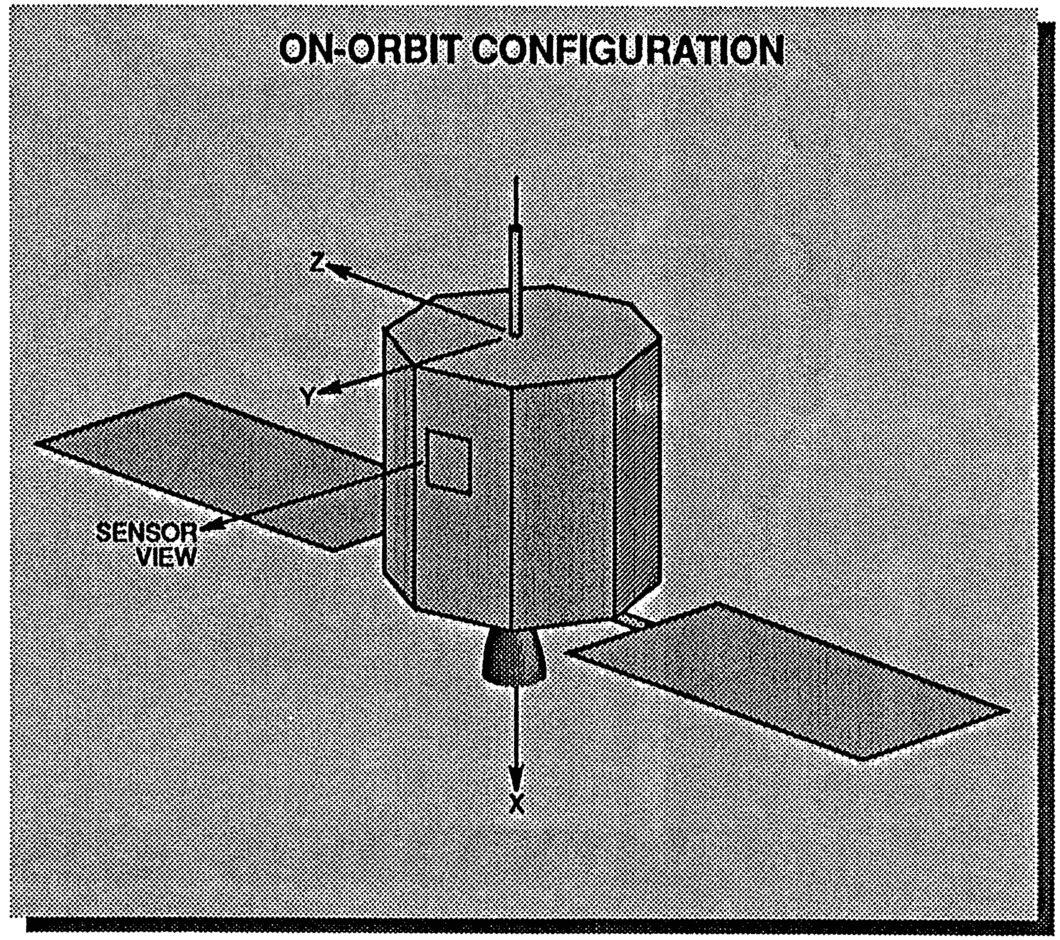




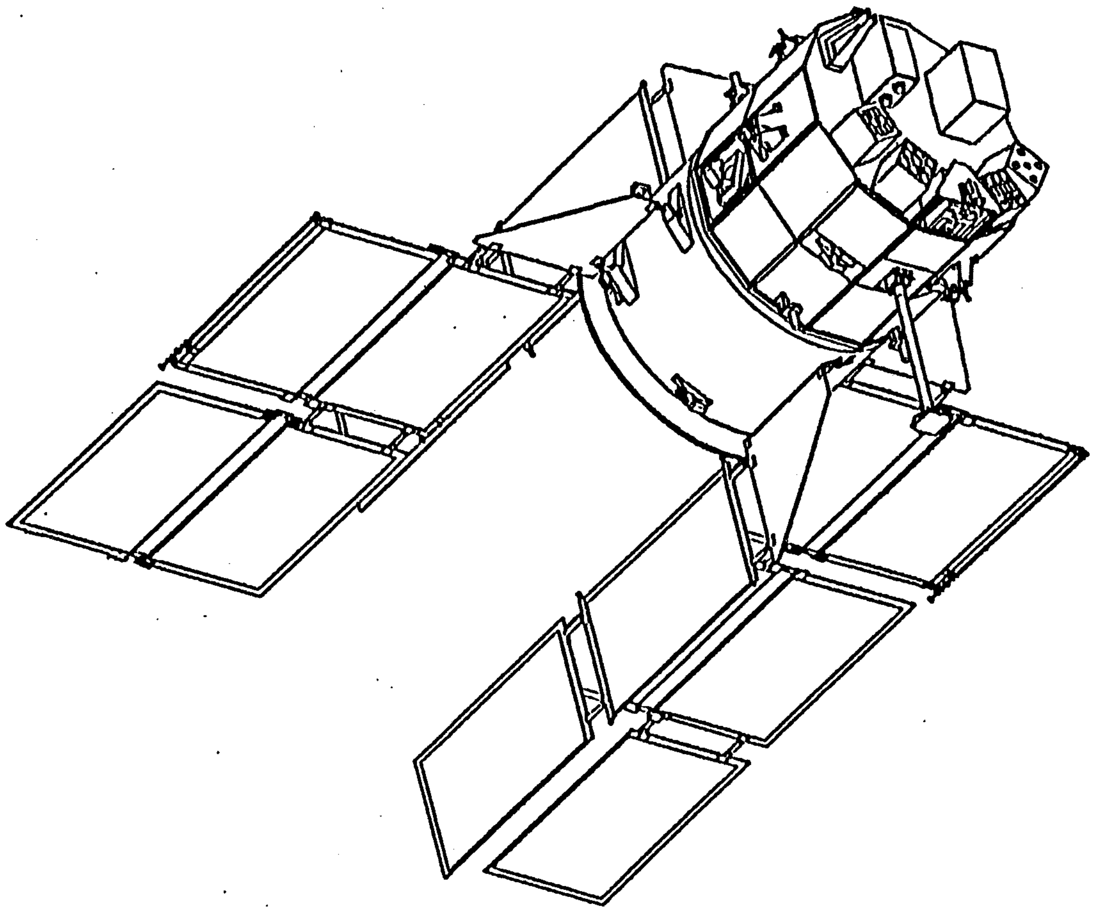
A1892.02

Clementine Spacecraft Configuration



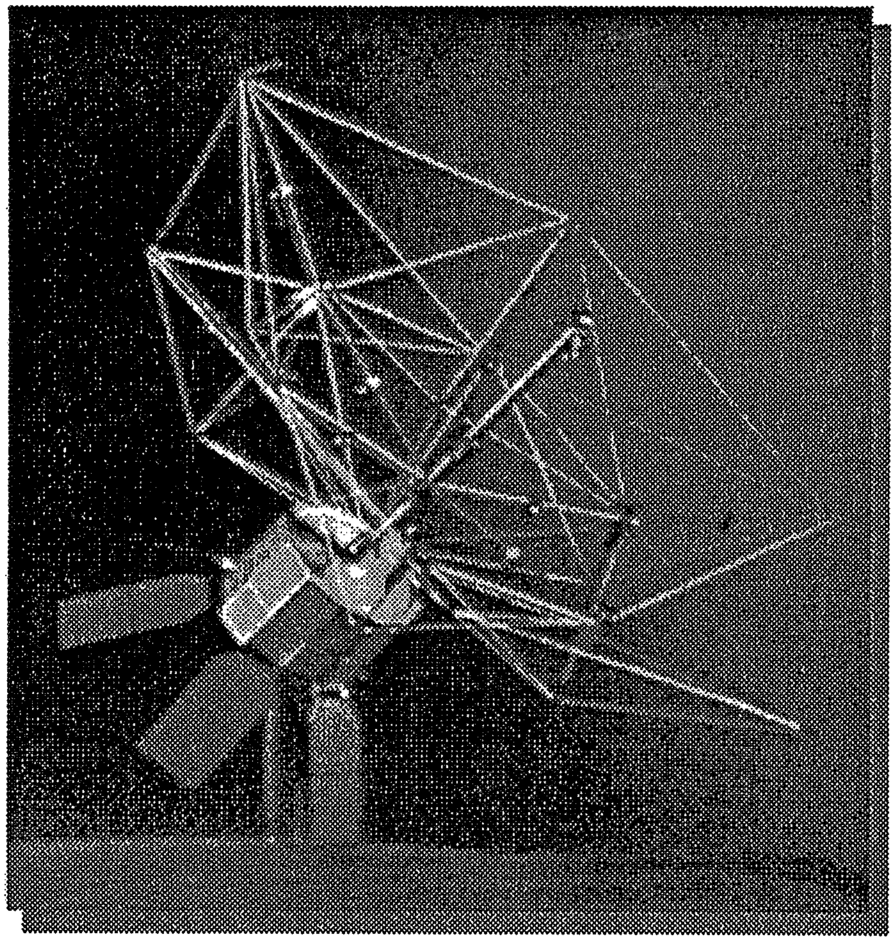


TECHSAT
ALL-COMPOSITE SPACECRAFT



B1182.02

Inexpensive Structures and Materials Flight Experiment (INFLEX)



OBJECTIVE

Integrated On-Orbit Demonstration of Advanced Structures, Materials, and Controls Technology for Precision Space Structures

DESCRIPTION

- 16-Foot Advanced Composite Deployable Antenna, Sized for Pegasus Launch
- Optical Sensing System for Antenna Shape Control
- Piezo Strut for Coupled 2-Body Dynamics
- High-Capacity Processor for Advanced Control Algorithms
- Structural Change Capability for Controller Reconfiguration

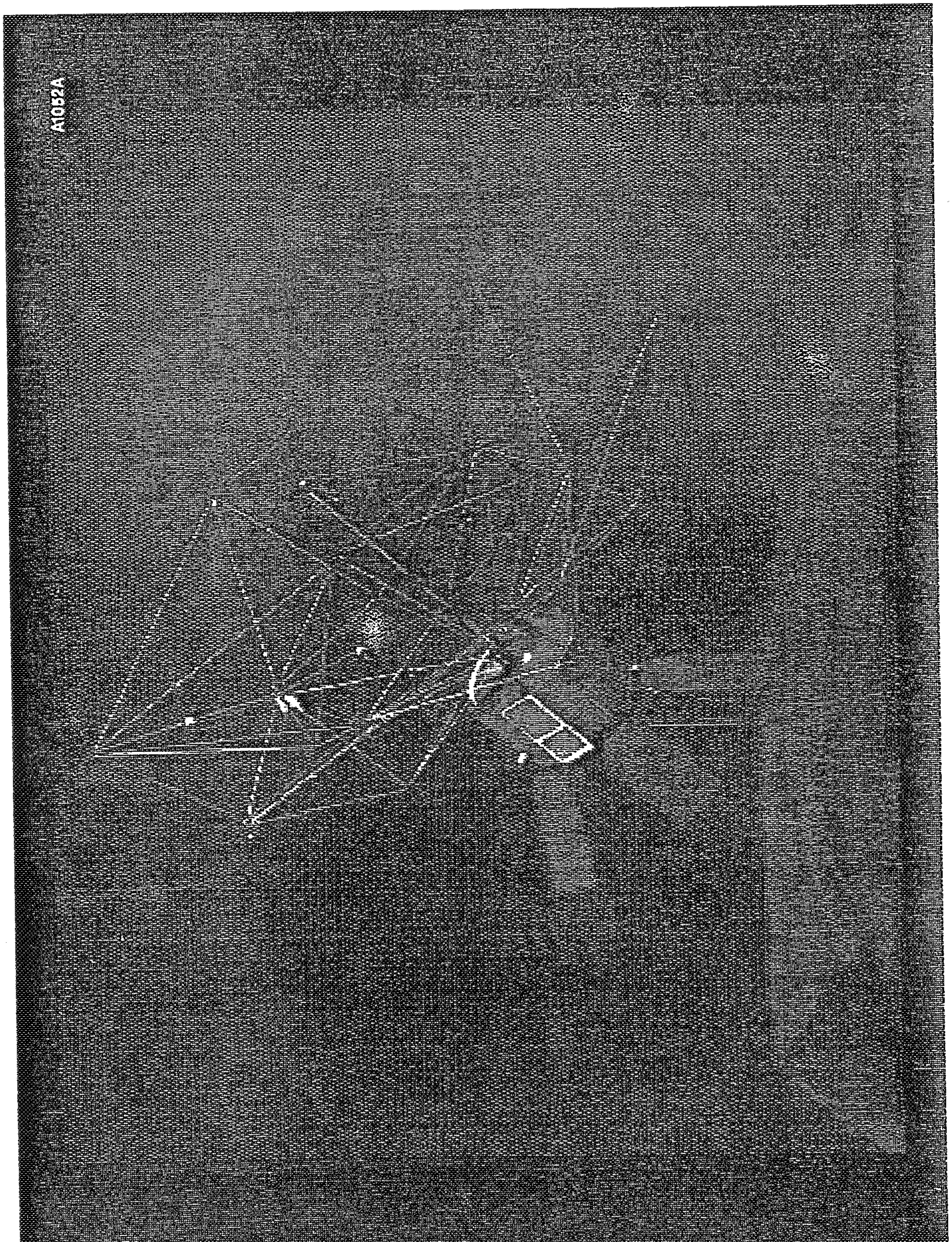
INFLEX PROGRAM STATUS

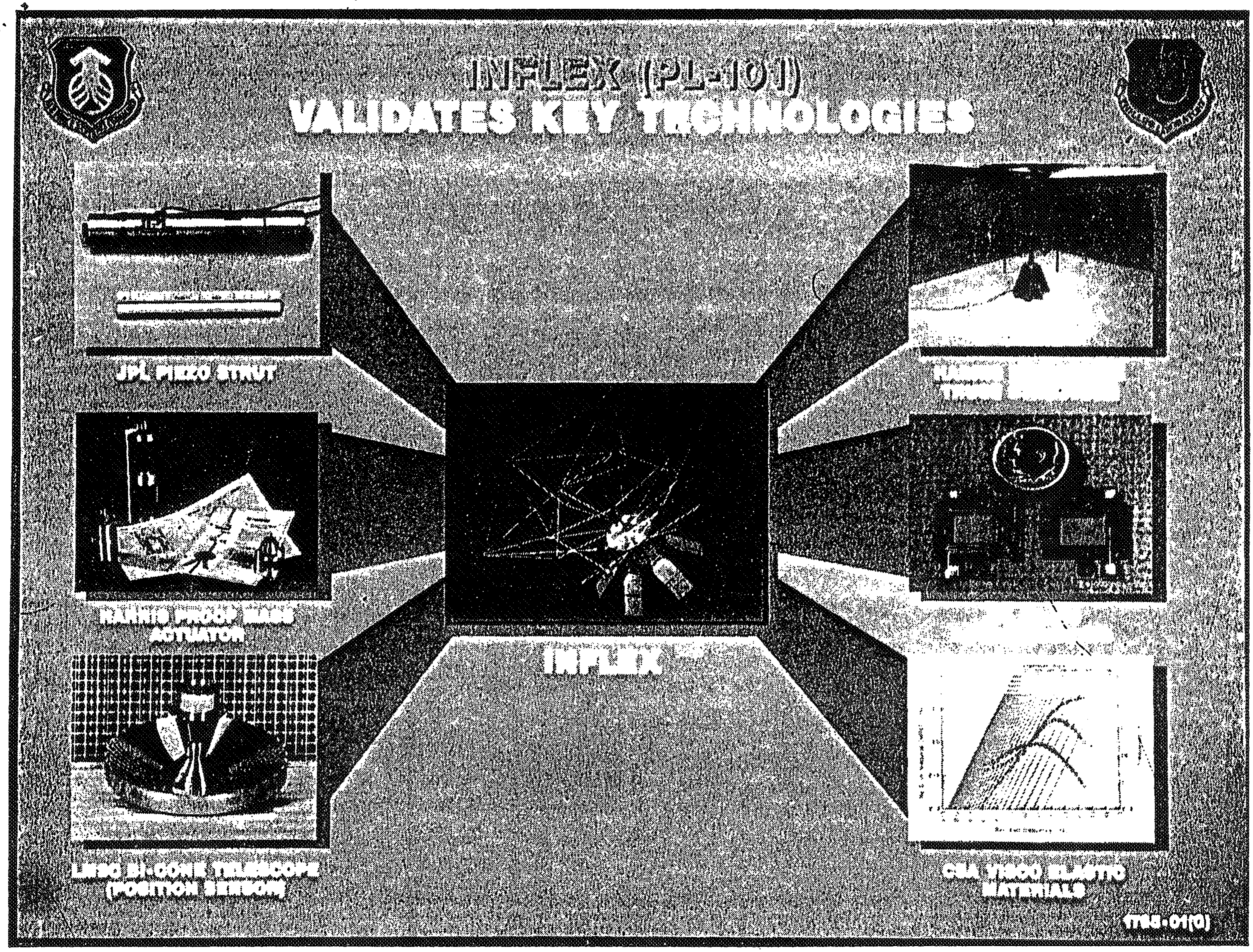
- $1.1M AIR FORCE FUNDING THROUGH PHASE II

- HARRIS CORP HAS COMPLETED ENGINEERING DRAWINGS FOR PRELIMINARY FLT EXP DESIGN

- EXPERIMENT FABRICATION TO COST $12M AND REQUIRE 2 YEARS FROM START DATE

- EXPERIMENT FABRICATION IN PHASE III IS CURRENTLY UNFUNDED


A1052A


INFLEX (PL-101)
VALIDATES KEY TECHNOLOGIES
JPL PIEZO STRUT
HARRIS PROOF MASS ACTUATOR
INFLEX
LMSC BI-CONE TELESCOPE (POSITION SENSOR)
CSA VISCO ELASTIC MATERIALS

CONCLUSIONS

- ON-ORBIT DEMONSTRATIONS ESSENTIAL TO TRANSITION ADVANCED TECHNOLOGY TO OPERATIONAL SPACE SYSTEMS
- SUCCESSFUL FLIGHT EXPERIMENTS ADDRESS SPECIFIC OPERATIONAL CONCERNS IN SMALL, NEAR-TERM TECHNOLOGY DEMONSTRATIONS
- JOINT EFFORTS GREATLY FACILITATE EFFORTS, E.G., SDIO FUNDING, AF TECHNOLOGY, NRL FLIGHT
- SDIO HAS PROVIDED STRONG SUPPORT FOR FLIGHT EXPERIMENTS IN AREA OF ADAPTIVE STRUCTURES

MIT
Space
Engineering
Research
Center

THE MODE FAMILY OF FACILITY CLASS EXPERIMENTS

Dr. David W. Miller MIT

NASA/DOD Flight Experiments Technical Interchange Meeting
October, 1992

N93-28705
56-18
159209

THE MODE FAMILY OF EXPERIMENTS

Fluid Test Article (FTA)	Structural Test Article (STA)	MACE Test Article
Coupled Non-Linear Dynamics of Fluids and Structures in Zero Gravity	Non-Linear Dynamics of Jointed Truss Structures in Zero Gravity	Influence of Gravity on the Active Control of a Multibody Platform

Flight # 1:
September 1991

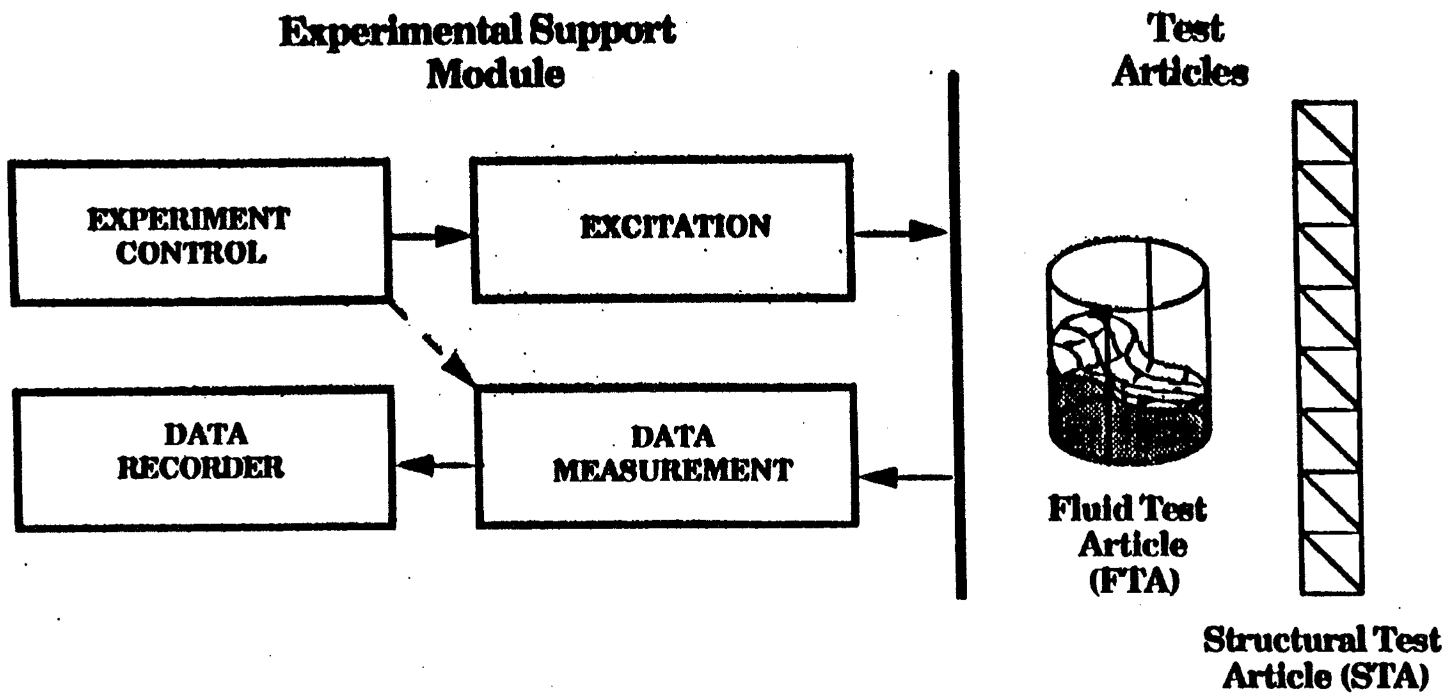
Flight #2:
June 1994

MACE is part of a logical sequence of cost-effective flight experiments designed to advance technology of interest to NASA in the area of controlled structures.

EXPERIMENTAL PHILOSOPHY

- Use the shuttle/station for engineering research (as opposed to demonstration or verification)
- Investigate (dynamics) phenomena which are influenced by gravity
- Use the Middeck/Laboratory Module as a shirt sleeve lab environment with heavy reliance on crew interaction
- Use scaling laws to build model which capture the essential physics of the problem and yield results of practical value, at modest size and cost

THE MIDDECK 0-GRAVITY DYNAMICS EXPERIMENT (MODE)



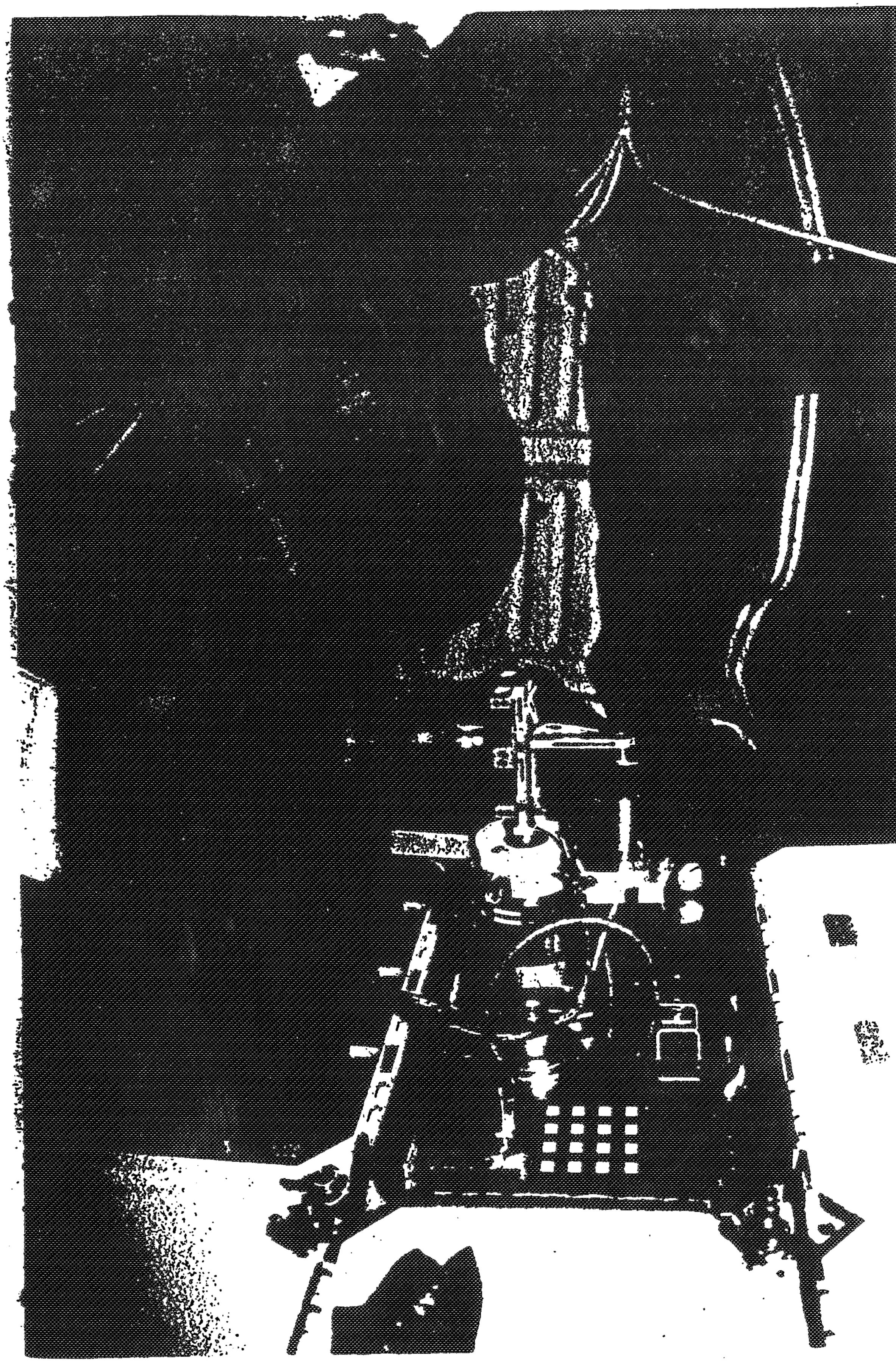
MODE provides a reusable dynamics test facility which will be used on the first flight to test two rather different types of test articles.

OBJECTIVES

- Engineering science objective is to characterize fundamental 0-g slosh behavior and obtain quantitative data on slosh force and spacecraft response for correlation of analytical model.

- Why:
  - Higher fluid mass fractions of on-board fluids.
  - Uncertainty in fuel behavior requires larger ΔV margins.
  - Nonlinear dynamics significant alters spacecraft dynamics and fluid cannot considered to be isolated from the spacecraft dynamics.

- To:
  - Design more efficient spacecraft.



NASA National Aeronautics and Space Administration

S48-03-032

Lyndon B. Johnson Space Center Houston, Texas 77058

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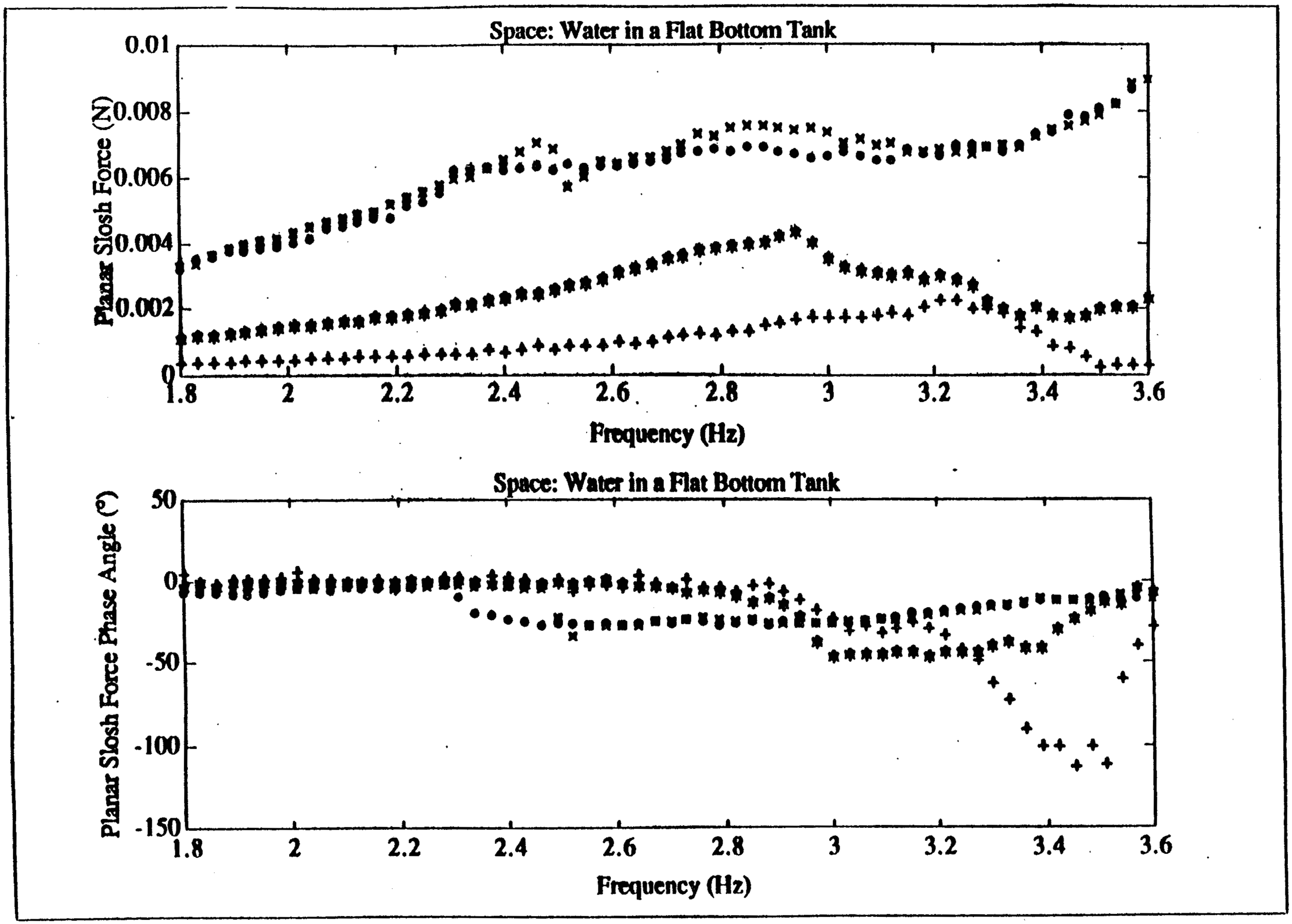
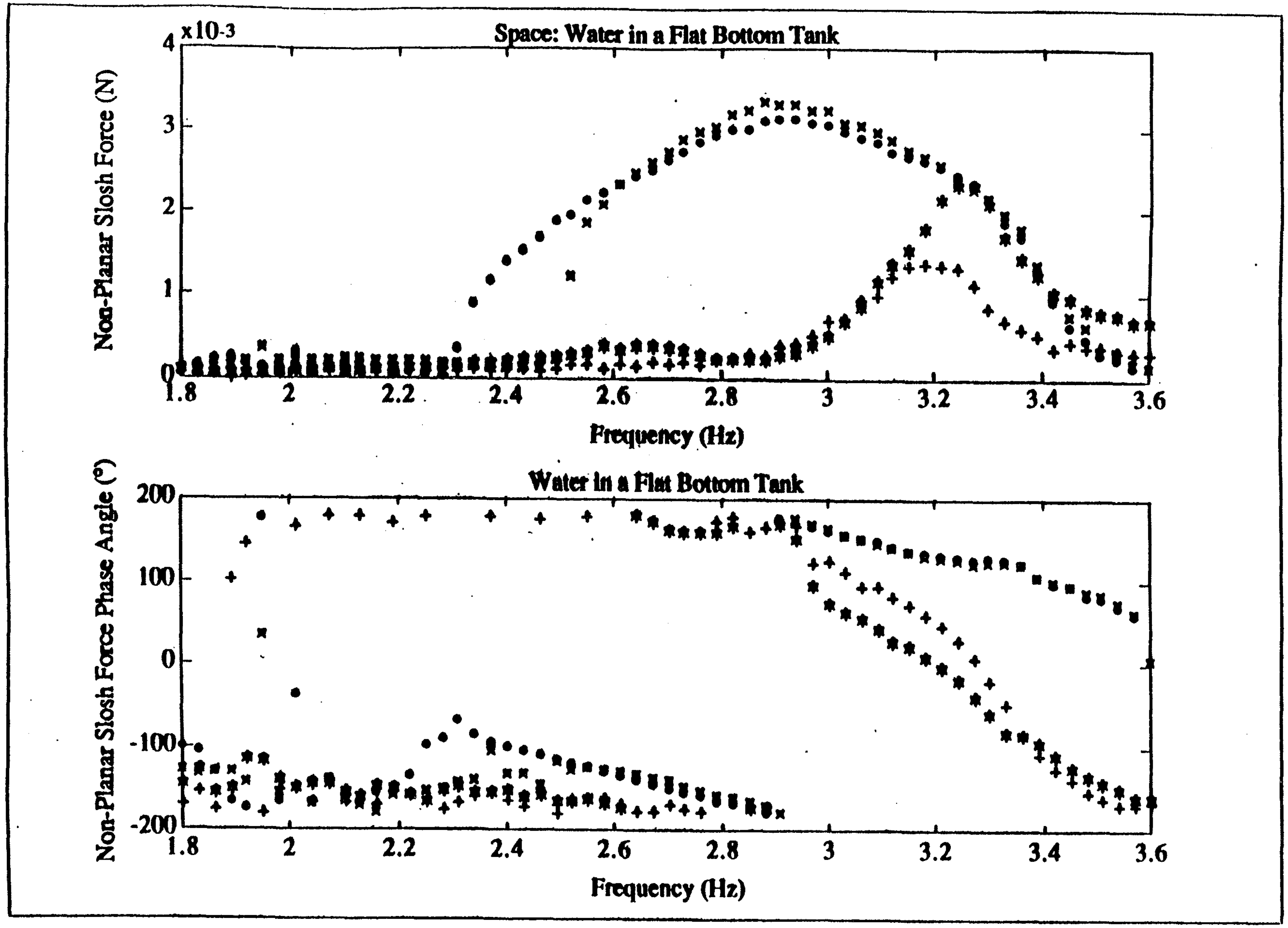
SPACE RESULTS



Uncoupled Test with Distilled Water as Test Fluid in a 3.1 cm Flat Bottom Cylindrical Tank. Planar Slosh Force.



Uncoupled Test with Distilled Water as Test Fluid in a 3.1 cm Flat Bottom Cylindrical Tank. Non-planar Slosh Force.

SUMMARY AND CONCLUSIONS

Space Experiments:

- More benign nonlinear behavior in space than observed on earth
- Modal damping ratios and frequencies significantly different from earth tests
- Demonstrated the ability to investigate fluid slosh in micro-gravity

Analytical Model

- Model more accurate for one-gravity conditions
- Nonlinear solution required that can find "all" the solutions
- Accurate prediction of slosh damping ratios a pre-requisite for an accurate prediction

Future

- Improve nonlinear solution technique
- More space experiments required to investigate effects of contact angle hysteresis, contact angles and dissipation rates.

STA OBJECTIVES, REQUIREMENTS & APPROACH

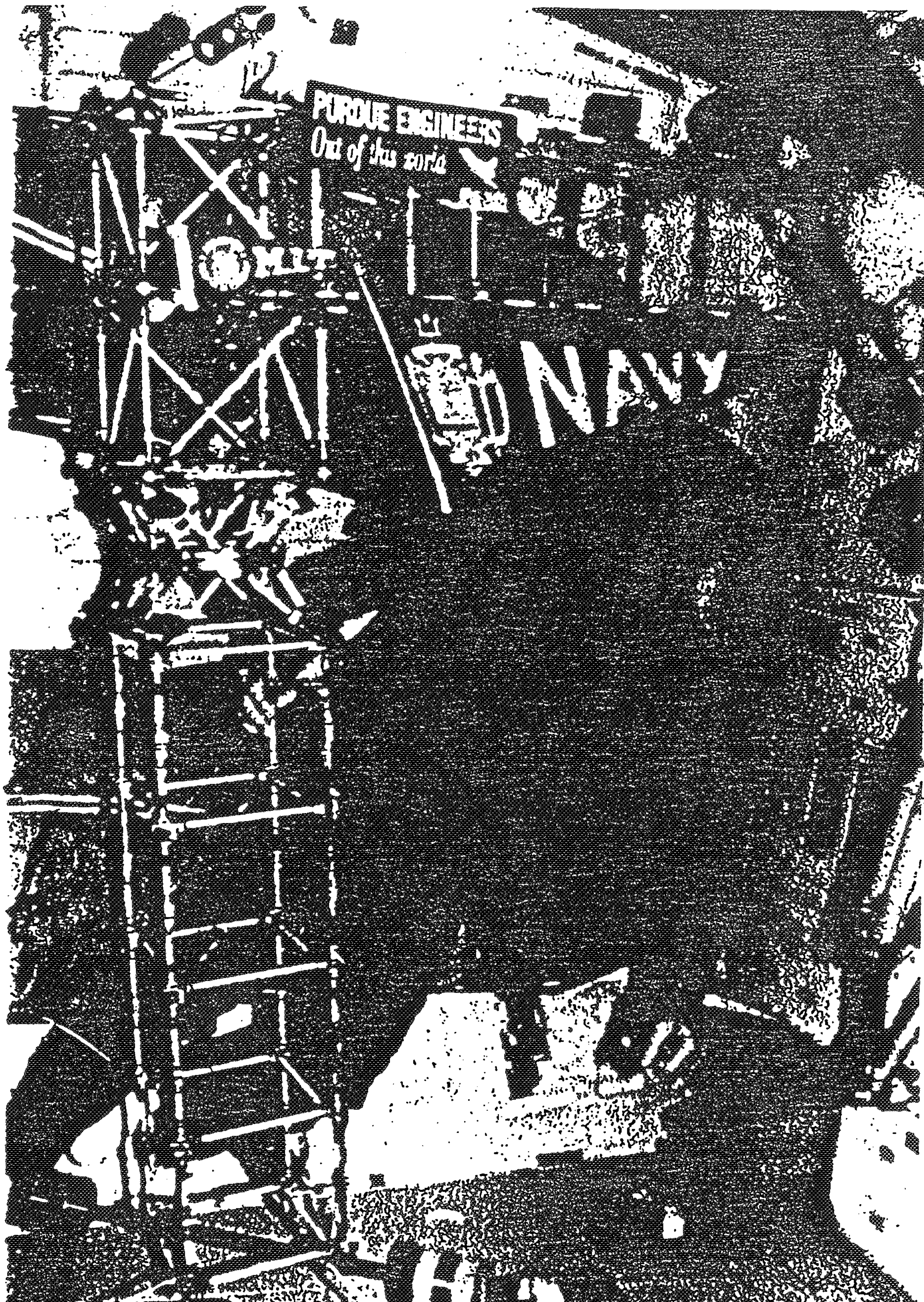
- Engineering science objectives are to characterize the fundamental changes in dynamics in 0-g due to absence of gravity on joints, to quantify the changes due to the absence of suspension and gravity load on members, and to obtain quantitative data for correlation with numerical models.

- Requirements

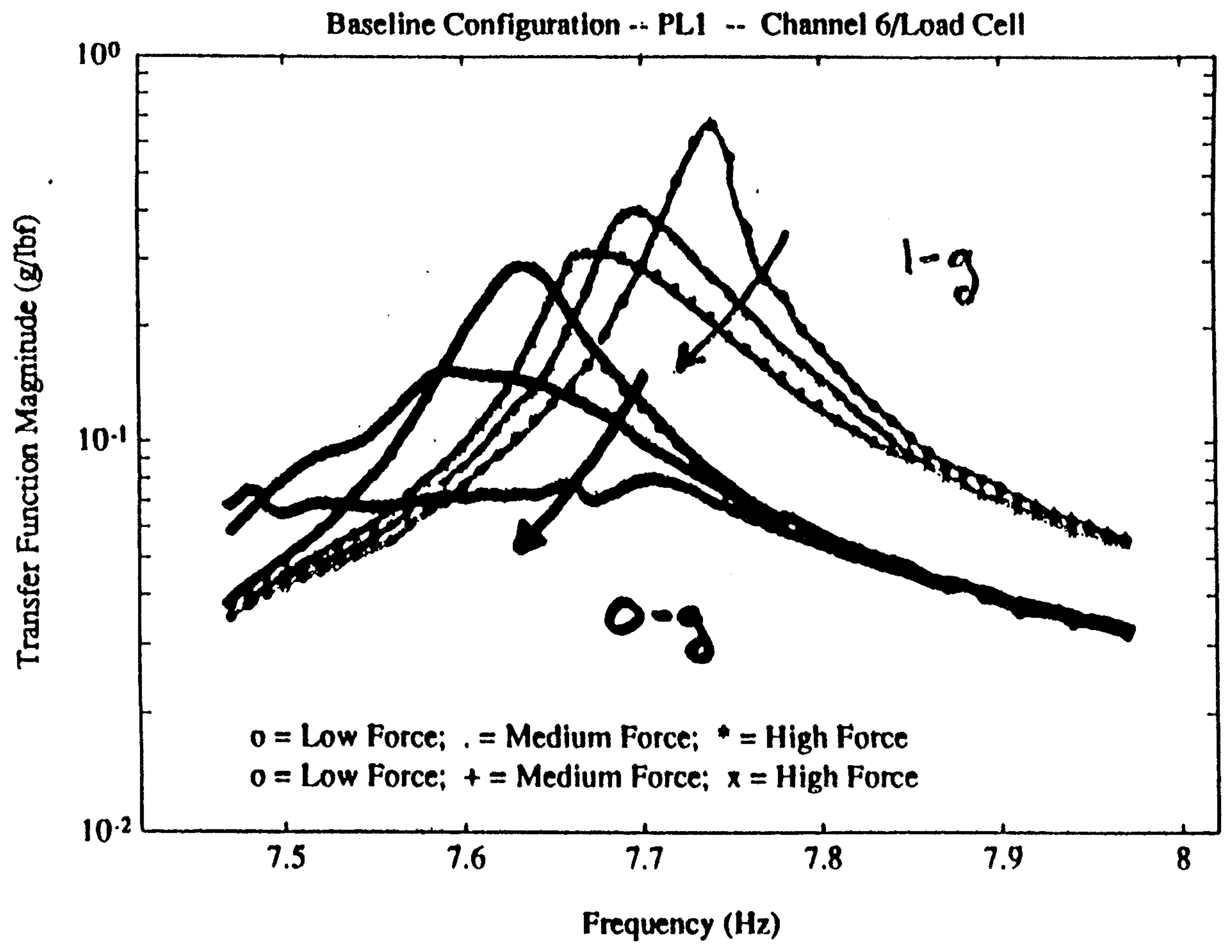
  Truss structure containing elements of future space structures.
  Nonlinear joints with variable pre-load to test nonlinear behavior in several gravity/joint pre-load conditions.
  Reconfigurable truss with deployable and erectable bays.

- Modelling approach

  Develop global linear model using FEM and modal test data.
  Develop Force-State Map of non-linear sub-components.
  Develop describing functions from Force-State Map.
  Insert describing functions into global model and solve for forced response using Harmonic Balance Method.
  Verify predictions with MODE flight and ground test results.

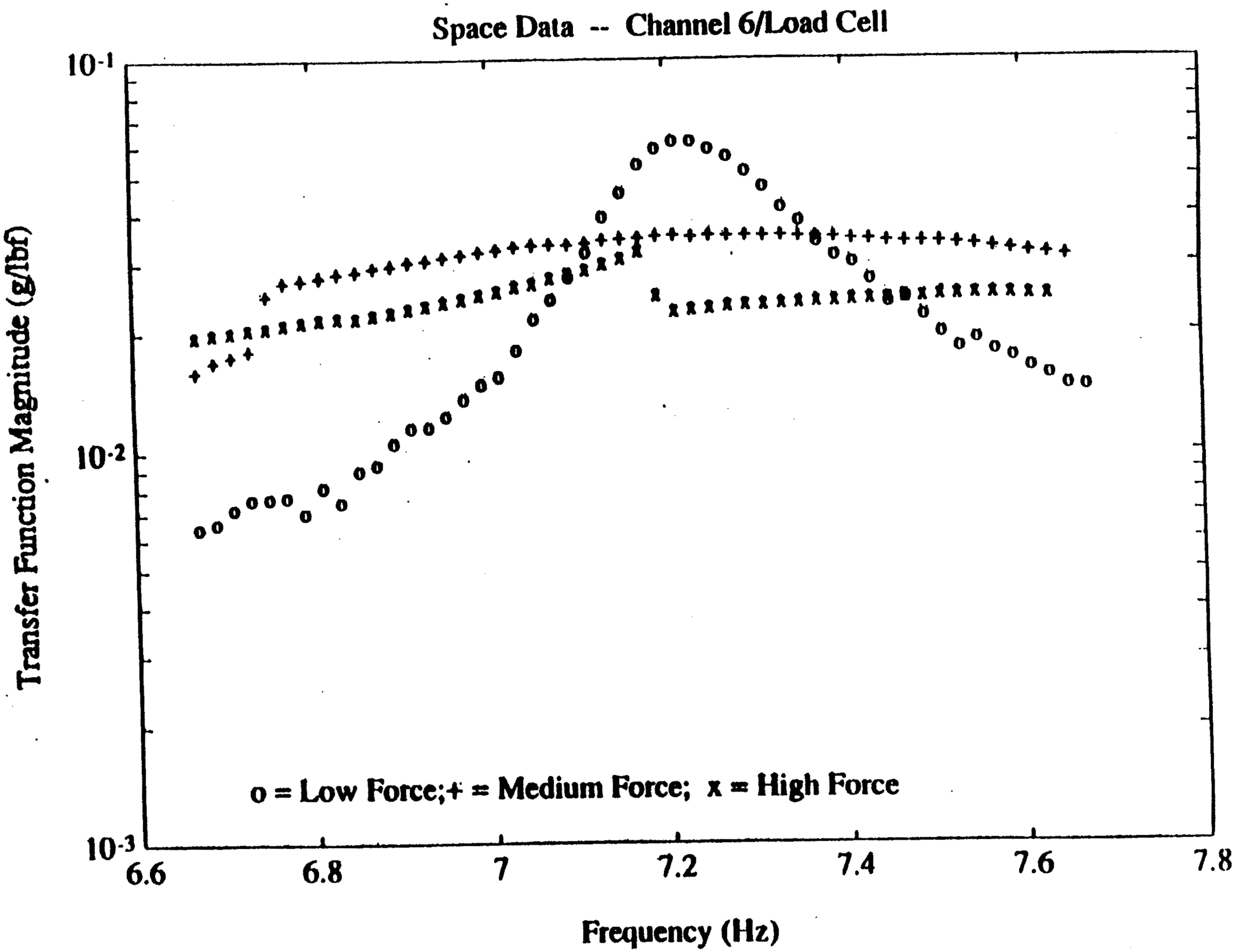

PURDUE ENGINEERS
Out of this world
MIT
NAVY

COMPARISON OF GROUND TO ORBITAL DATA FOR THE BASELINE CONFIGURATION



NOTE: Torsion Mode Only. High Pre-Load.

TORSION MODE ALPHA LOOSE

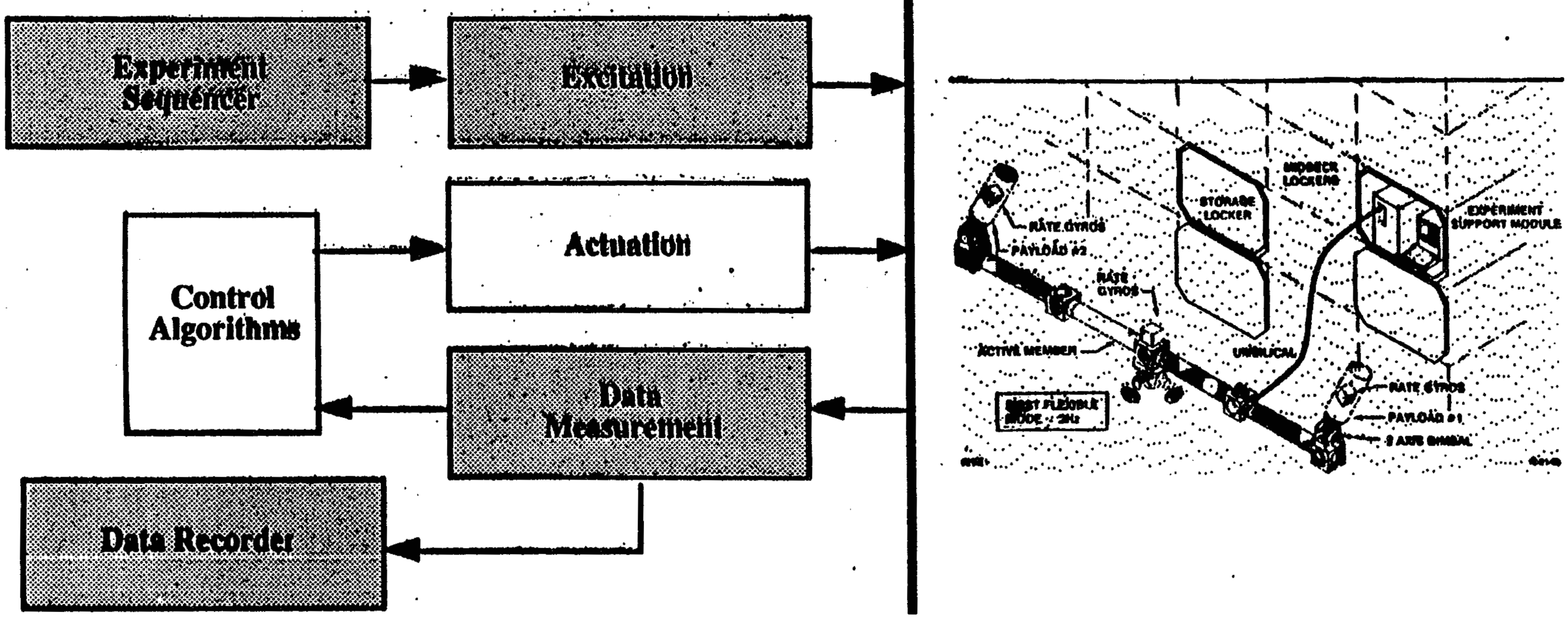


CONCLUSIONS OF ORBITAL TESTING

- Variation measured for erectable, deployable and articulated hardware as a function of force amplitude, joint preload and gravity loading
- Nonlinearities of the STA are more apparent in 0-gravity, especially the alpha loose, which loses resonant behavior
- Modes generally soften with increasing force, but increase in damping is significantly more pronounced
- Changes in frequency between earth and space are generally within the variance of ground testing for the baseline, but outside the variance for the alpha and L configurations.
- Changes in damping are well outside the variance of ground testing

FLIGHT EXPERIMENT RESOURCES

ESM
Experiment Support Module

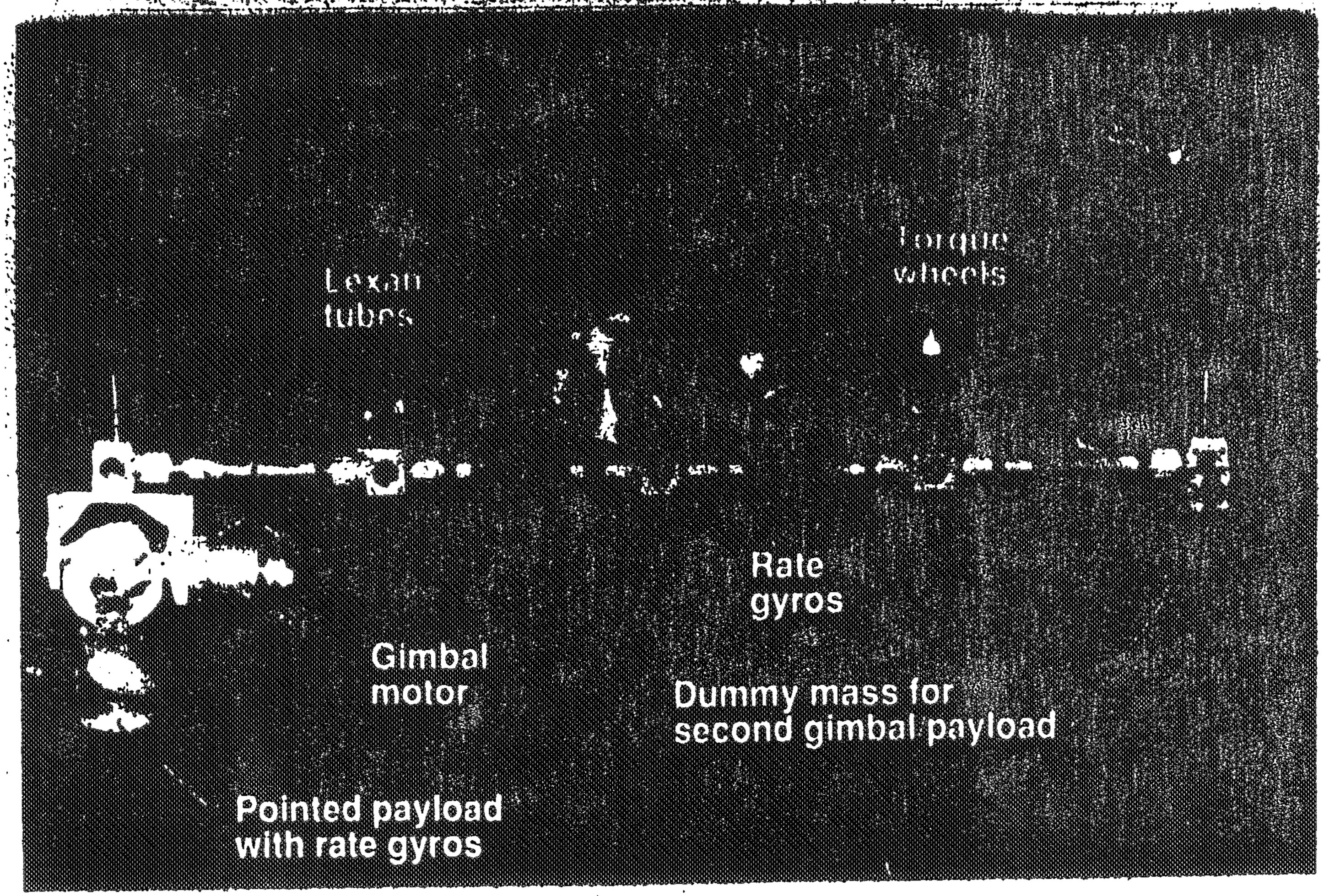


- Three eight hour days of one crew member.
- Test article and support equipment stored in 3 middeck lockers.
- ESM stored in a fourth middeck locker.
- Wiring is pre-integrated in the test article for ease of assembly.

MIDDECK ACTIVE CONTROL EXPERIMENT (MACE)

Development Model Lab Testing

(Flight unit will have smaller torque wheels and gimbal motors)



PROGRAM OBJECTIVES

- Science Objective

  To develop a verified set of methods that will allow designers of CSI/CST spacecraft, which cannot be dynamically tested on the ground in a sufficiently realistic 0-g simulation, to have confidence in the eventual orbital performance of such spacecraft.

- Implications

  Understand direct and indirect gravity effects and the relation between control authority and manifestation.

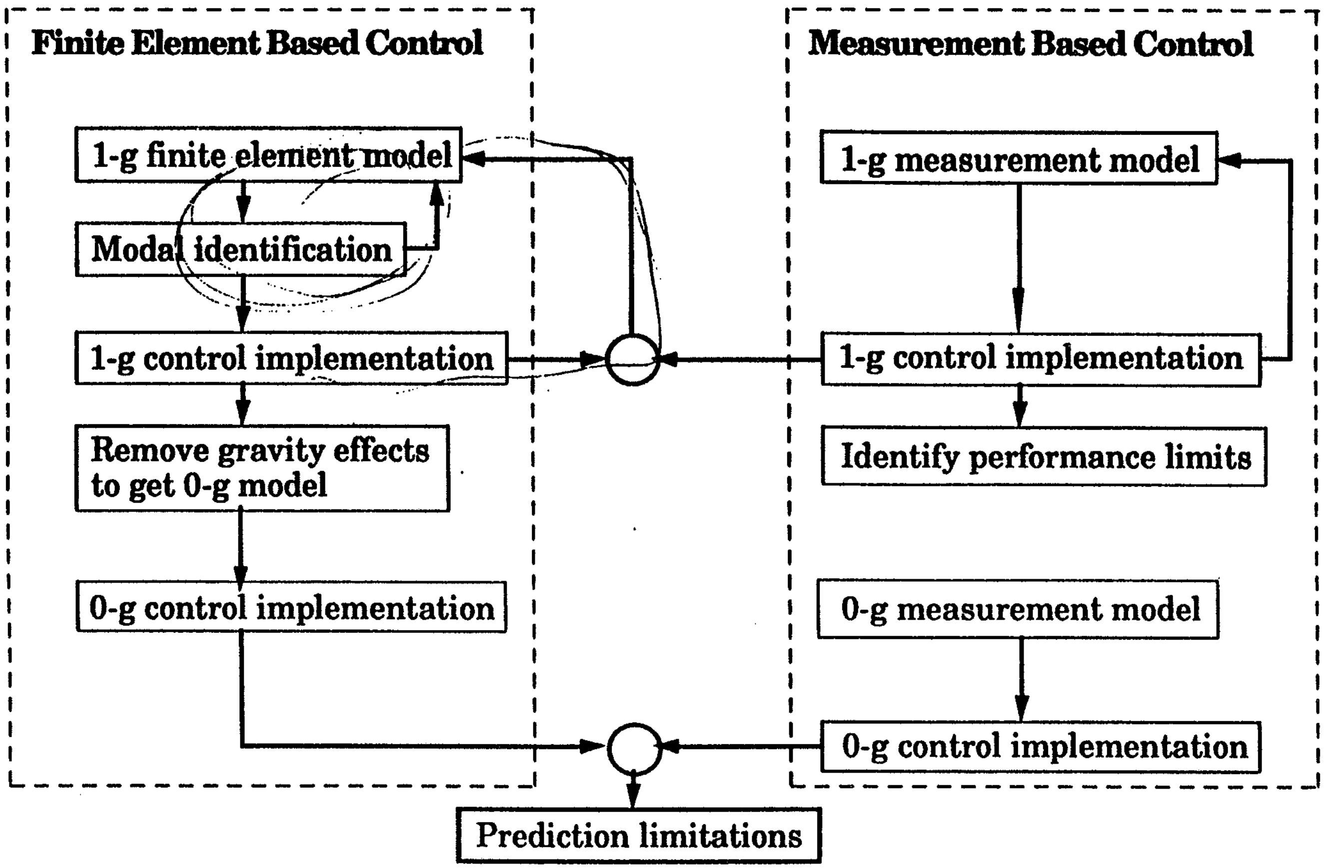
  Develop procedures for predicting on-orbit performance.

  Quantify prediction accuracy achievable through analysis and ground tests.

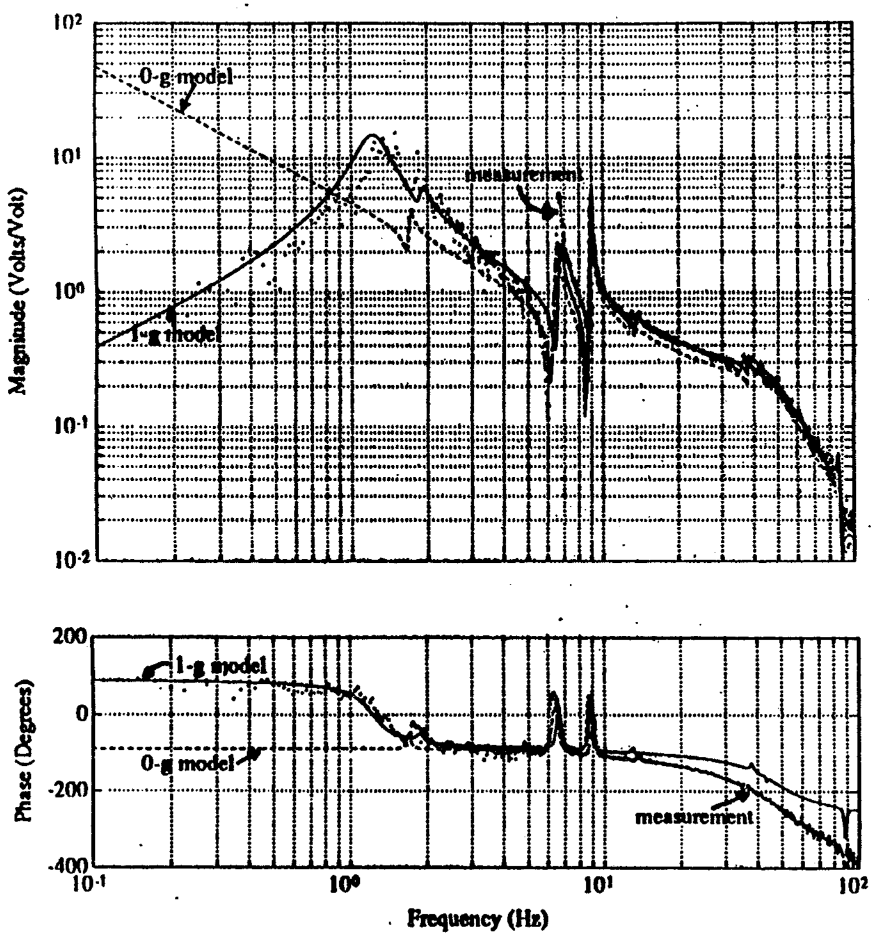
  Develop techniques for on-orbit identification.

  Quantify performance improvement through control redesign based upon on-orbit identification.

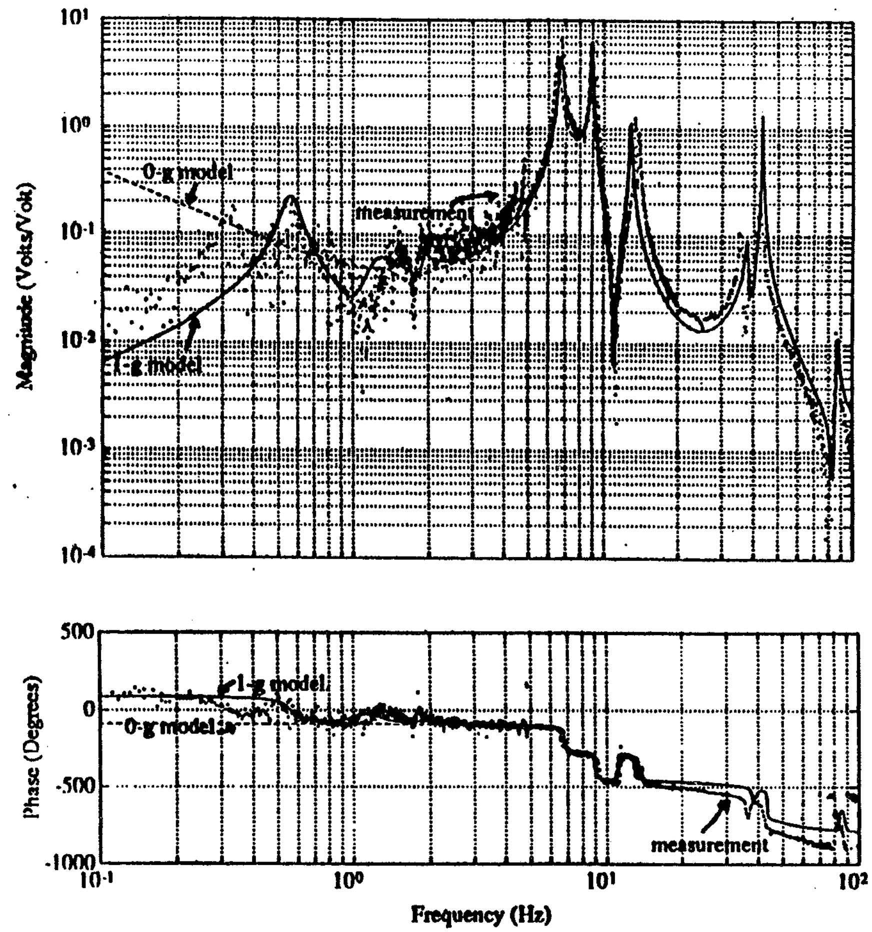
SCIENCE APPROACH



MACE 1-G AND 0-G MODELS

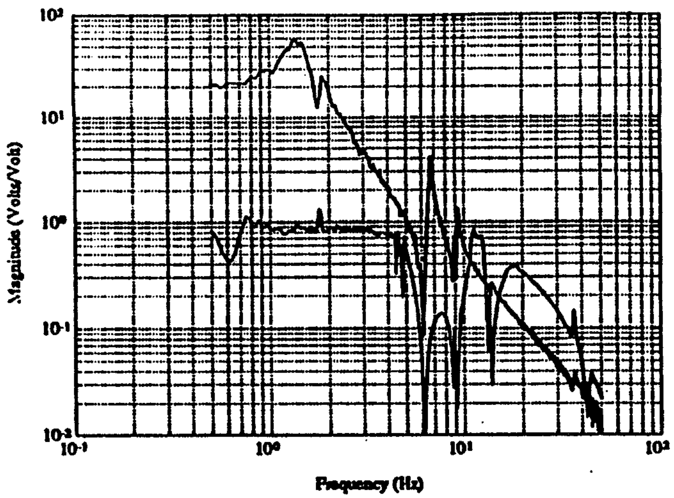


Transfer function from z-axis gimbal torque to z-axis payload inertial angular rate.

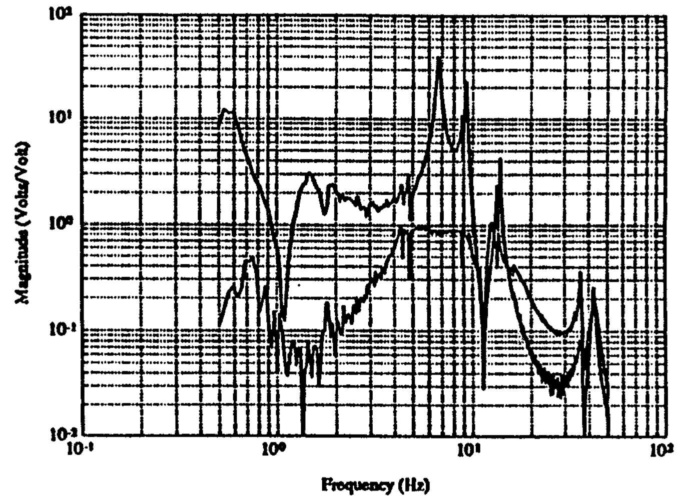


Transfer function from z-axis gimbal torque to z-axis bus inertial angular rate.

SINGLE-INPUT, TWO-OUTPUT WITH PAYLOAD AND BUS PENALTY

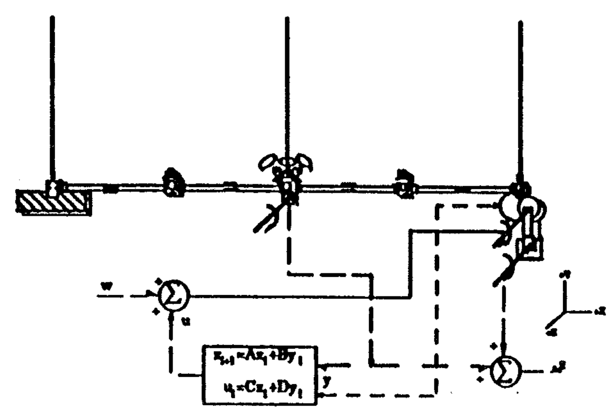


Transfer functions from gimbal to payload inertial angle



Transfer functions from gimbal to bus inertial angle

- Payload pendulum and first bending modes are suppressed
- In addition, higher frequency flexible modes are suppressed.
- An order of magnitude reduction in pointing error is achieved.



Control Topology #3

FUTURE EFFORTS

M.I.T. SERC is pursuing followon flight experiments which either reuse the MODE and MACE hardware or build upon the associated experience.

- MODE Reflight in 1993
  - Reflight of the Structural Test Article (STA) to investigate two unexpected physical phenomena which appeared during the on-orbit operation of MODE I.
  - Shifts in the modal frequencies in the most geometrically complex configuration were greater than expected and laid outside the preprogrammed test windows.
  - The alpha joint exhibited jump phenomena.
- Fluid Advanced Dynamic Experiment (FADE)

  The MODE ESM with Fluid Test Article (FTA) Assembly is available to conduct on-orbit fluid dynamic research.
- MODE on MIR

  M.I.T. SERC has received a written invitation to fly MODE hardware on the Russian Space Station (MIR) for acquisition of extended micro-gravity test data.

SUMMARY

- The MODE family of flight experiments is designed to verify analytical tools developed to predict the gravity dependent behavior of proposed space structures.
- The MODE family of flight experiments uses reusable dynamic and control test facilities and exploits the shirt sleeve environment on the STS middeck.
- MACE investigates gravity dependent phenomena pertinent to the closed-loop dynamics of proposed space structures.
- Gravity and suspension effects perturb the MACE test article flexible modes when tested on the ground.
- Suspension mode stabilization can obscure important gravity influences on the flexible behavior during 1-g closed-loop testing.
- Measurement models have been used successfully to achieve over an order of magnitude improvement in pointing accuracy.

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N93-28706
p. 21

Control-Structures Interaction Test of the LACE Satellite

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Shalom (Mike) Fisher
Naval Research Laboratory
Washington, DC 20375
202-767-3914

ABSTRACT

It is clear that additional experience and validation of Control-Structures Interaction (CSI) techniques are needed in controlling the structural dynamics of flexible spacecraft. It is also clear that the effects of the space environment such as weightlessness dictate that this be done in space. Unfortunately, orbital tests are difficult to achieve because of the high cost of the test article, the launch into orbit, the instrumentation and communication systems.

The LACE Satellite has provided an opportunity to achieve a CSI test in space for very little cost. First, the CSI test rode piggy-back and did not interfere with the primary objective of LACE. Second, the novel technique of using ground based measurements of vibration of the orbiting satellite was employed. The LACE has a heavy central body to which is attached booms with lengths as long as 150 feet. The ground measurements were obtained using a laser, Doppler radar at the MIT Lincoln Laboratory Firepond Facility.

The initial tests demonstrated the accuracy of the vibration measurements and obtained structural responses for enhancing the accuracy of the mathematical model of the structural dynamics. Germanium corner-cube retroreflectors attached to the central body and a boom deployed to 18 feet ensured a high strength return signal. Subsequent tests demonstrated the ability of an open-loop damper to attenuate the vibrations of the orbiting satellite. The LACE test results are important in (1) contributing to the validation of a CSI technique, and (2) demonstrating a novel ground measurement technique for orbital tests that is accurate but which has very low cost.

Flight Experiments Technical Interchange Meeting

Control - Structures Interaction Test of The LACE Satellite

Dr. Shalom ("Mike") Fisher
Naval Research Laboratory

Lawrence W. Taylor
NASA Langley Research Center
Hampton, VA

October 5 - 9, 1992

Monterey, California

Outline of Presentation

- LACE satellite description.
- LACE dynamics experiment description.
- Firepond Laser Radar facility.
- Illumination and vibration measurement history.
- Dynamic excitations and on-orbit responses.
- Concluding remarks.

Objectives of the Dynamics Experiment

- Unique opportunity to measure effects of disturbances on spacecraft flexure; demonstrate ground-based sensing.

- Perform on-orbit system identification: vibration frequencies, damping and amplitude ratios.

- Demonstrate "open-loop" active damping.

  gravity-gradient boom used as actuator
  preprogram timed retractions/deployments
  induce and damp oscillations
  finite element models (FEM): NASTRAN
  dynamics simulation models

- Facilitate control of jitter and rapid slews in future spacecraft.

Piggyback/Secondary Experiment

- **Disadvantages:**
  - Play second fiddle to main experiments:
    - Orbit, power, telemetry, attitude control, environmental, thermal, radiation
  - Must meet stringent host interface requirements and launch schedule.
  - Publicity: Low profile
- **Advantages:**
  - Low-cost, rapid results
  - Main advantage: CSI flight experiments are possible

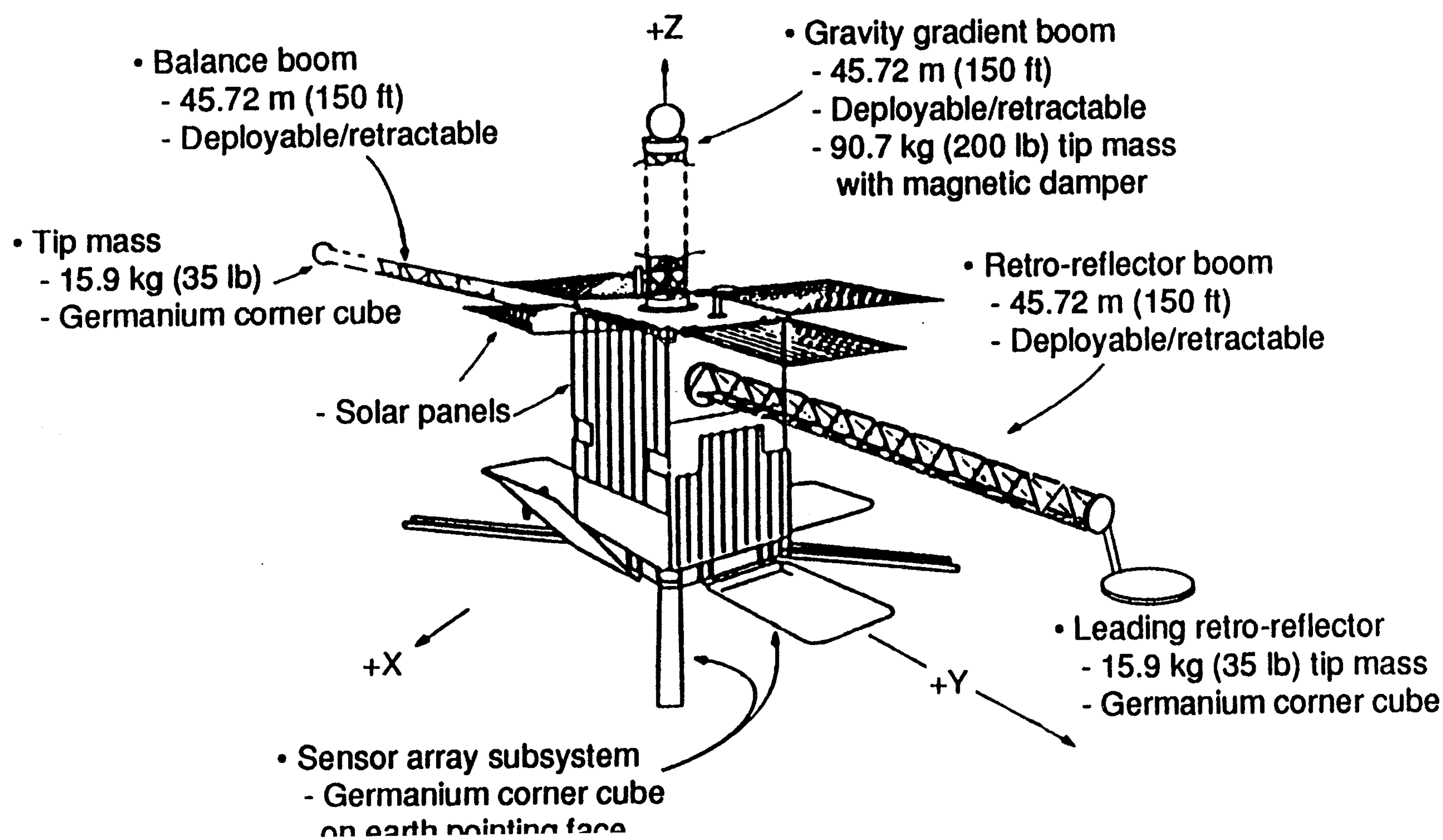
Key Points of the Experiment

- LACE spacecraft launched February 14, 1990

  Altitude at lauch 540 km, circular, 43 inclination

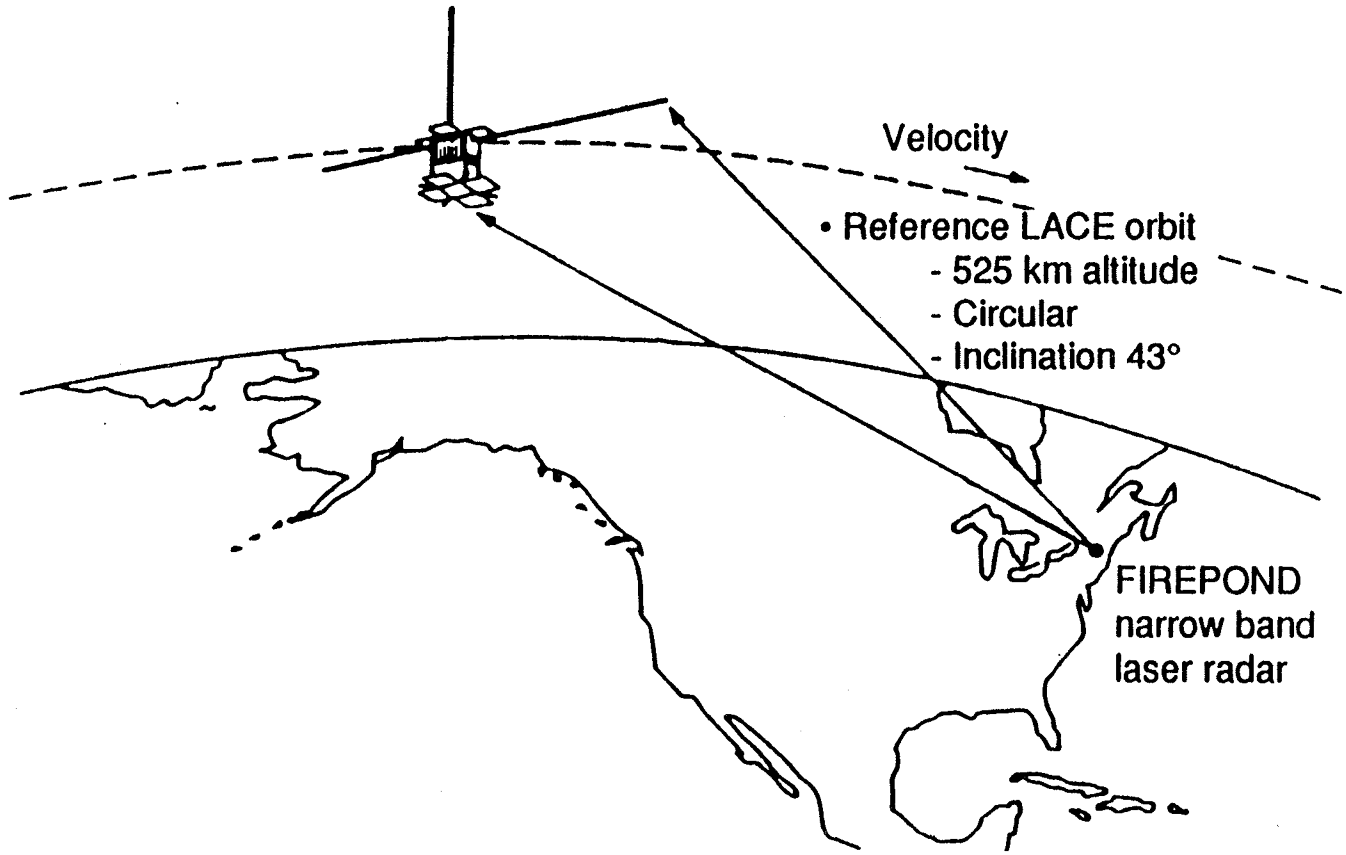
- Lace satellite built and launched by Naval Research Laboratory
- Dynamics experiment is a low - cost "piggyback" experiment.
- Germanium corner cubes (3) serve as targets for Firepond laser radar of MIT Lincoln Laboratory, Wastford, Massachusetts
- Laser Doppler data first collected January 1991 gave system id.
- Dynamic excitations observed and modelled August, September 1992

LACE Spacecraft



Dynamics Experiment

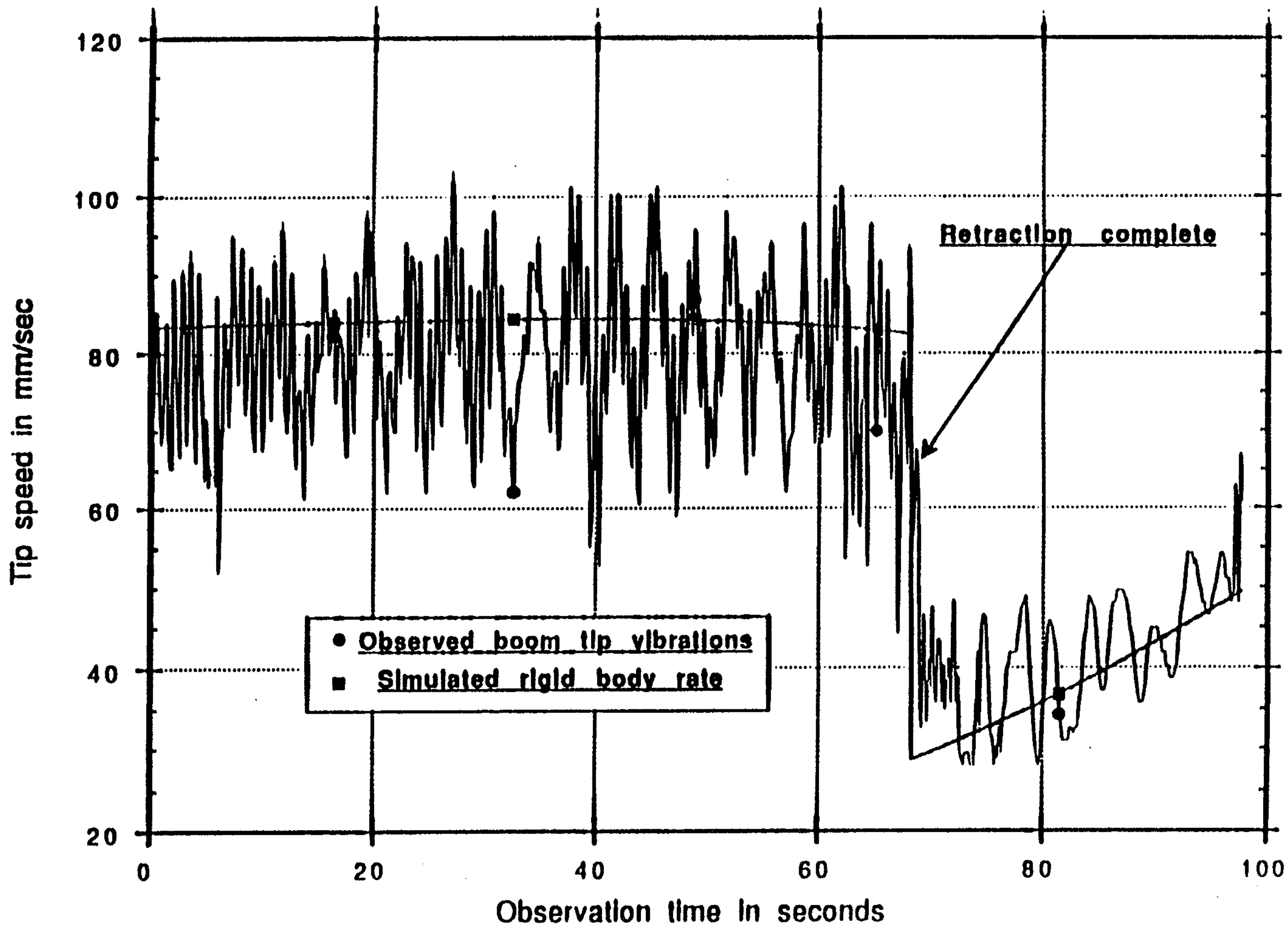
• Estimate satellite vibration modes from doppler resolved laser radar measurements



Boom Dynamics Experiment Observations

Date	Leading Boom (feet)	Trailing Boom (feet)	Tracking	Illumination
7 Jan 91	80 → 15	150	Active	Narrowband
8 Jan 91	80 → 15	150	Active	Narrowband
10 Jan 91	80 → 15	150	Active	Narrowband

VIBRATION OBSERVATIONS COMPARED WITH SIMULATED RIGID BODY RATES, DAY 91008



Comparison of observed with modes computed from FE modelling (stick model)

$EI = 1.55 * 10^4$ N - m^2 $GJ = 5.74 * 10^2$ N - m^2

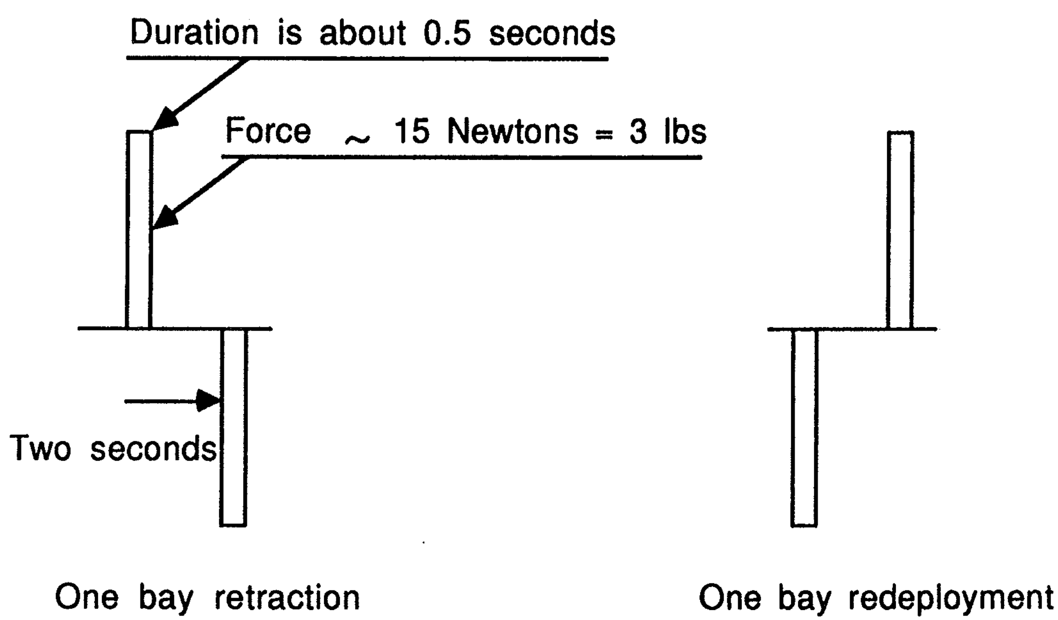
Obs freq	FEM freq	tip modal displacements Δz	Δx
*0.019 Hz	0.019 Hz	.010	--
	0.110 Hz	.001	.05
	0.112 Hz	.002	.09
•0.124 Hz	0.125 Hz	.09	.004
	0.258 Hz	.009	--
	0.297 Hz	--	.08
•0.335 Hz	0.316 Hz	.10	.006
	0.320 Hz	.02	.02
•0.547 Hz	0.577 Hz	.14	.124
	0.646 Hz	.127	.135
	0.819 Hz	.004	--

•Denotes modes observed.* Not positively identified

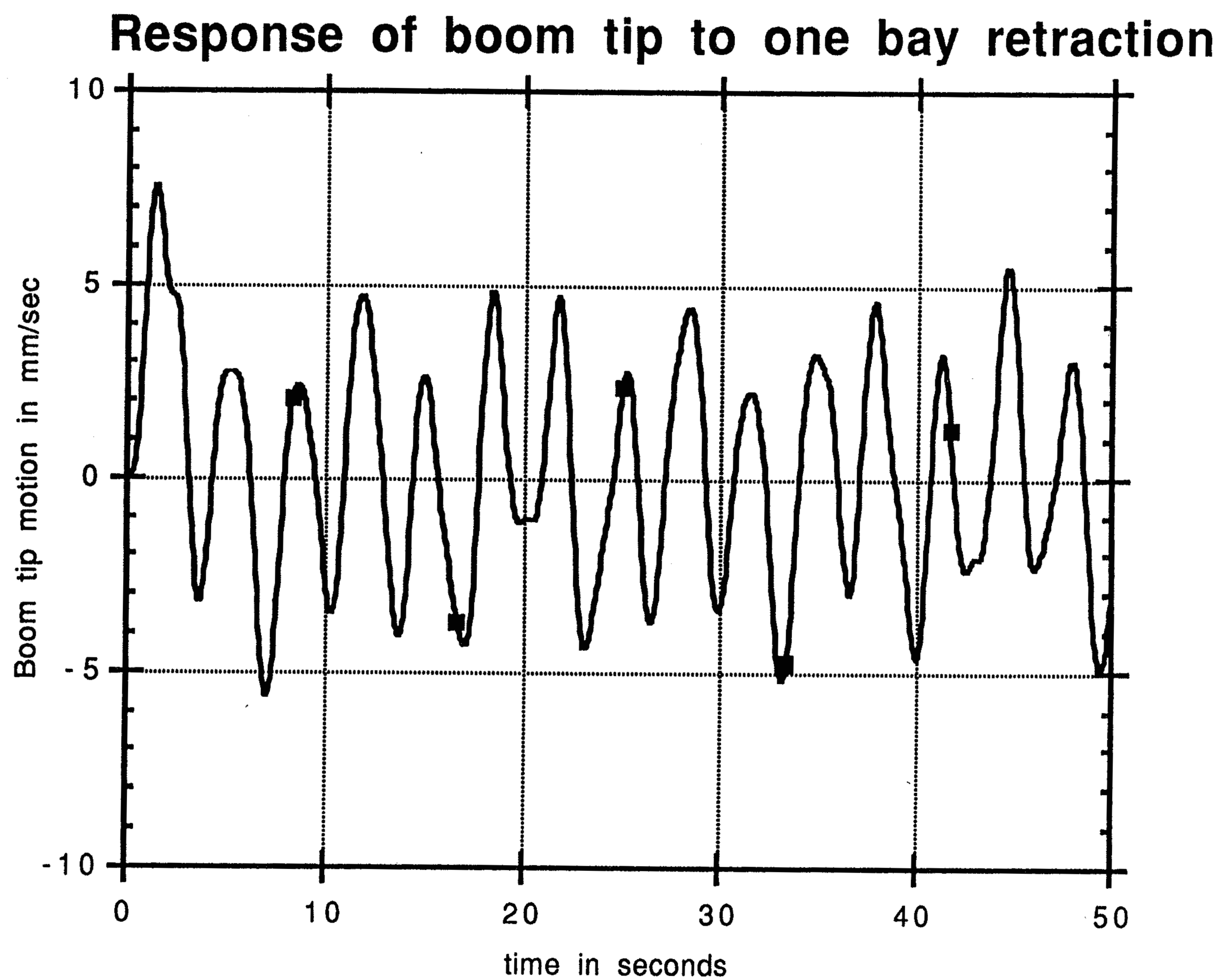
Use gravity-gradient boom for excitation

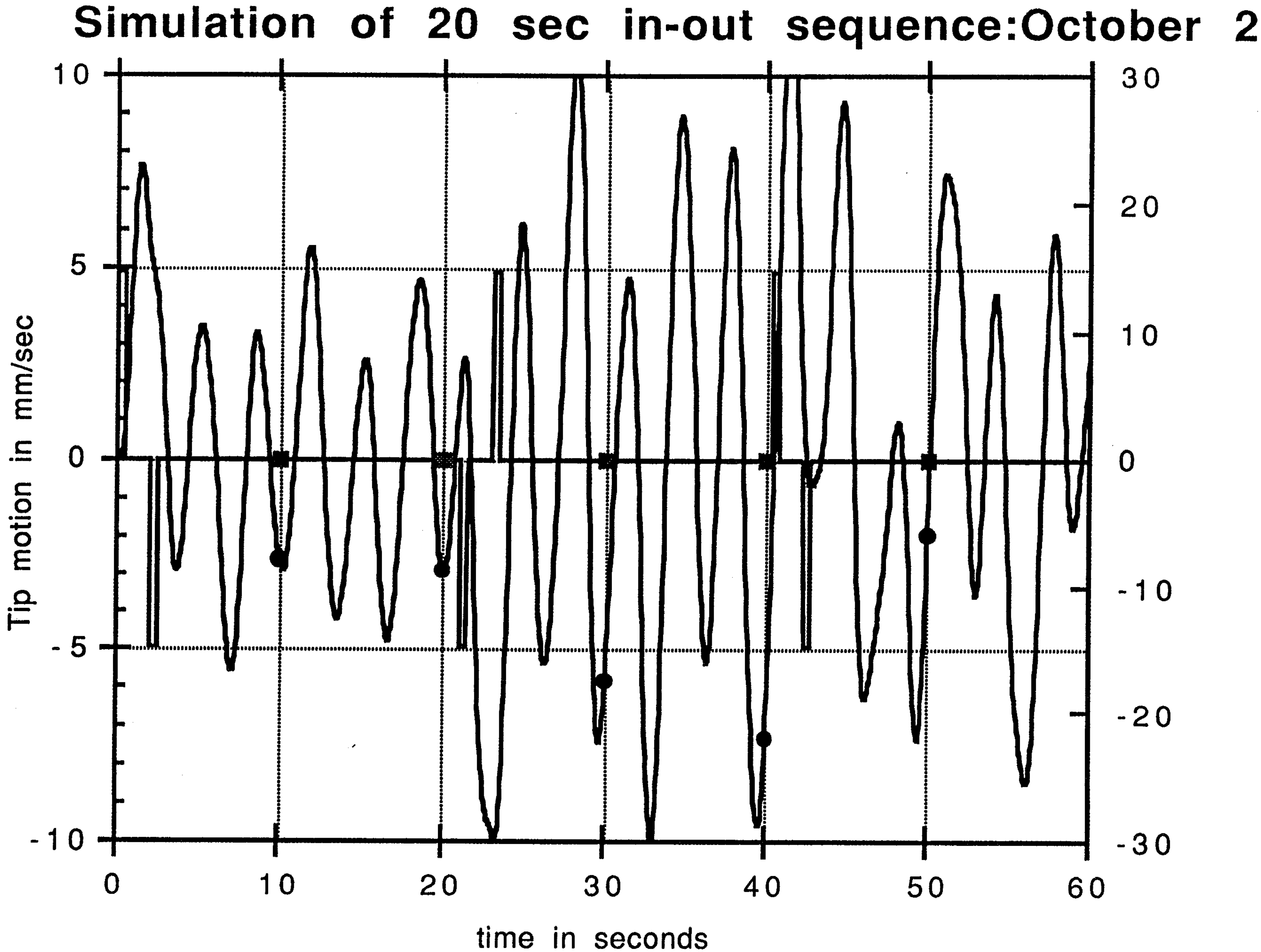
- Deploy or retract gravity-gradient boom 1 bay (6 inches)

    Gives 2 impulses to spacecraft.

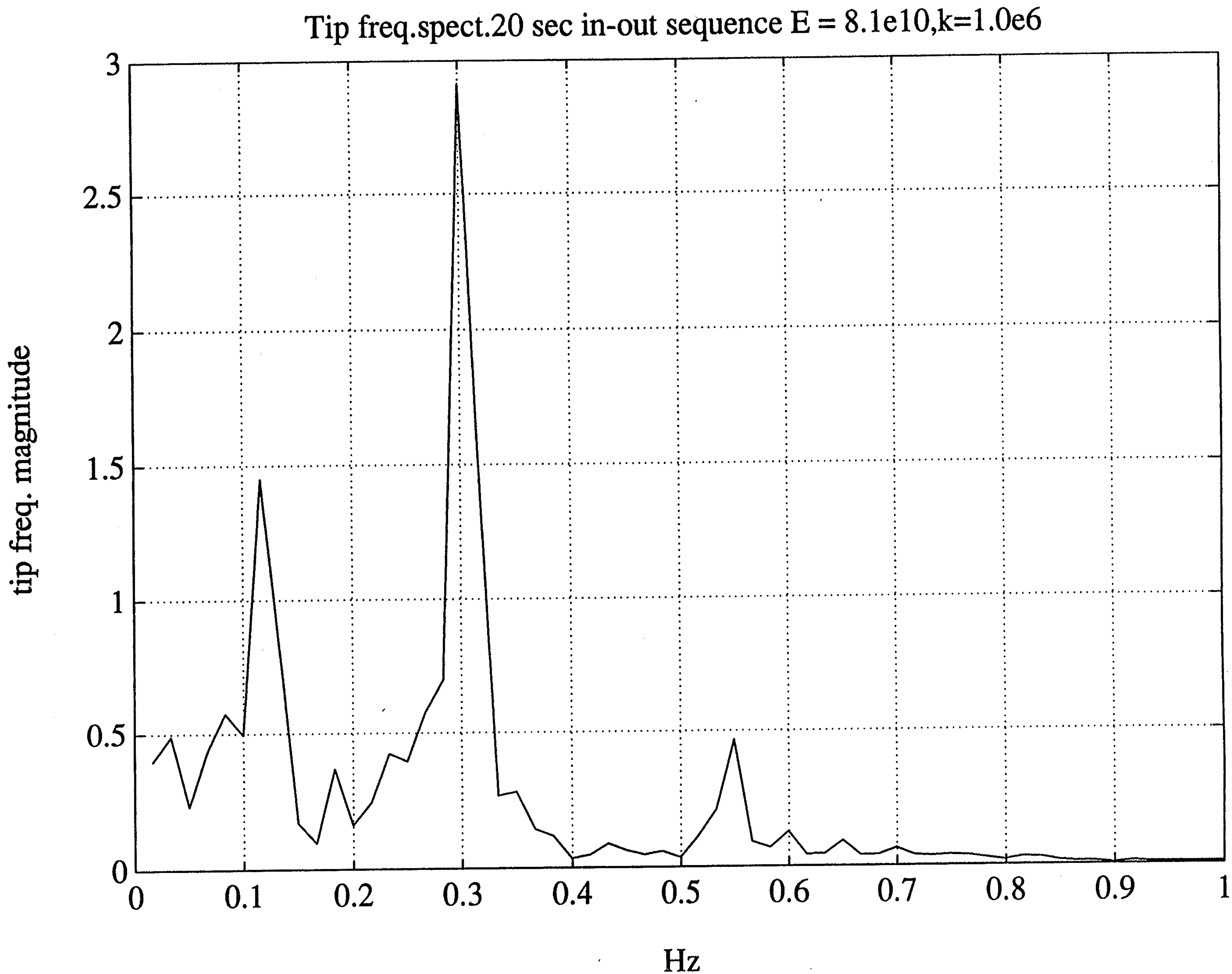


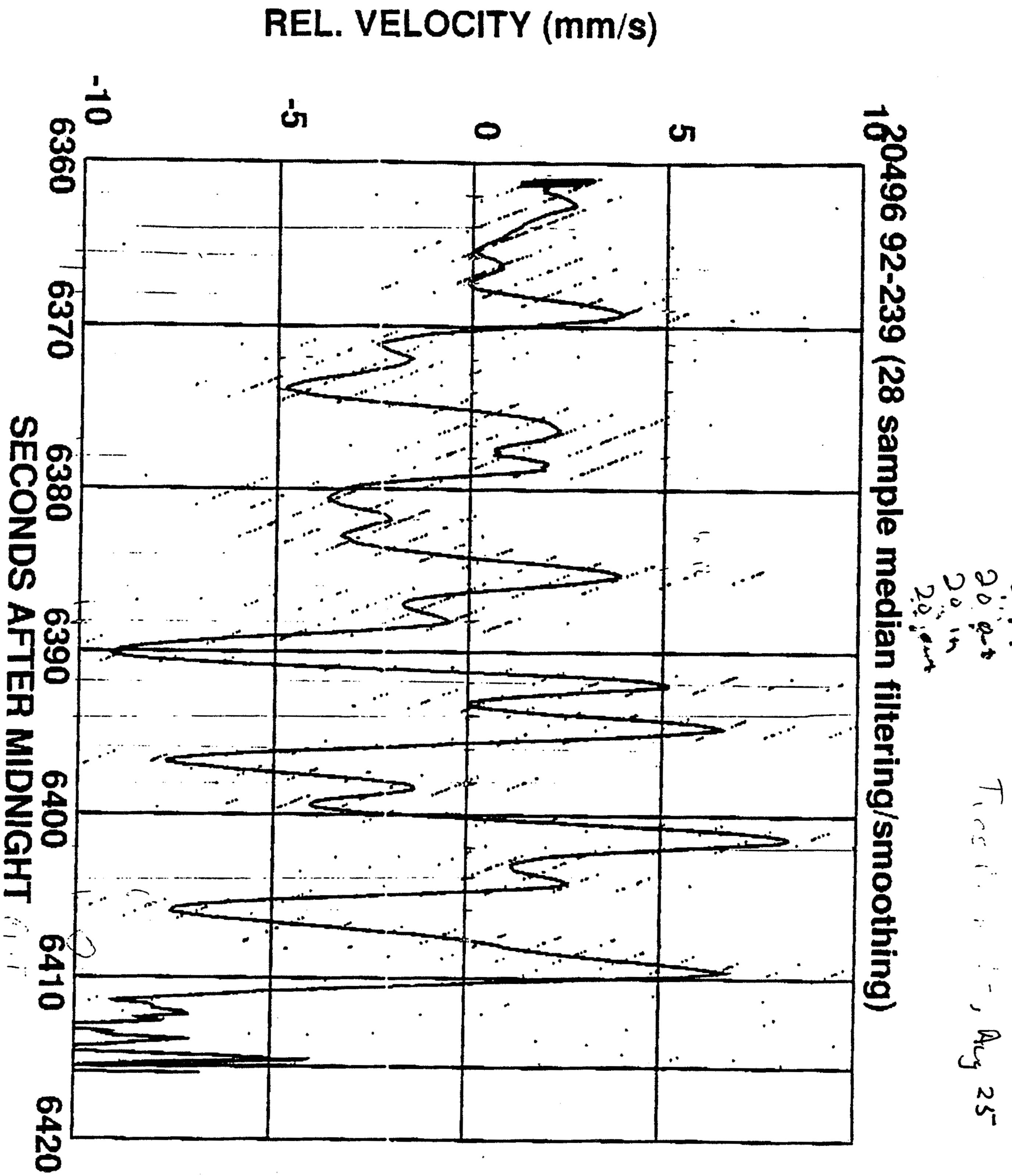
Boom starts up/stops in abt 0.5 sec
Retraction/deployment rate is abt 75 mm/sec = 3 in/sec


Response of boom tip to one bay retraction
Boom tip motion in mm/sec
10
5
0
-5
-10
0
10
20
30
40
50
time in seconds



Simulation of 20 sec in-out sequence:October 2
Tip motion in mm/sec
time in seconds
10
5
0
-5
-10
30
20
10
0
-10
-20
-30
0
10
20
30
40
50
60


Tip freq.spect.20 sec in-out sequence E = 8.1e10,k=1.0e6
tip freq. magnitude
Hz
0
0.5
1
1.5
2
2.5
3
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9



Aug 25

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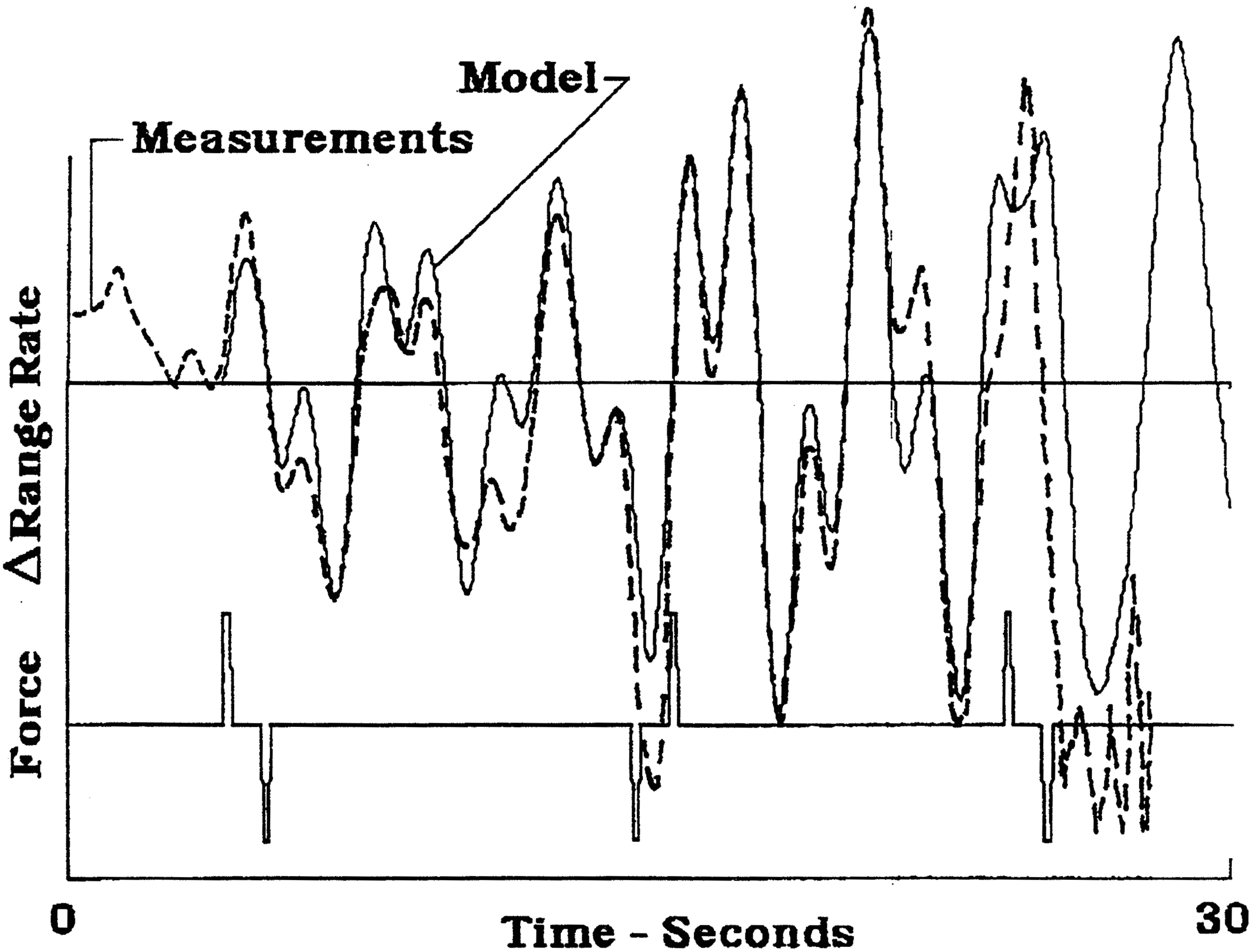
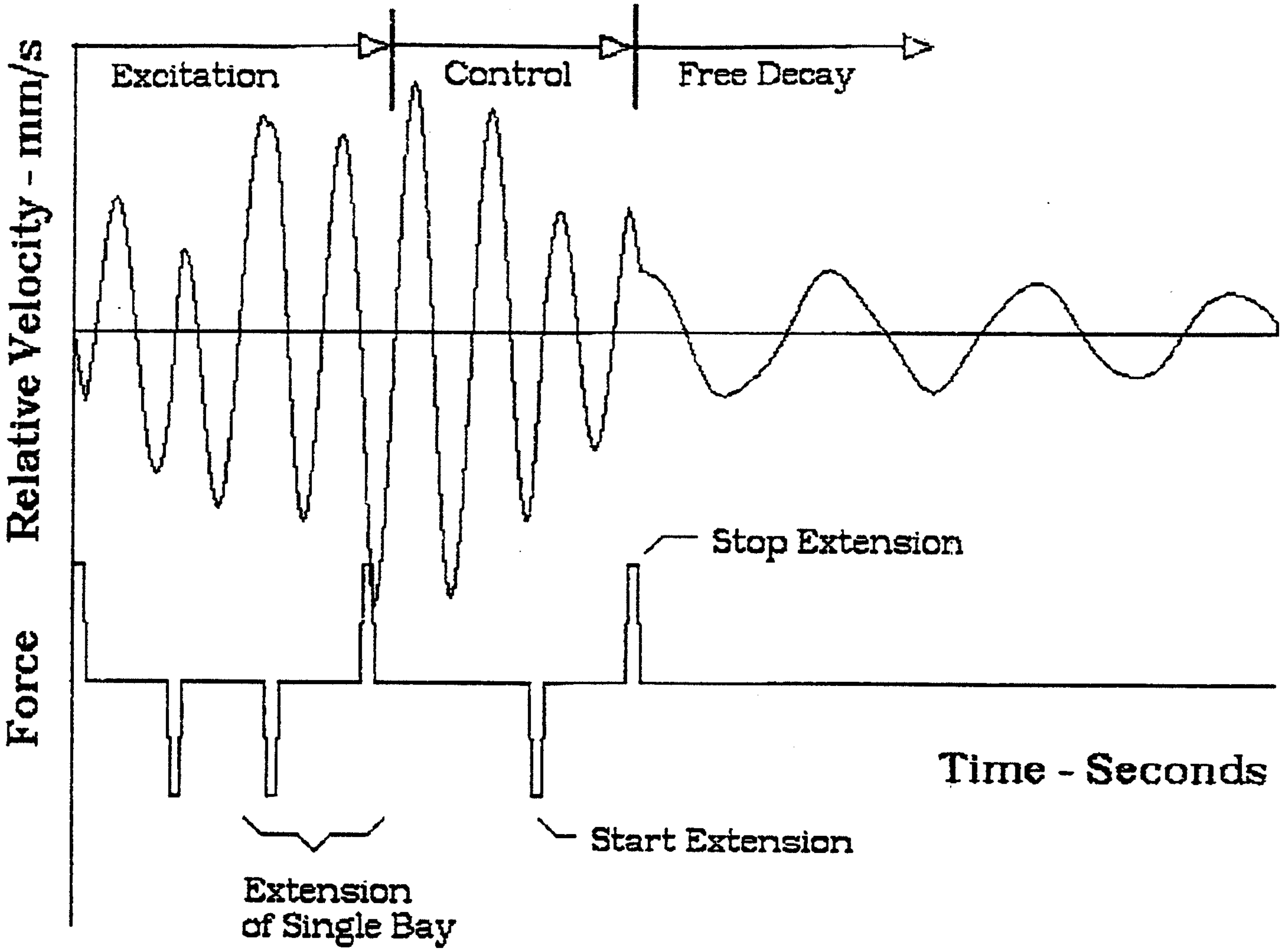
OF POOR QUALITY

Comparison of Responses



DAY 239



Targeted Open Loop Damping of .319 Hz Mode. Response of .125 Hz Mode Included.

Discussion of measurements and simulations

- Modes excited by the boom movements were quite observable by the Firepond ground-based system.

- Observations enabled refinements of models: FEM, PDE and modal models.

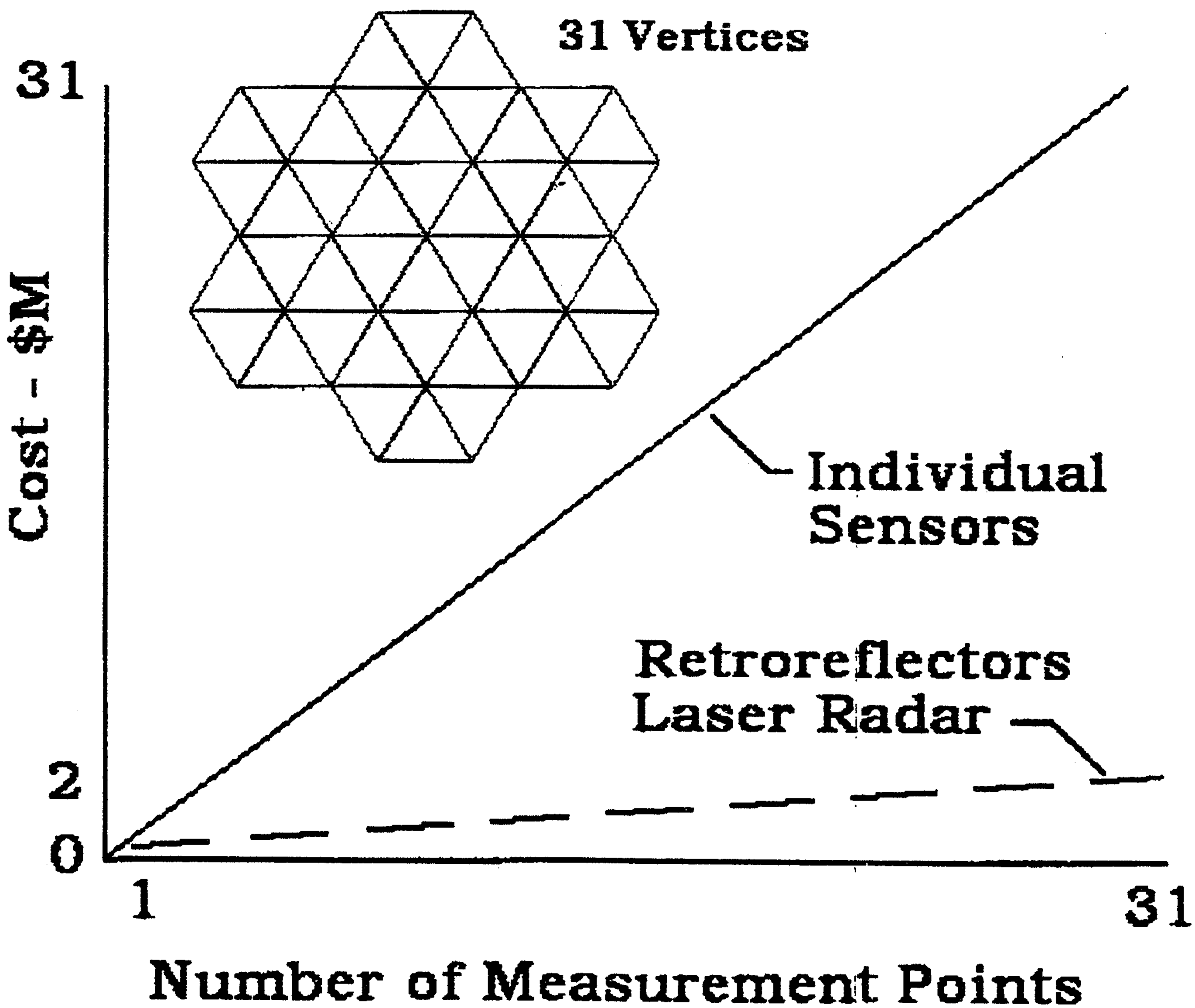
- Dynamics model gave good qualitative estimates of vibratior amplitudes (abt 10 %) and good prediction of which modes would be excited by the boom movements.

- Still refining our models: TREETOPS, DISCOS.

- Open-loop damping is difficult due to extreme sensitivity to system parameters and timing of boom movements.

- Method gives low-cost vibration measurements.

Cost Comparison



ACCOMPLISHMENTS OF LACE EXPERIMENT

- First ever ground-based laser measurements of vibrations of an orbiting satellite.

- Established feasibility of remote health monitoring & inspection of orbiting structures.

- Validated system modeling – but also found errors in model; showed need for flight experiments to validate modeling parameters.

CSI-Star: A Low-Cost CSI Orbital Testbed

Flight Experiments Technical Interchange Meeting

5 October 1992

D. EDBERG
(714) 896-5210

MCDONNELL DOUGLAS
SPACE SYSTEMS COMPANY

RATIONALE FOR AN ON-ORBIT CSI TEST FACILITY

ON-ORBIT OPEN- AND CLOSED-LOOP TESTING OF A CONTROLLED STRUCTURE TESTBED IS REQUIRED

- SPACECRAFT QUAL TESTS WILL USUALLY OCCUR ON THE GROUND WHERE GRAVITY AND SUSPENSION EFFECTS WILL CAUSE THE 1-G AND 0-G DYNAMICS TO DIFFER

- DIFFERENCES BETWEEN GROUND AND ON-ORBIT ENVIRONMENTS CAN SIGNIFICANTLY ALTER THE OPEN- AND CLOSED-LOOP BEHAVIOR, BOTH IN THE SHORT- AND LONG-TERM - UARS JITTER, HST DYNAMICS

- CONTROLLER STABILITY AND PERFORMANCE ROBUSTNESS REQUIRE MODEL FIDELITY THAT IS INTIMATELY RELATED TO THE LEVEL OF APPLIED CONTROL AUTHORITY - THE REQUIRED LEVEL OF 1-G AND 0-G CONTROL AUTHORITY WILL DIFFER

CSI FLIGHT EXPERIMENT OBJECTIVES

NASA

OVERALL OBJECTIVE: PROVIDE ON-ORBIT TESTBED NECESSARY TO DEVELOP & VERIFY CSI TECHNOLOGY TOOLS

ON-ORBIT DEMONSTRATION / VALIDATION OF CSI TECHNOLOGY FOR:

- HEALTH MONITORING AND SYSTEM IDENTIFICATION OF CHANGING ON-ORBIT CONFIGURATION (ENVIRONMENTAL EFFECTS AND DEPLOYMENT DYNAMICS)
- GLOBAL VIBRATION SUPPRESSION WITH FIXED GEOMETRY USING SEVERALTYPES OF ACTUATORS
- MICRO-AMPLITUDE VIBRATION SUPPRESSION DURING PRECISION PAYLOAD POINTING
- GLOBAL VIBRATION SUPPRESSION DURING PRECISION POINTING OF MULTIPLE INSTRUMENTS/PAYLOADS
- GLOBAL VIBRATION SUPPRESSION DURING LARGE ANGLE REORIENTATION SLEWING OF WHOLE TEST ARTICLE
- GLOBAL VIBRATION SUPPRESSION DURING LARGE ANGLE ARTICULATION OF PAYLOAD WITH FLEXIBLE APPENDAGES

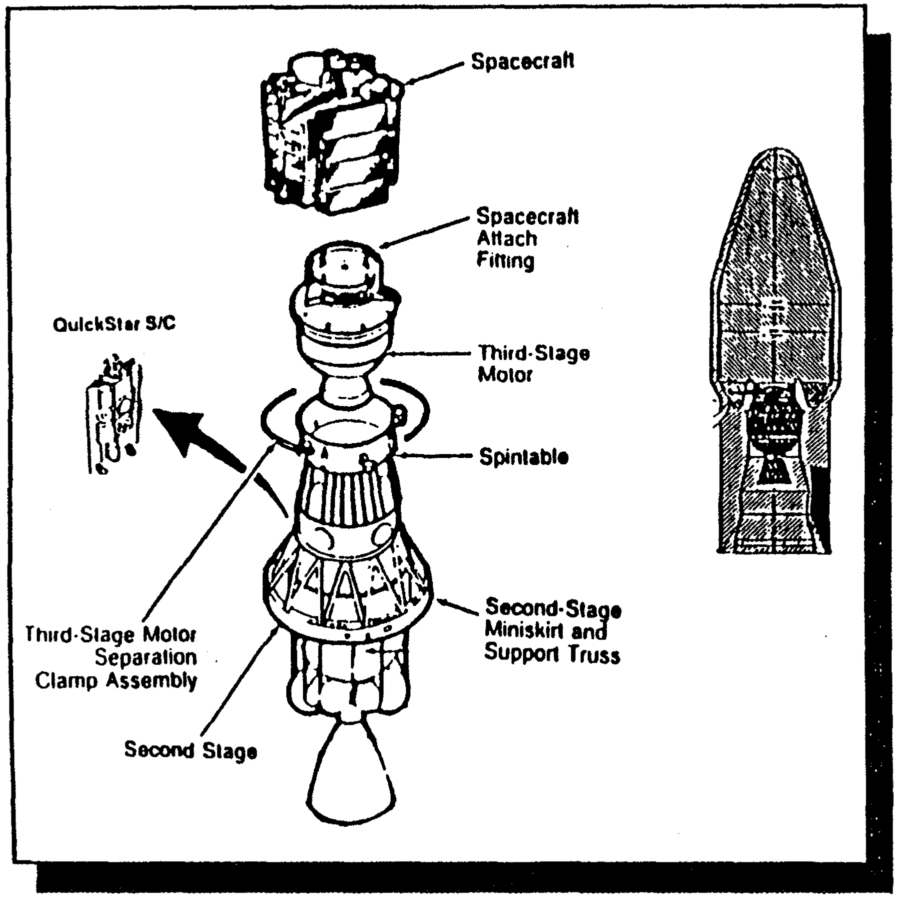
FEASIBILITY STUDY OBJECTIVES

- <u>OVERALL OBJECTIVE:</u> DEFINE A LOW-COST CSI ORBITAL FACILITY USING THE QUICKSTAR SPACECRAFT BUS LAUNCHED AS A DELTA II SECONDARY PAYLOAD
- DEFINE EXPERIMENT PACKAGES THAT ARE COMPLEMENTARY TO EXISTING FLIGHT PROGRAMS
- ALL DEFINED CONCEPTS SHALL HAVE THE FOLLOWING:
  - ON-ORBIT LIFE OF AT LEAST 1 YEAR
  - FIRST FLEX MODE BELOW 1 Hz
  - CLOSELY-SPACED/COUPLED FLEX MODES
  - SUFFICIENT SENSORS FOR QUALITY SYSTEM ID
  - REPROGRAMABLE CONTROL ALGORITHMS VIA UPLINK
- AT LEAST ONE DEFINED CONCEPT SHALL HAVE ALL THE FOLLOWING:
  - MULTIPLE, INTERACTING CONTROL SYSTEMS
  - OPTICAL PATH LENGTH OR LINE-OF-SIGHT CONTROL
  - AUTOMATED ALIGNMENT/CALIBRATION OF SENSORS/ACTUATORS
  - SUB-ARC-MINUTE PAYLOAD OR SPACECRAFT POINTING
- PRESENT NASA WITH REALISTIC COST AND RISK ESTIMATES

CSI FREE-FLYER SOLUTION

NASA

QUICKSTAR SPACECRAFT BUS AS DELTA II SECONDARY PAYLOAD



- MAJOR PRIMARY PAYLOAD IMPACT ISSUES RESOLVED (LOSAT-X)
- QUICKSTAR CAPABILITIES GREATER THAN TYPICAL SMALLSATS
- MANY LAUNCH OPPORTUNITIES EXIST
- LOW BUS COST ~ $10M - $15M
- LOW LAUNCH COSTS ~ $2M - $3M
- CURRENTLY THE ONLY FLOWN, SECONDARY S/C BUS AVAILABLE

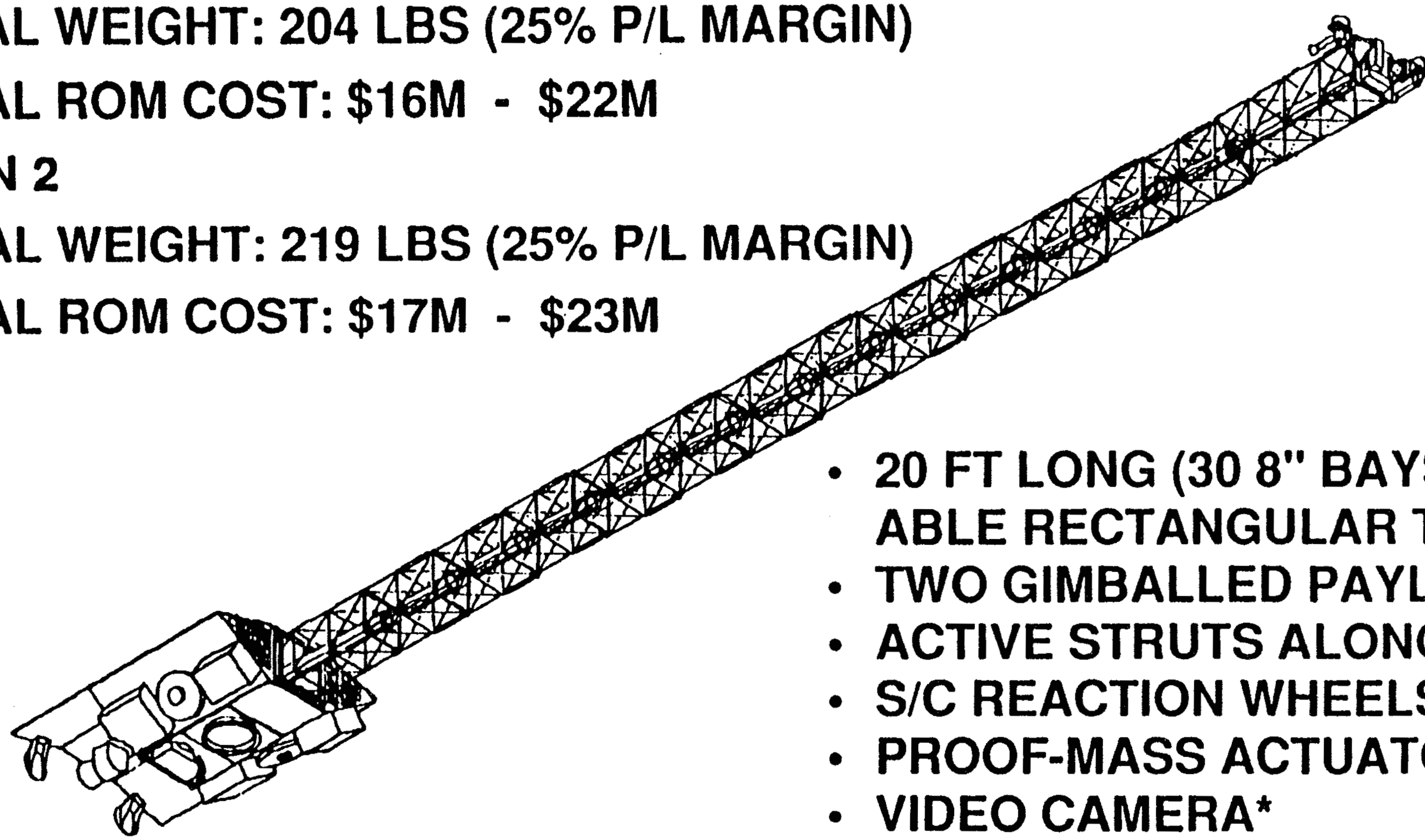
FEASIBILITY STUDY TECHNICAL STATUS SUMMARY (AS OF 6-30-92)

- QUICKSTAR S/C BUS REQUIREMENTS DEVELOPED
- SEVERAL DELTA II LAUNCH OPPORTUNITIES IDENTIFIED
  - TWO IN 1997 BASELINED
- CANDIDATE EXPERIMENT COMPONENTS IDENTIFIED
- TWO EXPERIMENT PACKAGES CONCEPTUALLY DESIGNED
- COMPONENT SIZING COMPLETED (FIRST PASS)
- EXPERIMENT ROM COST ESTIMATES (FIRST PASS) COMPLETED
- S/C BUS INTEGRATION DESIGN (FIRST PASS) ACTIVITIES COMPLETED
- S/C BUS REQUIRED CONTROL FUNCTIONS VERIFIED (FIRST PASS)
- EVALUATIONS OF CONCEPTS IN PROGRESS

CSI-Star
A LOW-COST CSI FREE FLYER

NASA

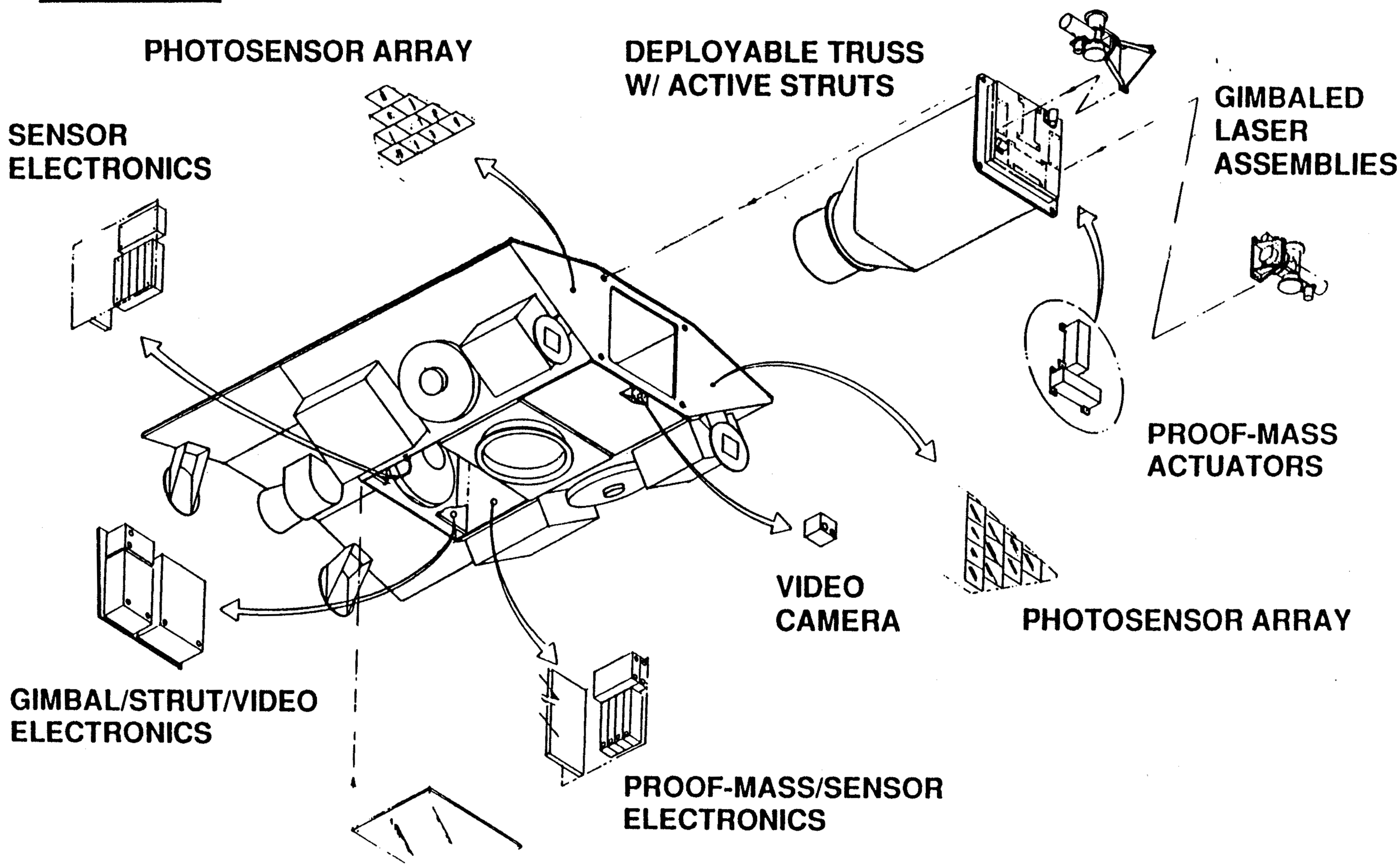
- TWO EXPERIMENT PACKAGE OPTIONS - WEIGHT DRIVEN
- BASELINE (OPTION 1)
  - TOTAL WEIGHT: 204 LBS (25% P/L MARGIN)
  - TOTAL ROM COST: $16M - $22M
- OPTION 2
  - TOTAL WEIGHT: 219 LBS (25% P/L MARGIN)
  - TOTAL ROM COST: $17M - $23M



- 20 FT LONG (30 8" BAYS) DEPLOYABLE RECTANGULAR TRUSS
- TWO GIMBALLED PAYLOADS
- ACTIVE STRUTS ALONG TRUSS
- S/C REACTION WHEELS
- PROOF-MASS ACTUATORS*
- VIDEO CAMERA*
- PZT PAYLOAD ISOLATOR TRIPOD*

* OPTION 2 ADDITIONS

CONCEPTUAL EXPERIMENT DESIGN
OPTION 2 CONFIGURATION



EXPERIMENT DESIGN DEPENDS ON PRIMARY MISSION & BUS CAPABILITIES AND COST CONSTRAINTS

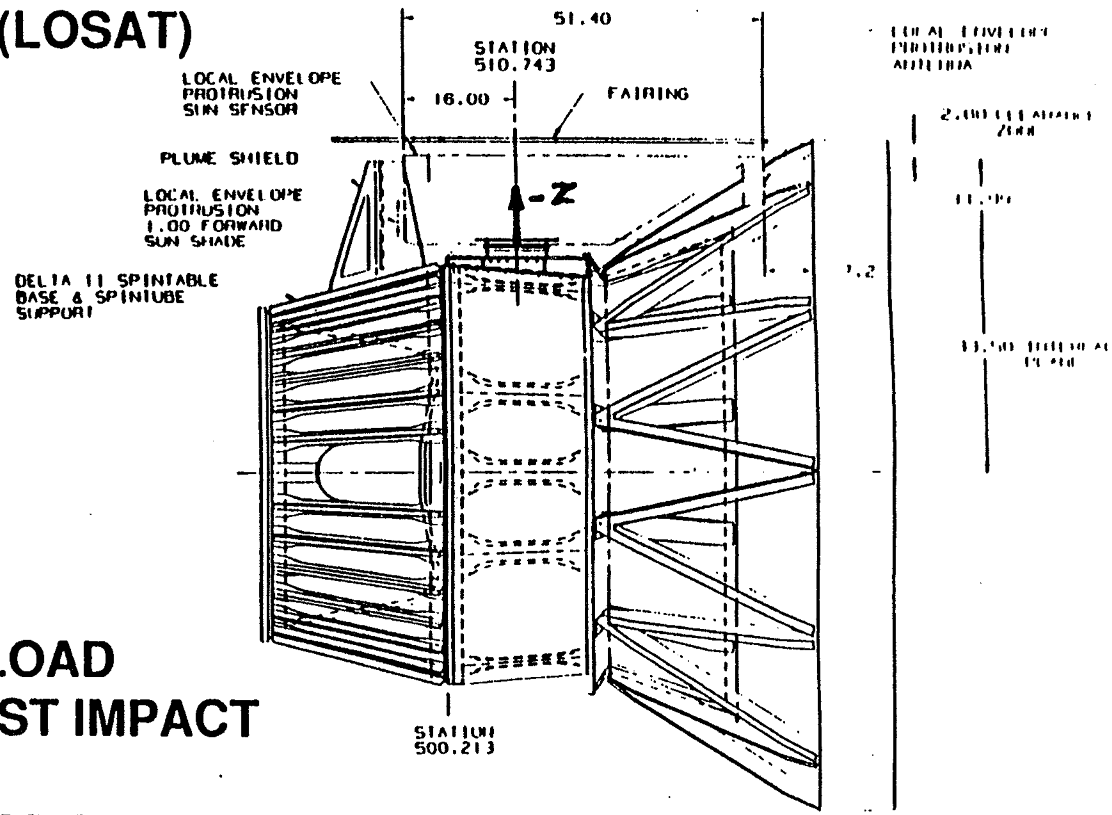
- PRIMARY MISSION
  - WEIGHT MARGIN
  - ORBIT
  - LAUNCH VEHICLE CAPABILITIES
    - CLAMPBAND LIMITS
    - C.G. LIMITS
- S/C BUS CAPABILITIES
  - DELTA-V (IF NEEDED)
  - CPU THROUGHPUT
  - PAYLOAD VOLUME
  - POWER
- COST CONSTRAINTS
  - "OFF-THE-SHELF" H/W
  - MINIMAL H/W DEVELOPMENT

DELTA II - QUICKSTAR INTERFACE CLAMPBAND CAPABILITY

CSI NASA

- CLAMPBAND CAPABILITY DETERMINES MAX PAYLOAD WEIGHT

- CURRENT CLAMPBAND DESIGN (LOSAT)
  - 200 LBS PAYLOAD
  - ±1.0", ±1.0", -6.0" PAYLOAD C.G. ENVELOPE
  - STATIC LOAD TEST REQUIRED
    - MINIMAL COST IMPACT



- REDESIGNED CLAMPBAND
  - 220 LBS PAYLOAD
  - ±1.0", ±1.0", -6.0" PAYLOAD C.G. ENVELOPE
  - REDESIGN EFFORT & STATIC LOAD TEST REQUIRED - MINIMAL COST IMPACT

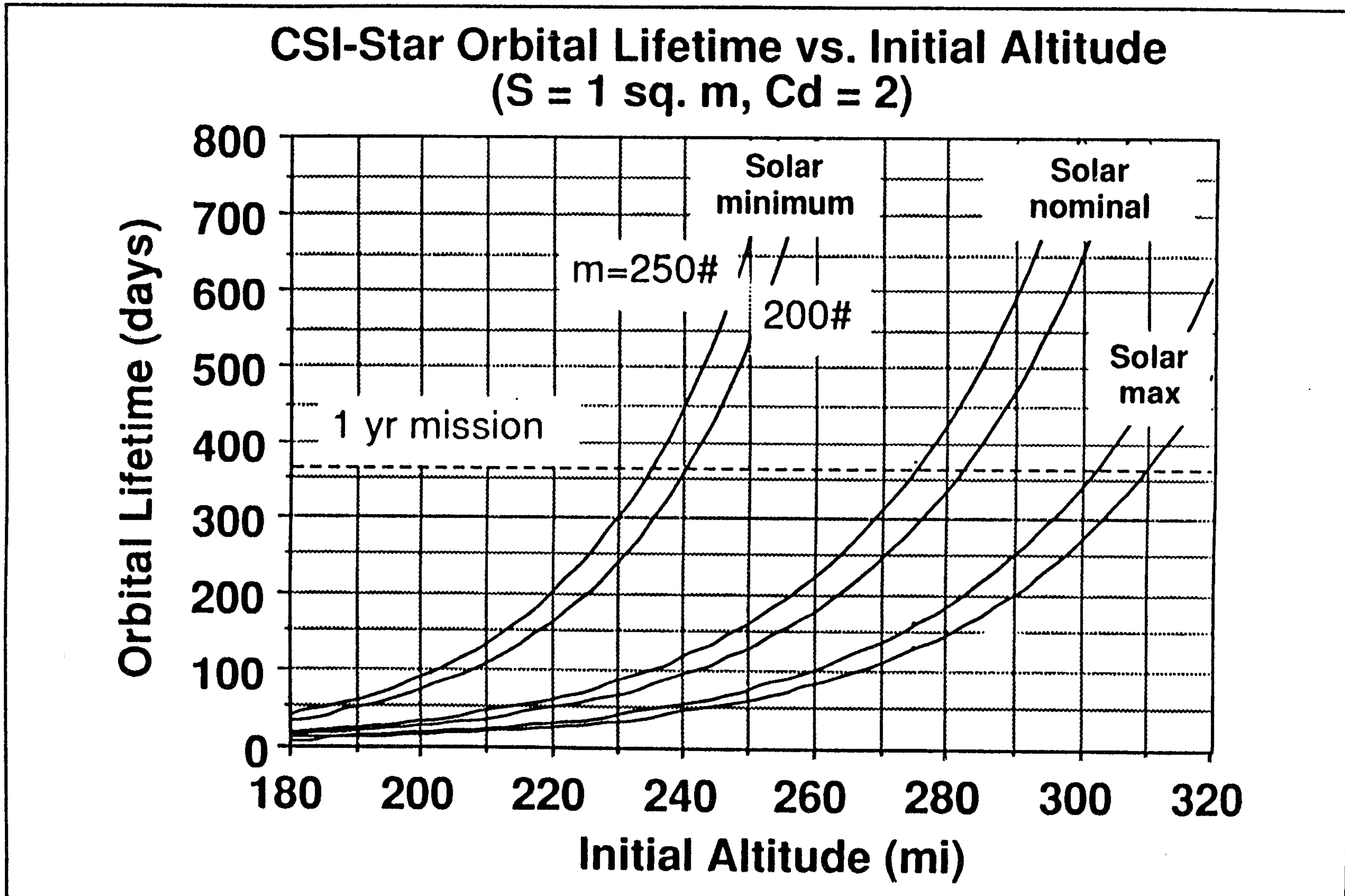
- GREATER THAN 220 LBS PAYLOAD REQUIRES GUIDANCE SECTION IMPACT TO BE CONSIDERED - SIGNIFICANT COSTS POSSIBLE

1997 BASELINE
DELTA LAUNCH OPPORTUNITIES

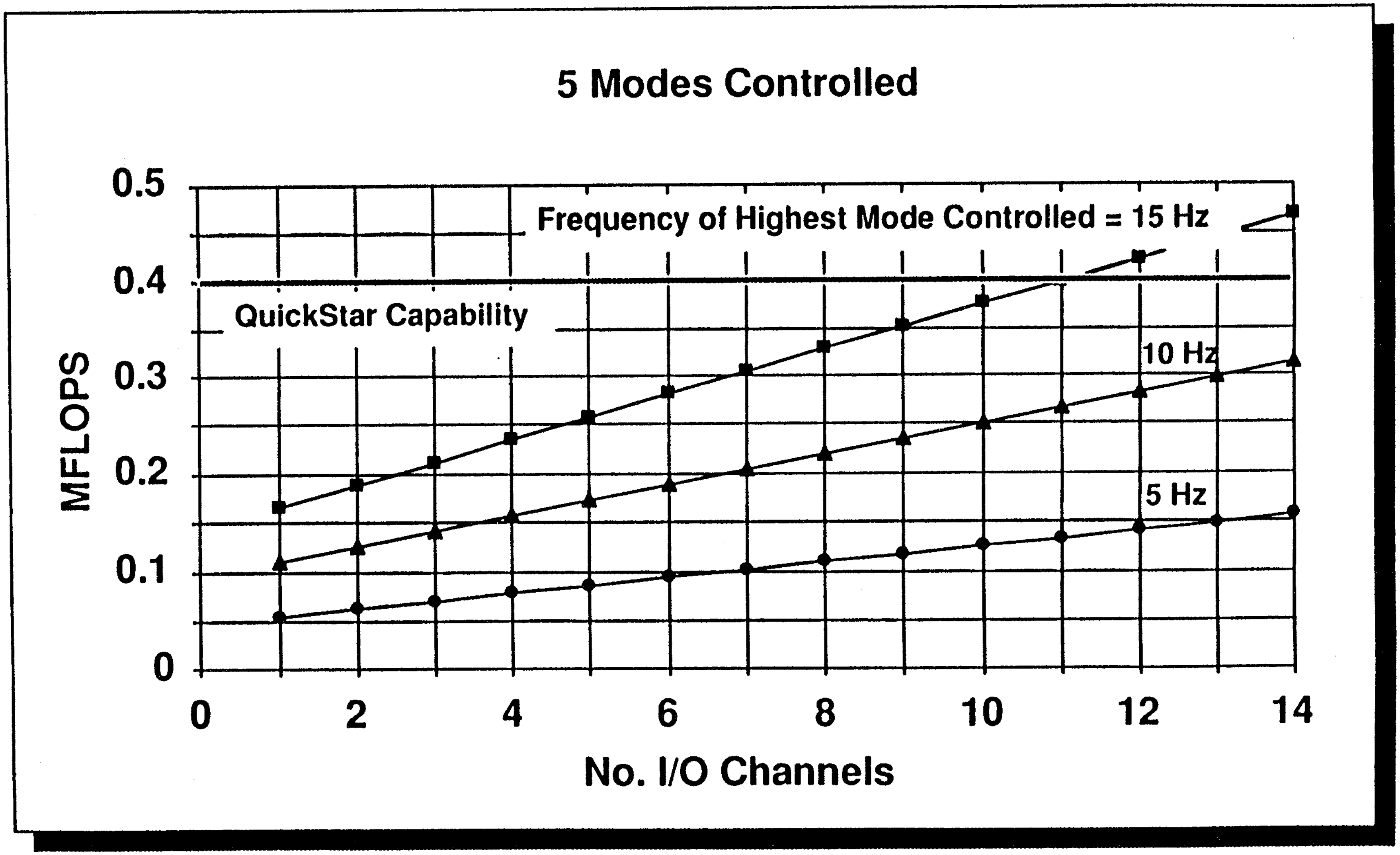
- ACE MISSION
  - LAUNCH DATE: JULY 1997
  - ORBIT ALTITUDE: 167 KM (100 MI) X L1 POINT
  - ORBIT INCLINATION: 28.7 DEG
  - WEIGHT MARGIN: 1415 LBS

- ATMOS MISSION
  - LAUNCH DATE: OCTOBER 1997
  - ORBIT ALTITUDE: 792 KM (475 MI) CIRCULAR
  - ORBIT INCLINATION: 98.5 DEG (SUN SYNCHRONOUS)
  - WEIGHT MARGIN: 400 LBS

CSI-Star CAN ACHIEVE ORBITAL LIFETIME OF MORE THAN 1 YEAR

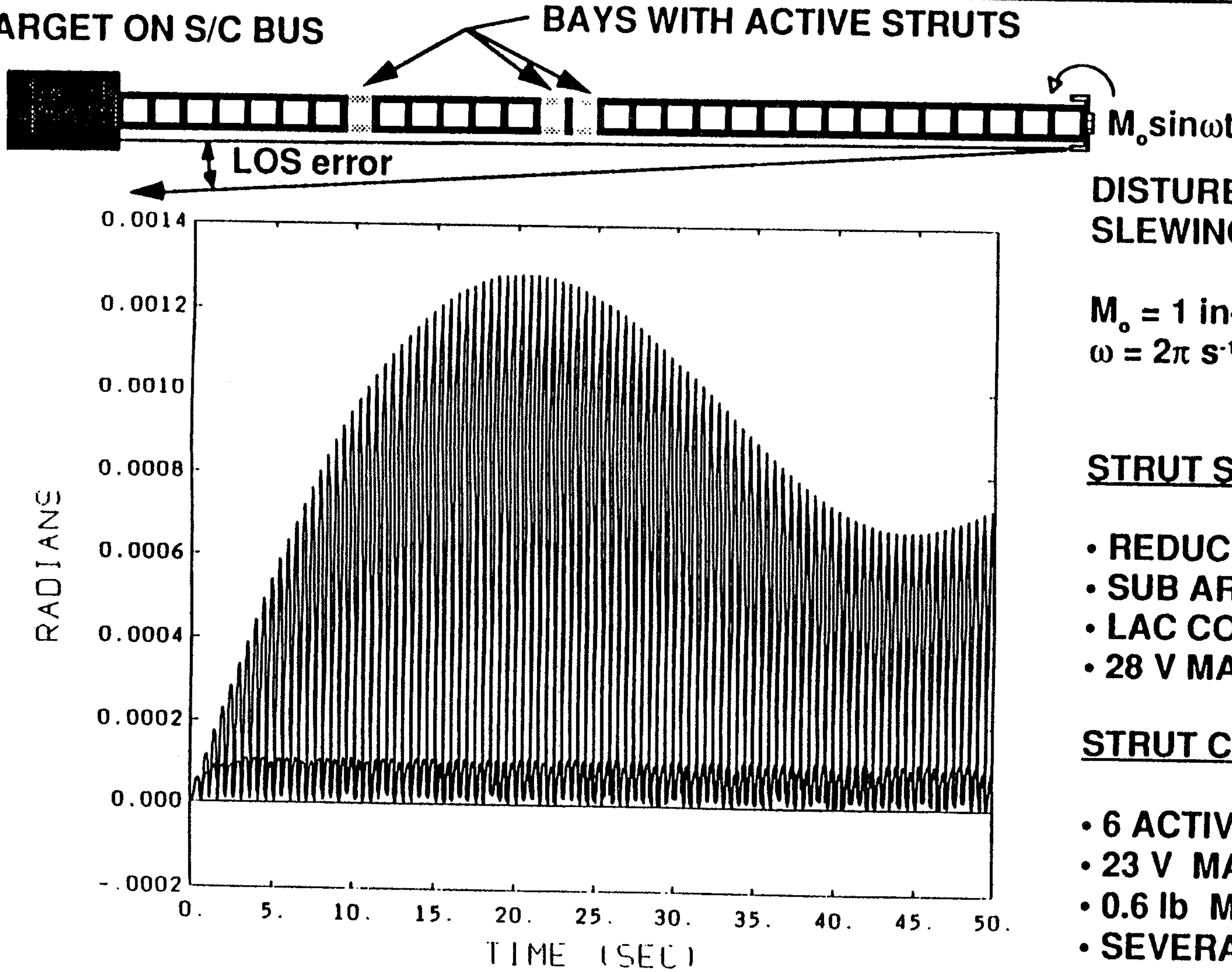


QUICKSTAR BUS CPU CAN PROVIDE THE REQUIRED THROUGHPUT



OPEN & CLOSED LOOP RESPONSE OF BASELINED TRUSS WITH ACTIVE STRUTS

CSI NASA



DISTURBANCE APPLIED BY SLEWING PAYLOAD WITH

M_o = 1 in-lb
$\omega = 2\pi\ s^{-1}$ (f = 1 Hz)

STRUT SIZING REQUIREMENTS

- REDUCE RESPONSE BY 10X
- SUB ARC-MINUTE POINTING
- LAC CONTROLLER
- 28 V MAX

STRUT CHARACTERISTICS

- 6 ACTIVE STRUT PAIRS
- 23 V MAX VOLTAGE
- 0.6 lb MAX FORCE
- SEVERAL LOW V, F OPTIONS

CSI

EXPERIMENT WEIGHT
BASELINE (OPTION 1) CONFIGURATION

NASA

DESCRIPTION	SIZE	UNIT WEIGHT* (LBS)	QUANTITY REQUIRED	TOTALWEIGHT (LBS)
ACCELEROMETER	.625 HEX,.38 HI	0.01	20	0.20
STRAIN GAGE	small	0.01	20	0.20
ACTIVE STRUT, ASTROMAST	.5DIA X 6.0	0.25	12	3.00
TEMP SENSORS	small	0.05	6	0.30
ASTROMAST	9.2 X 9.2 X 16.8	21.00	1	21.00
PHOTO SENSOR	2.8 X 2.8 X 0.4	0.18	10	1.80
STRUT ELECTRONICS	6.5 X 4.0 X 2.5	2.25	1	2.25
SENSOR ELECTRONICS	5.5 X 1.2 X 0.9	0.20	9	1.80
DISTURB PAYLOAD	1.5 DIA X 4.0	2.00	1	2.00
LASER	1.5 DIA X 4.0	2.10	1	2.10
GIMBAL ASSY, 2 AXIS	BALL ESTIMATE	3.70	1	3.70
GIMBAL ASSY, 1 AXIS	BALL ESTIMATE	1.90	1	1.90
GIMBAL DR. ELECTRONICS	6.0 X 8.0 X 1.0	2.00	2	4.00
LASER ELECTRONICS	3.0 X 5.0 X 1.0	0.50	1	0.50
STRUCTURE	ENG. ESTIMATE	0.88	3	0.88
MISC.	ENG. ESTIMATE	0.38	A/R	0.38
MLI	PAST EXPERIENCE	.084 #/Ft2	3.75 Ft2	0.32
HARNESS	15 X.125#/ CONN	1.4#/1000FT	3112 Ft	6.23
TOTAL				52.56

* WEIGHTS ESTIMATED WITH A MINIMUM 25% MARGIN

EXPERIMENT WEIGHT
OPTION 2 CONFIGURATION

DESCRIPTION	SIZE	UNIT WEIGHT (LBS)	QUANTITY REQUIRED	TOTALWEIGHT (LBS)
ACCELEROMETER	.625 HEX,.38 HI	0.01	20	0.20
LOAD CELL	.67DIA X 1.23	0.10	2	0.20
STRAIN GAGE	small	0.01	20	0.20
PROOF MASS ACT.	5.0 X 1.5 X 1.5	3.00	2	6.00
ACTIVE STRUT, ASTROMAST	.5DIA X 6.0	0.25	12	3.00
VIDEO CAMERA	2.0 X 2.0 X 2.3	1.00	1	1.00
TEMP SENSORS	small	0.05	6	0.30
ASTROMAST	8.6 X 8.6 X 27	21.00	1	21.00
PHOTO SENSOR	2.8 X 2.8 X 0.4	0.18	20	3.60
STRUT ELECTRONICS	6.5 X 4.0 X 2.5	2.25	1	2.25
SENSOR ELECTRONICS	5.5 X 1.2 X 0.9	0.20	9	1.80
PROOF MASS ELECTRONICS	3.0 x 5.0 x 1.0	0.50	2	1.00
TRIPOD, ACTIVE LASER	.38DIA X 2.5	1.00	1	1.00
LASER	1.5 DIA X 4.0	2.10	2	4.20
GIMBAL ASSY, 2 AXIS	BALL ESTIMATE	3.70	1	3.70
GIMBAL ASSY, 1 AXIS	BALL ESTIMATE	1.90	1	1.90
GIMBAL DR. ELECTRONICS	6.0 X 8.0 X 3.0	2.00	2	4.00
VIDEO ELECTRONICS	4.0 X 2.6 X 2.6	4.00	1	4.00
LASER ELECTRONICS	3.0 X 5.0 X 1.0	0.50	1	0.50
STRUCTURE	ENG. ESTIMATE	0.88	3	0.88
MISC.	ENG. ESTIMATE	0.38	A/R	0.38
MLI	PAST EXPERIENCE	.084 #/Ft2	3.75 Ft2	0.32
HARNESS	20 X.125#/ CONN	1.4#/1000FT	3512 Ft	7.40
TOTAL				68.83

* WEIGHTS ESTIMATED WITH A MINIMUM 25% MARGIN

EXPERIMENT POWER
BASELINE (OPTION 1) CONFIGURATION

DESCRIPTION	POWER (W)	QUANTITY REQUIRED	TOTAL POWER (W)
ACCELEROMETER	0.06	20	1.12
ACTIVE STRUT, ASTROMAST	0.15	12	1.86
ASTROMAST DRIVE ELECT.	20.00	1	20.00
GIMBAL DRIVE ELECTRONICS	2.00	1	2.00
GIMBAL DRIVE MOTOR	2.00	3	6.00
INSTRUMENT HEATING	10.00	1	10.00
LASER POWER	0.90	1	0.90
LASER ELECTRONICS	3.00	1	3.00
PHOTO SENSOR	0.03	10	0.25
SENSOR ELECTRONICS	0.00	9	0.01
STRAIN GAGES	0.13	20	2.50
STRUT ELECTRONICS	0.13	1	0.13
TEMP SENSORS	0.00	6	0.00
TOTAL			47.77

EXPERIMENT POWER
OPTION 2 CONFIGURARATION

DESCRIPTION	POWER (W)	QUANTITY REQUIRED	TOTAL POWER (W)
ACCELEROMETER	0.06	20	1.12
ACTIVE STRUT, ASTROMAST	0.15	12	1.86
ASTROMAST DRIVE ELECT.	20.00	1	20.00
GIMBAL DRIVE ELECTRONICS	2.00	3	6.00
GIMBAL DRIVE MOTOR	2.00	3	6.00
INSTRUMENT HEATING	10.00	1	10.00
LASER POWER	0.90	2	1.80
LASER ELECTRONICS	3.00	1	3.00
LOAD CELL	0.90	2	1.80
PHOTO SENSOR	0.03	20	0.50
PROOF MASS ACTUATOR	9.04	2	18.08
PROOF MASS ELECTRONICS	0.08	1	0.08
SENSOR ELECTRONICS	0.00	9	0.01
STRAIN GAGES	0.13	20	2.50
STRUT ELECTRONICS	0.25	1	0.25
TEMP SENSORS	0.00	6	0.00
VIDEO CAMERA	5.00	1	5.00
VIDEO ELECTRONICS	15.00	1	15.00
TOTAL			93.00

CSI QuickStar Capabilities/Requirements

QuickStar Characteristic	QuickStar Capability	Payload Requirement
• Payload Weight	70 lbm	52.66 lbm (option 1) 67.11 lbm (option 2)
• Onboard processing	2 MIPS	0.9 MIPS
• Data storage	500 Mbits	80 - 400 Mbits
• D/L data rate	1 Mbps	0.83 Mbps
• Payload power	75 W Peak 40 W orbit ave	28 W (option 1) 58 W (option 2)

Configuraton Requirements

• Use Quickstar bus	Meets reqt
• Fit in space between Guidance section and fairing	Needs work
• Package CSI experiment	Meets reqt
• Incorporate longer torque rods	Meets reqt
• Option 1 weight $\leq$ 200 lbs	204 lbs
• Option 2 weight $\leq$ 220 lbs	219 lbs
• C.G. at Z $\leq$ 6.0 inches	5.9 inches

Attitude Determination and Control

- Moments of inertia vs boom length and tip mass
- Disturbance torques
- Management of disturbance torques
- Maneuverability of deployed system
- Control performance
- System stability

Torque Management Results

- Moment of inertia ~500 kg • m^2 with 25 lb tip mass and 20 ft boom
- Disturbance torques managed with 30 Amp • m^2 control magents
- Wheel capacity of 1.0 N • m • s is adequate
- System is maneuverable at 60° in ~10 minutes

Baseline Control Parameters

- 0.05 Hz effective bandwidth (low)
- 10 Hz gyro sampling rate
- 20 msec gyro sampling (transport) delay
- 3 arcsec (15 microrad) rate quantization
- 10 Hz wheel torque commanding rate
- 6 oz-in (0.042 N • m) max wheel torque
- 1 Hz first boom free-free mode

System is highly stable and contributes to boom damping.

CSI QuickStar Bus ROM Costs

Based on our current understanding of the mission and payload requirements, the cost of a Ball QuickStar spacecraft for the CSI vehicle would range from

$9.6 to $12.2 M*.

* Note:
- Does not include launch vehicle integration , launch vehicle, or operations costs.
- A more refined estimate can be provided as the specifications and interfaces are more fully defined.

WORK REMAINING

- RECHECK WEIGHT & POWER ESTIMATES
- IMPROVE COST ESTIMATES
- RISK ASSESSMENT
- EVALUATION OF EXPERIMENT OPTIONS
  - DATA POTENTIAL FOR TYPICAL CSI TESTS
  - BENEFITS STUDY
- FINAL BRIEFING & REPORT

SUMMARY

- OUR PRIMARY OBJECTIVE IS TO DEFINE A FREE-FLYING CONCEPT THAT SATISFIES MOST CSI OBJECTIVES FOR THE LOWEST COST
- CSI-Star IS A TECHNOLOGY DEMONSTRATION NOT A TECHNOLOGY DEVELOPMENT PROGRAM, I.E., ALL H/W USED WILL HAVE BEEN PREVIOUSLY DEVELOPED
- QUICKSTAR'S FLIGHT-TESTED CAPABILITIES OFFER THE BEST POTENTIAL FOR A LOW-COST CSI FLIGHT EXPERIMENT
- LOSAT-X DEMONSTRATED NO ADVERSE IMPACT TO DELTA LAUNCH VEHICLE OR THE PRIMARY PAYLOAD, WHICH INCREASES CSI-STAR'S CHANCES OF FINDING A RIDE
- OUR STUDY END PRODUCT WILL BE 2 OR 3 LOW-COST CONCEPTS FOR AN ORBITAL CSI FACILITY THAT INCLUDES REALISTIC COST AND RISK ESTIMATES THAT WILL ALLOW NASA TO DETERMINE IF THE POTENTIAL DATA RETURN JUSTIFIES THE COST AND THAT THE RISK IS ACCEPTABLE

UPPER ATMOSPHERE RESEARCH SATELLITE JITTER STUDY

STANLEY WOODARD *
MIKE GARNEK **
JOHN MOLNAR **
WILLIAM GRANTHAM *

* GUIDANCE AND CONTROL DIVISION
NASA LANGLEY RESEARCH CENTER
** GENERAL ELECTRIC ASTRO SPACE DIVISION

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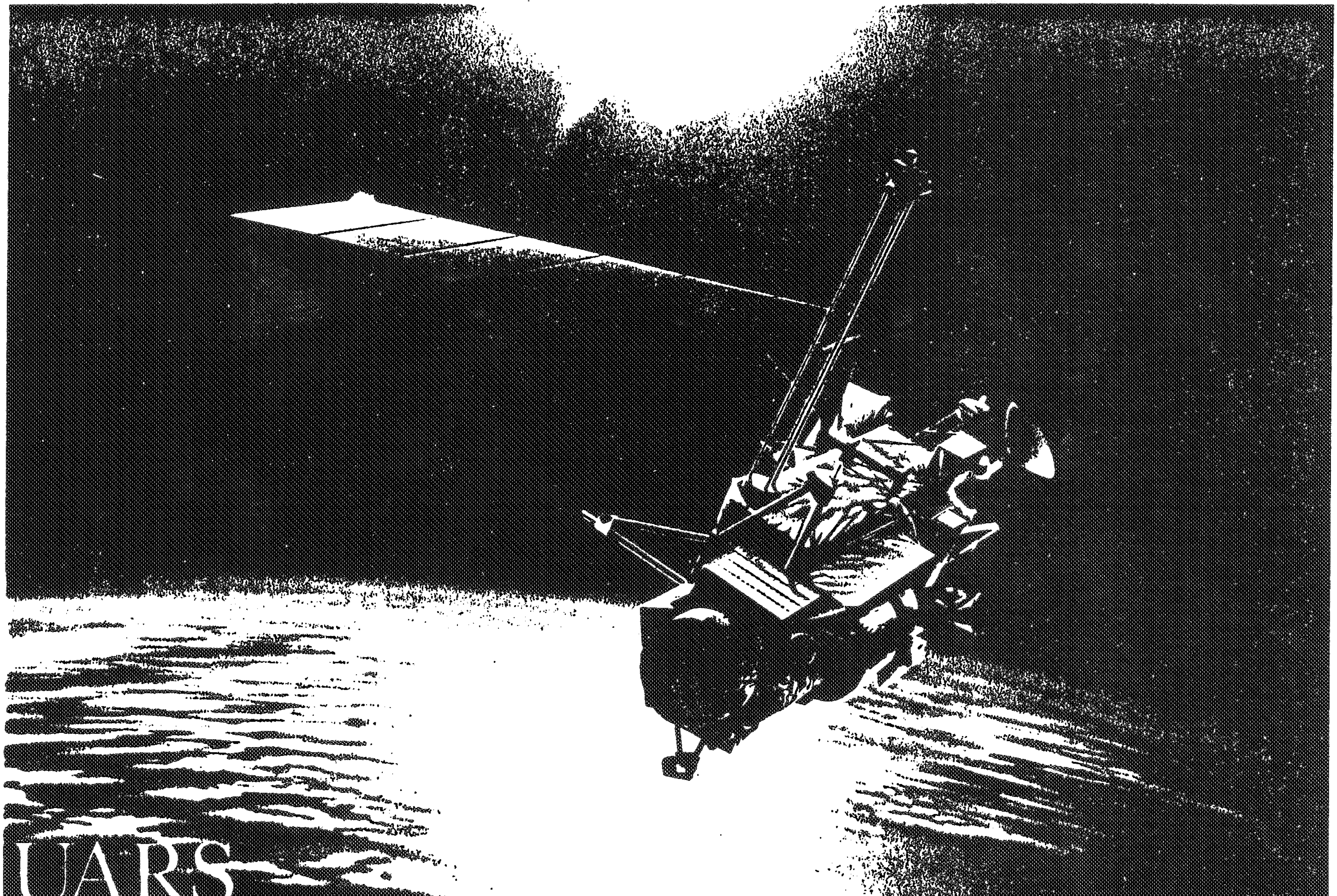
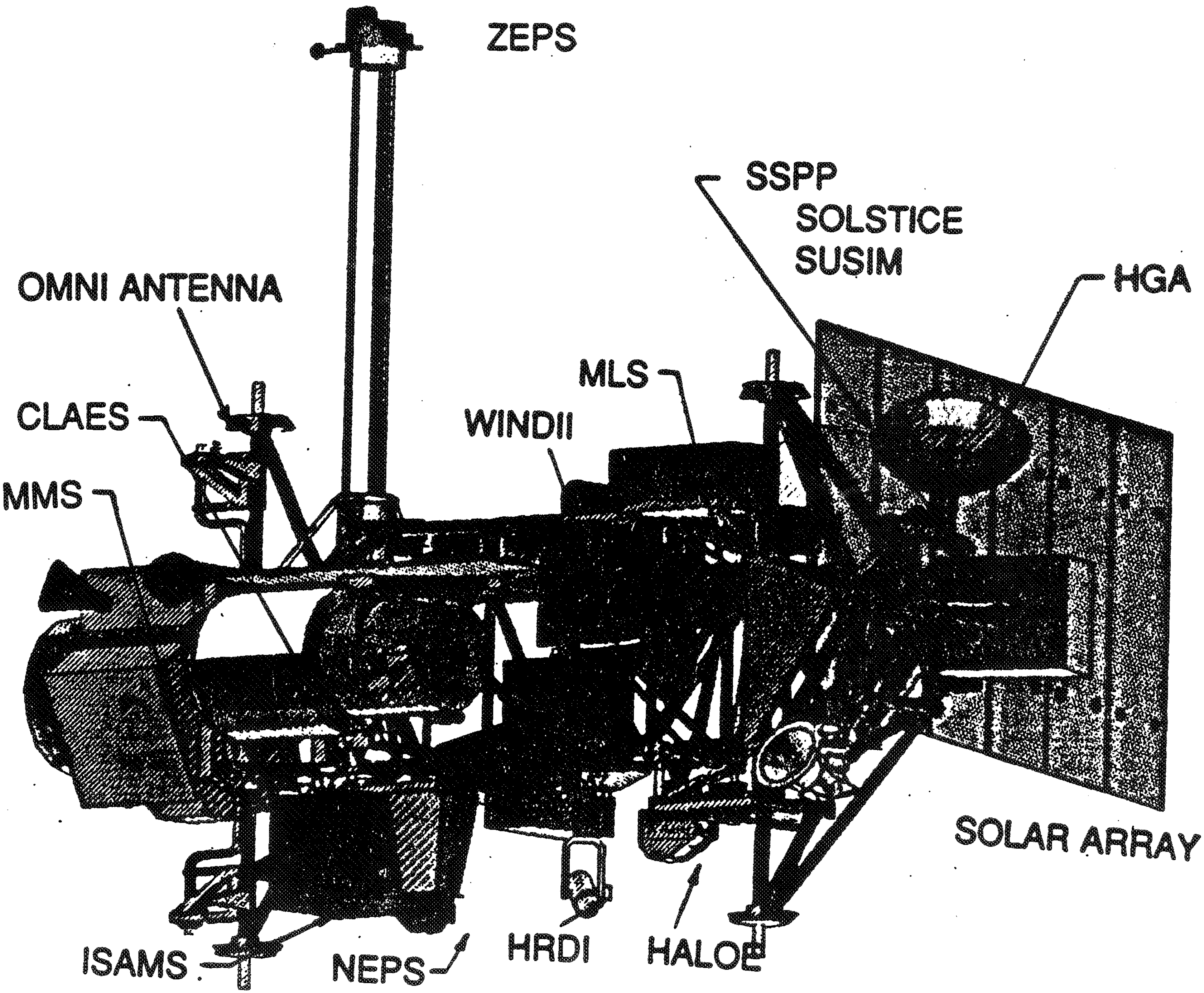
OCTOBER 5, 1992


UARS
A Global Probe
of Earth's Upper
Atmosphere

UARS JITTER STUDY OBJECTIVES

- ANALYSIS OF IN-FLIGHT JITTER
- EVALUATE DIFFERENT MODELS OF UARS
- DETERMINE JITTER PREDICTION ACCURACY SUCH THAT ADEQUATE (BUT NOT EXCESSIVE) DESIGN MARGINS WILL ASSUME FUTURE MISSION SUCCESS


ZEPS
SSPP
SOLSTICE
SUSIM
HGA
OMNI ANTENNA
MLS
CLAES
WINDII
MMS
SOLAR ARRAY
ISAMS
NEPS
HRDI
HALOE

UARS CHRONOLOGY

SEPT. 12, 1991	LAUNCH
MAY 1, 1992	DISTURBANCE EXPERIMENT
JUNE 1, 1992	SOLAR ARRAY DRIVE ANOMALY
JUNE 3, 1992	SOLAR ARRAY PARKED
JULY 13, 1992	SOLAR ARRAY NORMAL OPS

DATA CASES :

- DISTURBANCE EXPERIMENT
- YAW MANEUVER, ORBIT ADJUST
- SOLAR ARRAY ANOMALY
- HALOE SUN SENSOR
- NO INSTRUMENT DISTURBANCES
- NUMEROUS ORBITS OF NORMAL OPERATIONS
- SKEW REACTION POWERED DOWN
- MLS, SSPP, HGA, REACTION WHEELS
- THERMAL SNAP

UARS DISTURBANCE EXPERIMENT OBJECTIVES

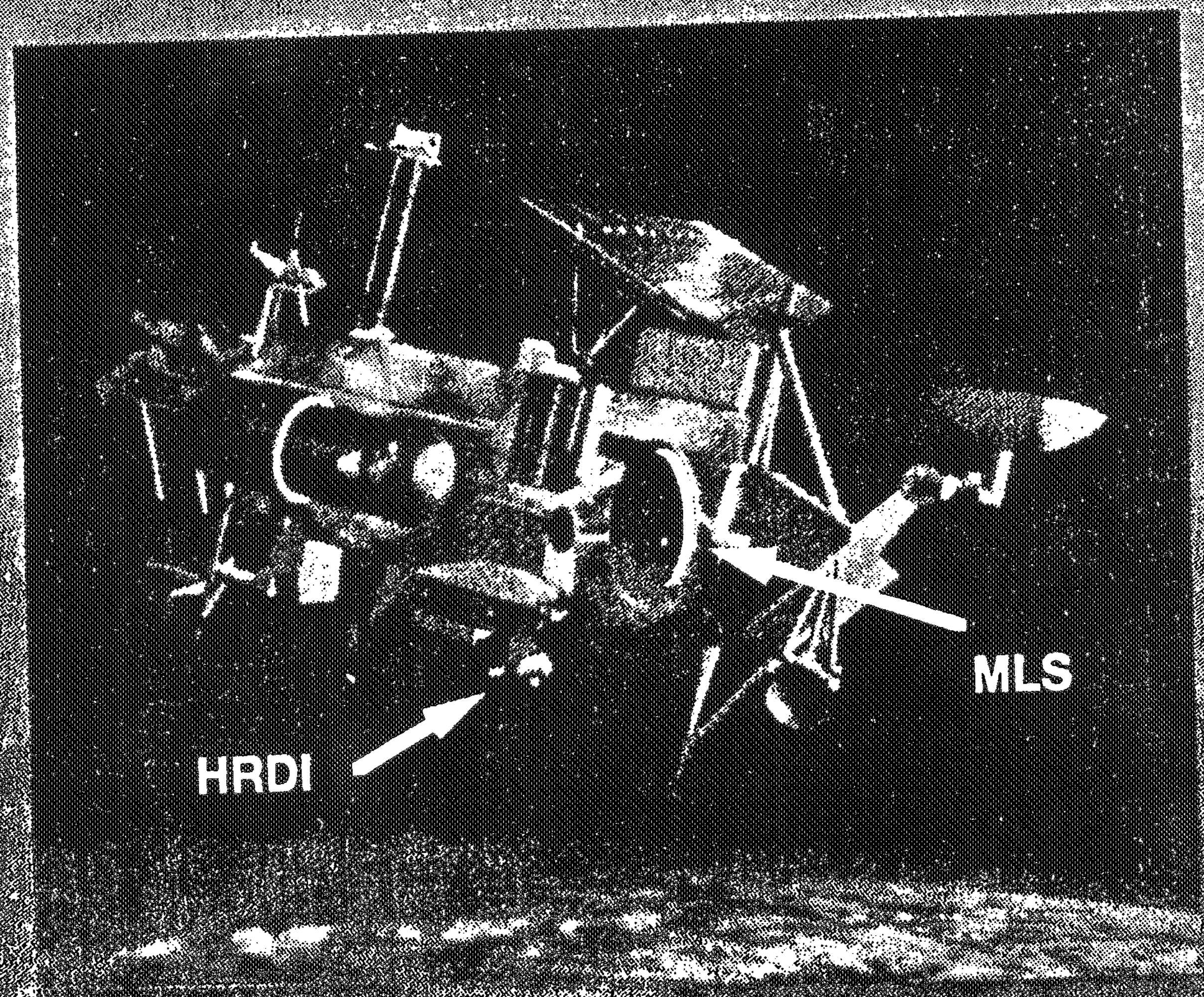
UARS Disturbance Experiment on May 1, 1992

- Pointing jitter due to each individual instrument
- Pointing jitter due to concurrent disturbances
- "No disturbance case"
- System I. D. responses
- Participants : LaRC, GSFC, General Electric

  JPL Microwave Limb Sounder Team

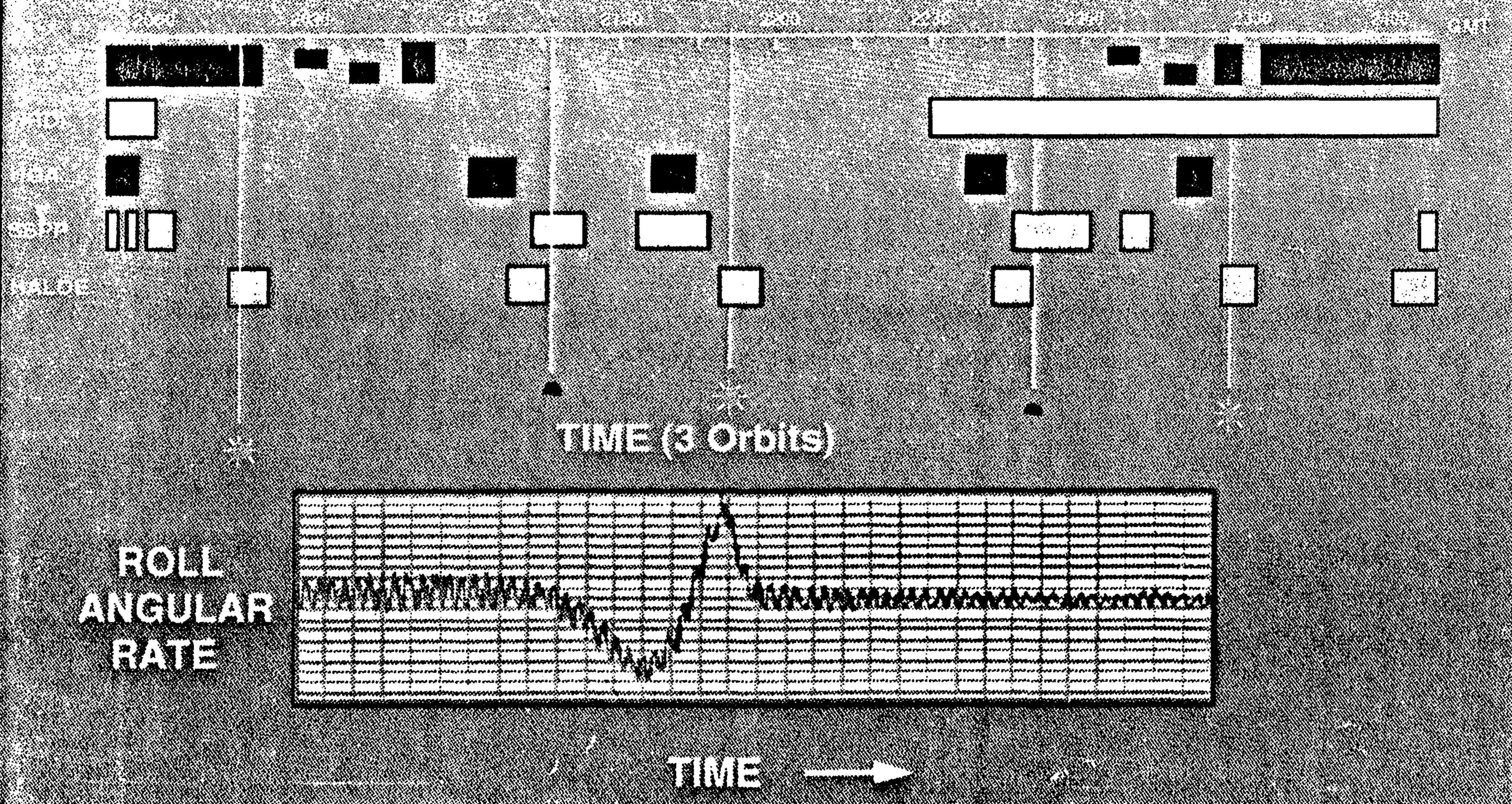
  University of Michigan - High Resolution Doppler Imager

UARS DISTURBANCE EXPERIMENT

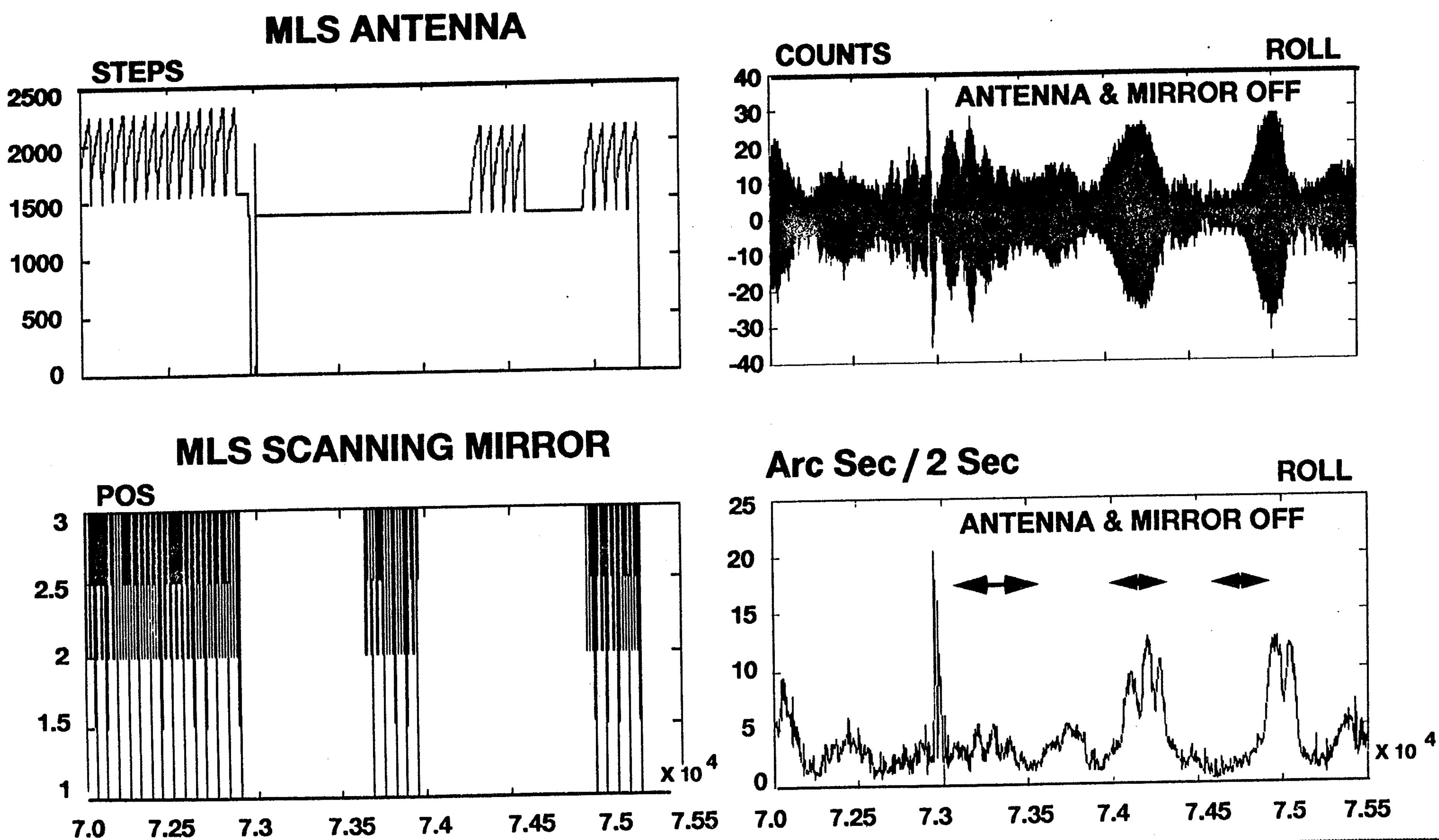


UARS DISTURBANCE EXPERIMENT

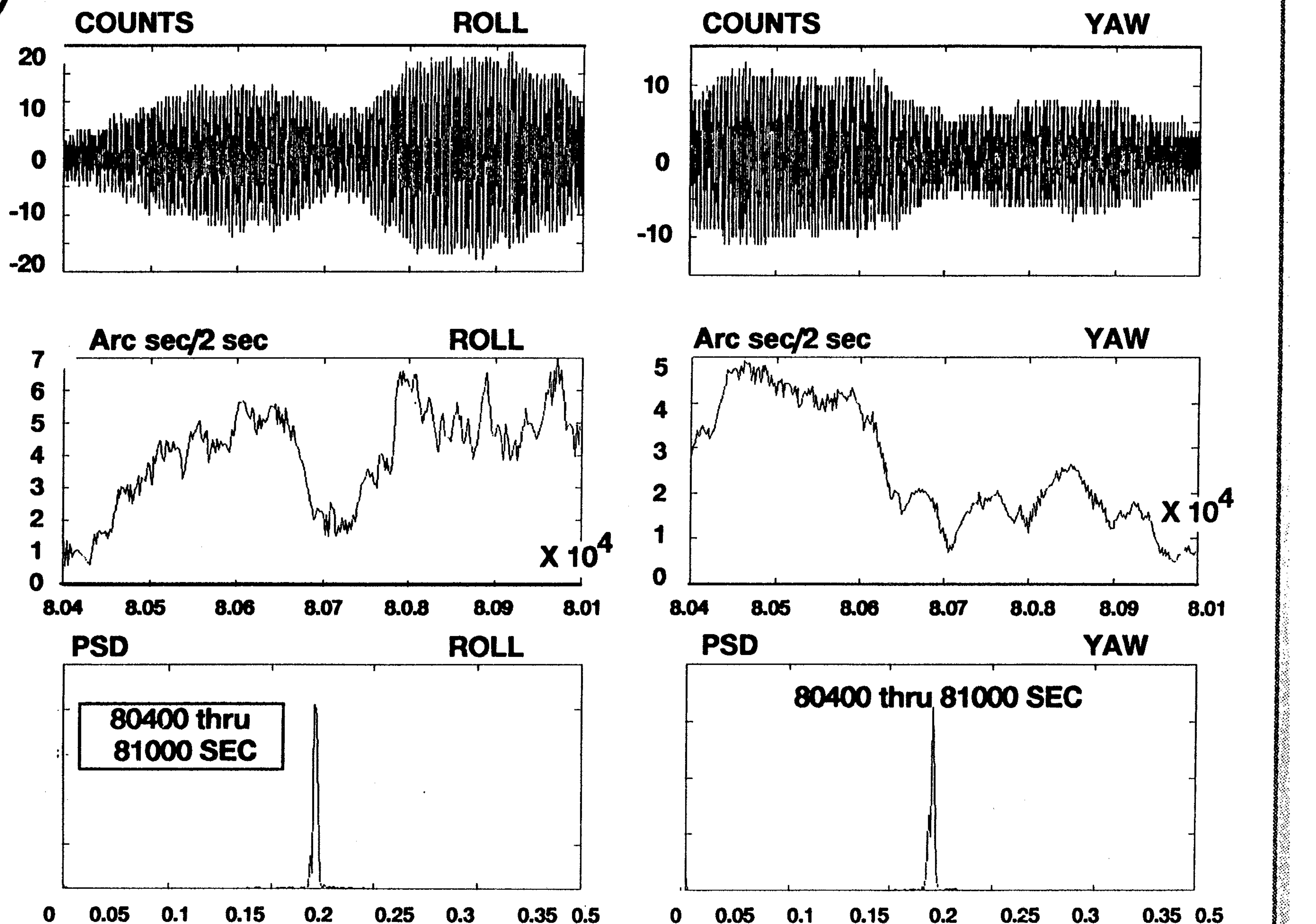
MAY 1, 1992



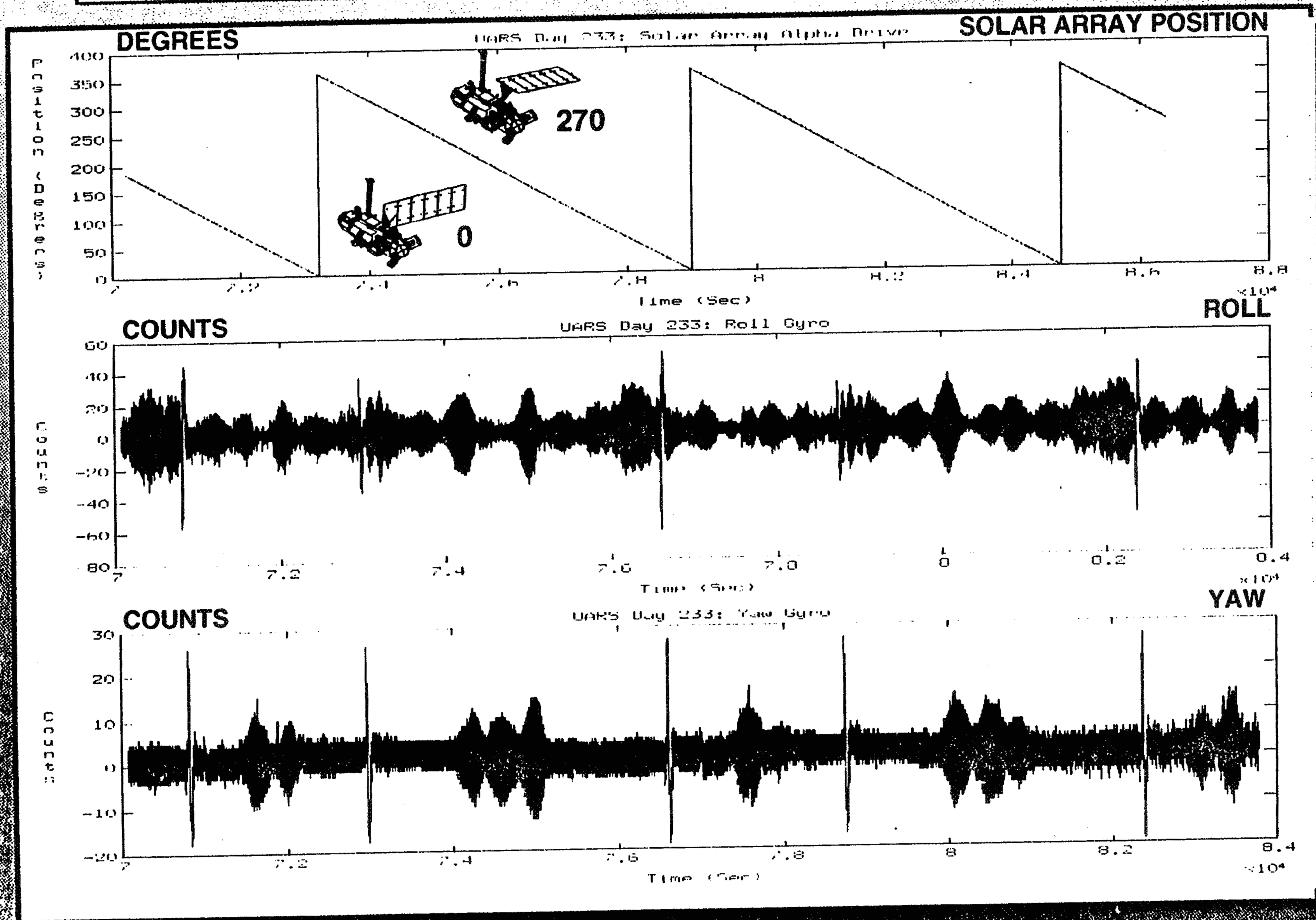
MICROWAVE LIMB SOUNDER
ANTENNA & MIRROR TIME PROFILES



SOLAR ARRAY DRIVE DISTURBANCE

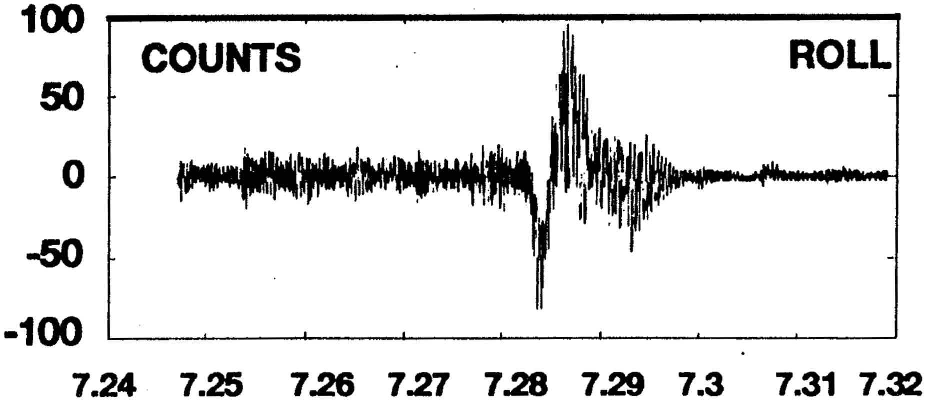


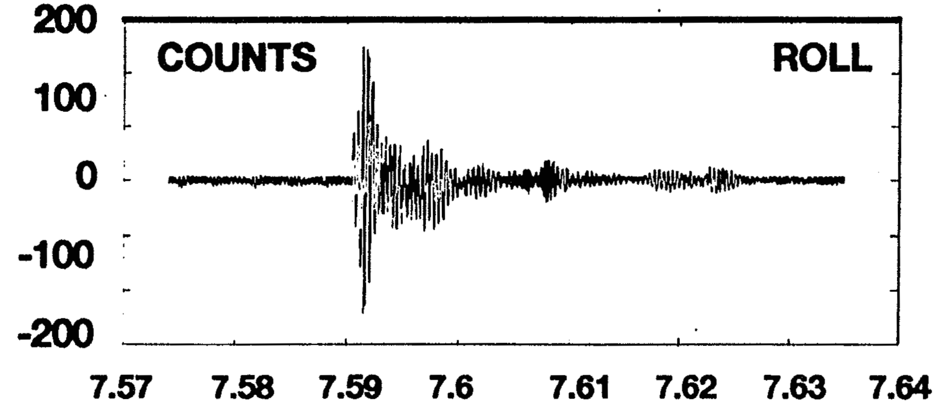
ROLL & YAW JITTER CORRELATION WITH SOLAR ARRAY POSITION



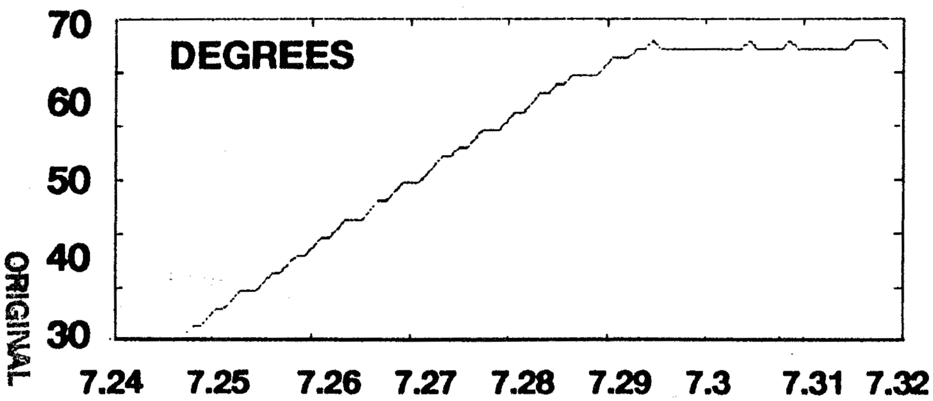
SOLAR ARRAY ROTATION

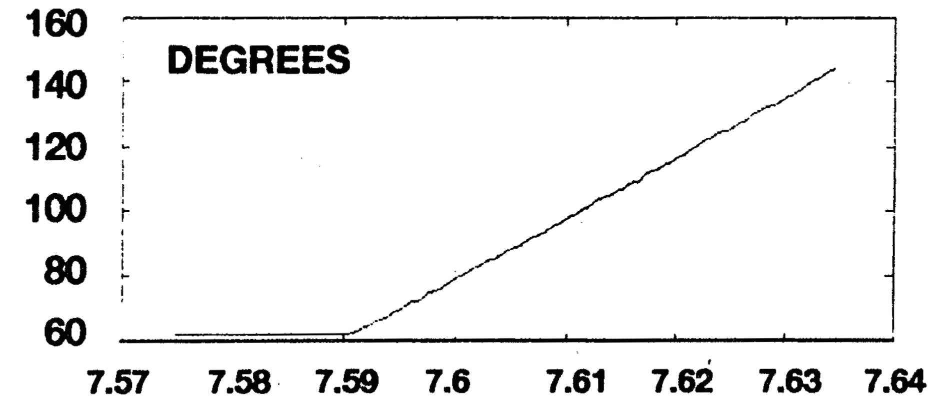
JUNE 2, 1992





SOLAR ARRAY POSITION





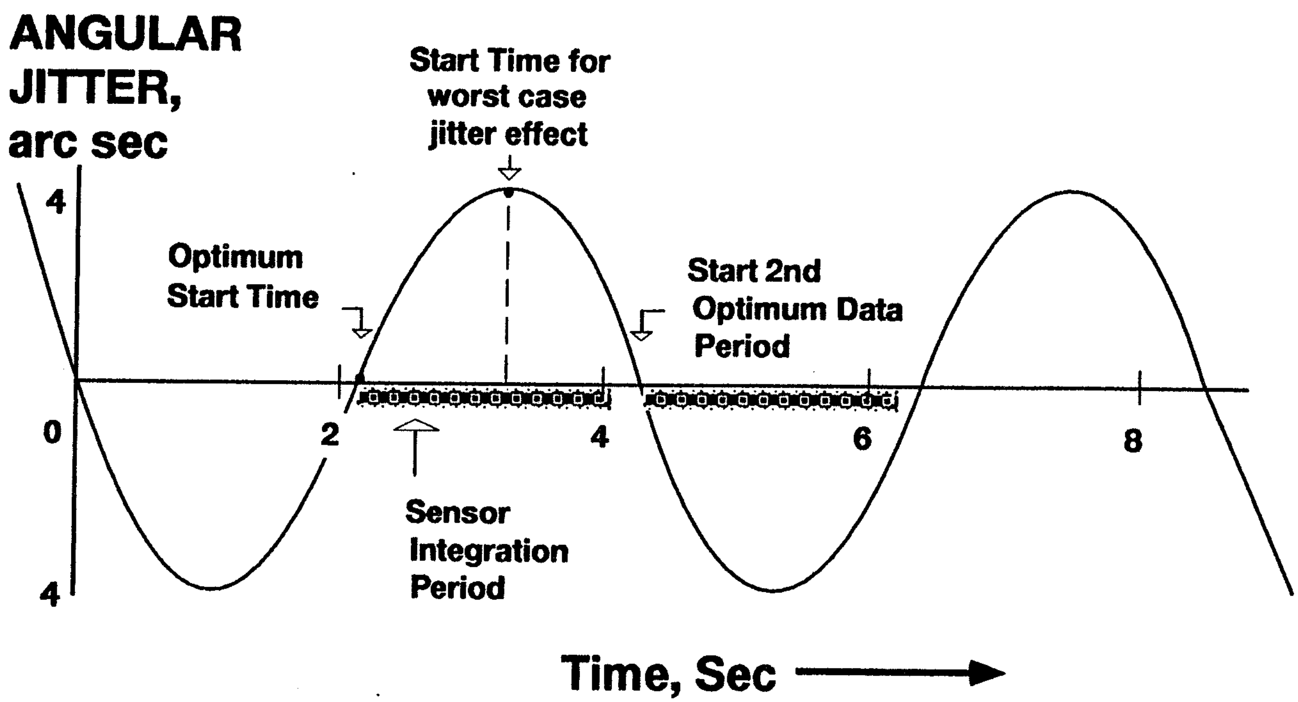
Solar Array is Dominant Disturbance Source

Damping 2.8%

UARS DISTURBANCE SUMMARY

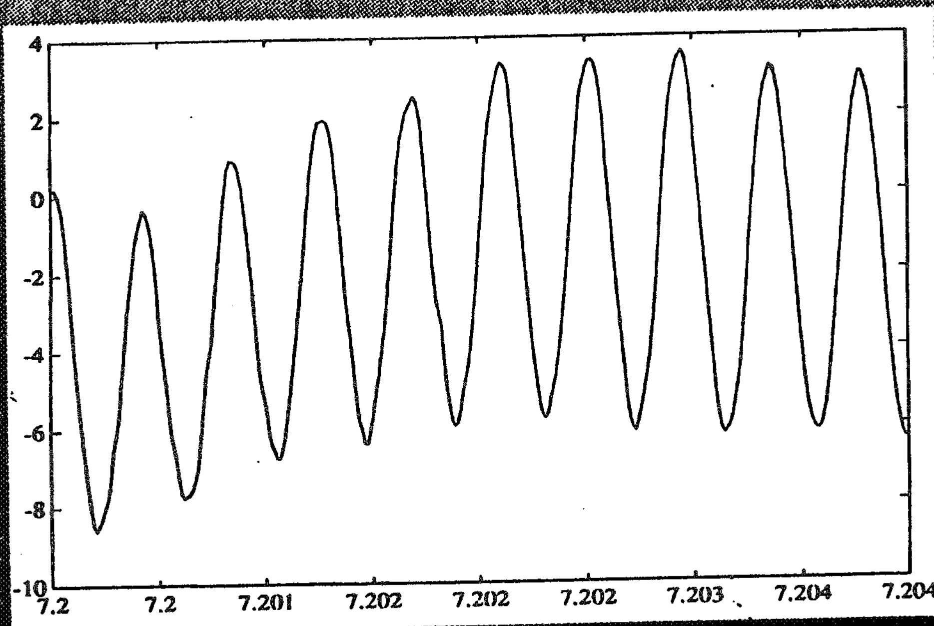
SOURCE	RIGID-BODY PLATFORM MOTION			JITTER (ARC SEC/2 SEC)			FLEXIBLE MODES EXCITED (HZ)		
	ROLL	PITCH	YAW	ROLL	PITCH	YAW	ROLL	PITCH	YAW
MLS	YES	NO	NO	2.25	0.50	0.50 0.10	0.256 0.988 1.005	0.2622 0.2837 2.2110	2.9329 0.2407
HALOE **	YES	YES	YES	2.50	1.80	1.40	0.951	0.8070	0.9510
HRDI DAY ** SCAN	YES	YES	YES			1.60	0.2478 0.2404	0.2709	2.9330 0.2365
HRDI NIGHT ** SCAN	YES	YES	YES			1.90	0.945 0.2365	0.9664 0.2450	0.9492 2.9286
SOLAR ARRAY	NO	NO	NO	17.0, VARIES WITH POSITION/DIRECTION			0.240	0.2686	0.2422
WHEELS WITH CONSTANT RPM	NO	NO	NO	0.50	0.50	0.50	2.942 0.245	0.2454 1.9627	2.9420 0.2454

CONCEPT for REDUCTION of JITTER EFFECT

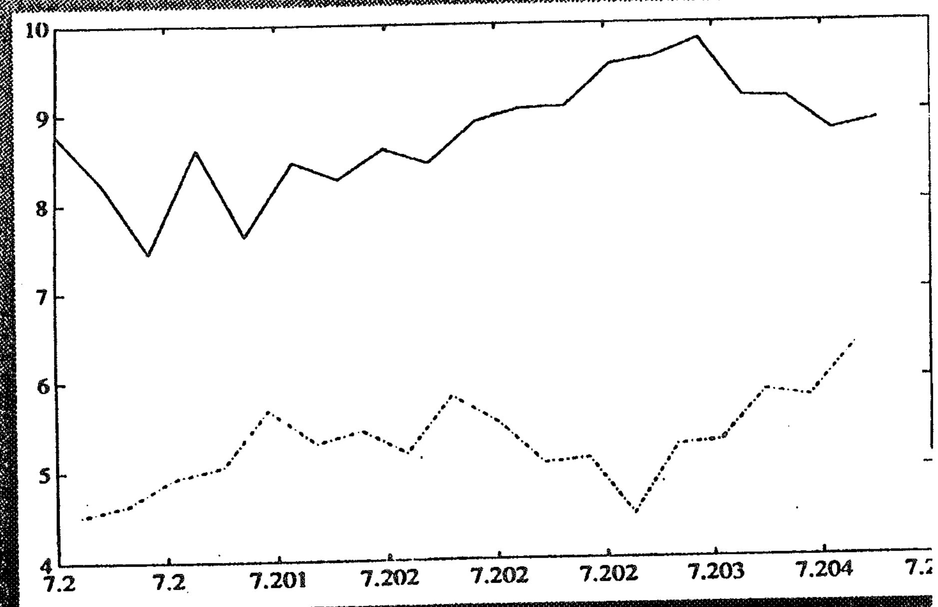


JITTER REDUCTION USING WINDOW SCHEDULING

ROLL DISPLACEMENT



ROLL JITTER



MAY 1, 1992 - UARS DAY 0233

HIGHLIGHTS / LESSONS LEARNED

- NEED FOR JITTER STUDY AND ACCELEROMETER
- INSTRUMENT / SUBSYSTEM DISTURBANCE ANALYSIS
- SOLAR ARRAY DRIVE
  - Major jitter source correlated with ground track
- UARS DISTURBANCE EXPERIMENT
  - May 1 experiment data used solar anomaly analysis

JITTER REDUCTION METHOD FOR WIND II

DAMPING

RESULTS APPLIED TO EOS :
- SOLAR DRIVE DYNAMICS
- REACTION WHEEL DYNAMICS

omit

SESSION 5:

PROPULSION

Co-Chaired by:
Mr. James Moses, NASA Marshall Space Flight Center
Mr. Lee G. Meyer, USAF Phillips Laboratory

ELITE Program

Electric Insertion Transfer Experiment

Robert Vondra

Phillips Lab/UDRI

NASA/DOD Flight Experiments

Technical Interchange Meeting

Monterey, CA

5-9 Oct 1992

159213

P. 12

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Contents

- Concept
- CRDA (Cooperative Research & Development Agreement)
- Spacecraft
- Benefits
  - EOTV
  - Orbit Repositioning
  - Comet, Asteroid Rendezvous
- Program Milestones
- Subsystem Status
- Partners
- Payloads
- Summary

PHILLIPS LABORATORY

CRDA

- 1986 Federal Technology Transfer Act- make US industry more competitive

- ELITE CRDA signed by AF & TRW 1991, & ratified by Gen. Rankine, AFSC/XT Jan 92

- AF Provides major subsystems
  - Arcjet, photovoltaic arrays, diagnostics
  - Launch vehicle, ground segment, s/c & lv integration

- TRW provides spacecraft
  - Spacecraft, systems engineering. & flight support
  - Flight software, launch & missions operations support

Concept

O OBJECTIVE

- System level demonstration of Electric Orbit Transfer Vehicle (EOTV) adequate to establish LEO to GEO transfer capability, orbit maneuvers

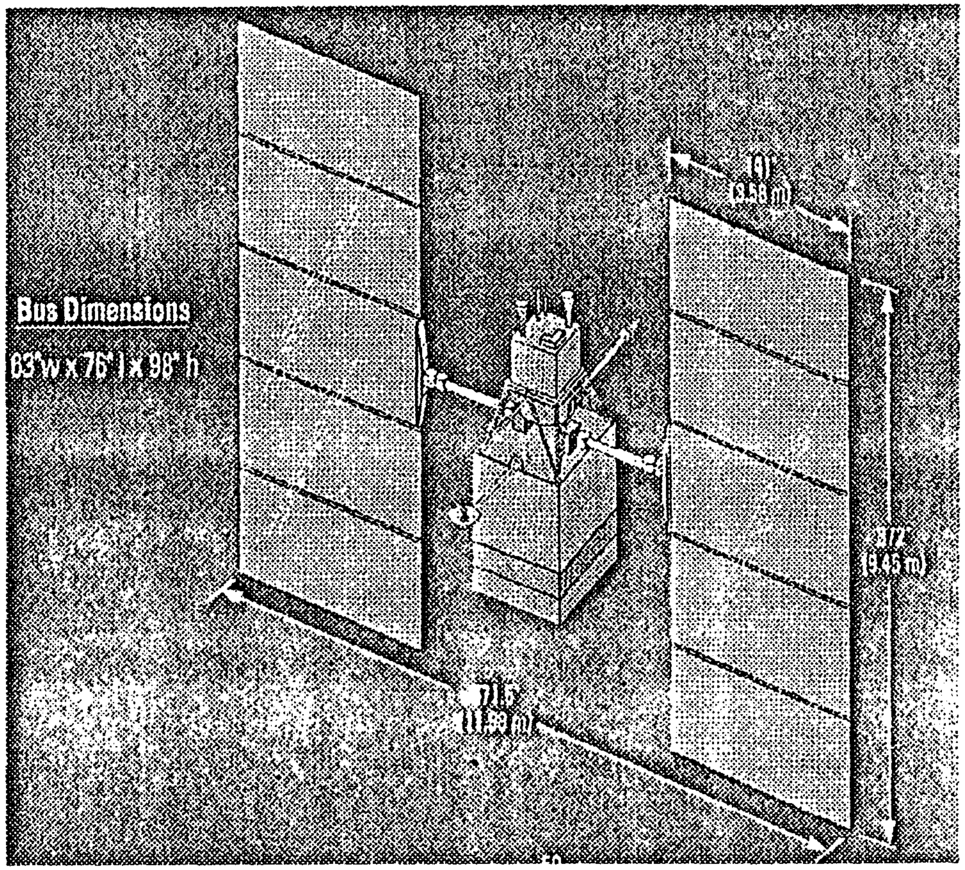
O DESCRIPTION

- Demonstrate EOTV system
    - o Autonomous orbit transfer
    - o Orbit repositioning
    - o Survivability in Van Allen belts

- Demonstrate critical subsystems
    - o Arcjet
    - o Solar array
    - o Autonomous GN&C

Electric Insertion Transfer Experiment

- O Electric orbit transfer vehicle (EOTV) and reposition demonstration
  - Validate LEO-GEO EOTV transfer
  - Demonstrate rapid orbit reposition
  - High power arcjet and photovoltaic arrays
  - Autonomous GN&C



Deployed ELITE Spacecraft

- O Experiment
  - 10kW NH3 arcjet, photovoltaic array
  - 200 to 2150 nmi alt.
  - 63.5 to 60.5 deg inclination
  - 45 day transfer
  - Total radiation flux = LEO-GEO transfer
  - 100 deg reposition at 2150 nmi, 6 days

- O Operational System
  - 30kW H2 arcjet, photovoltaic array
  - 200 to 20,000 nmi alt.
  - 28.5 to 0 deg inclination
  - 6 month transfer
  - Total radiation flux = LEO-GEO transfer
  - Orbit reposition at GEO

EOTV
New Space Business

- Faster or more on-orbit maneuvers
- Increase MLV capability to HLV range
- Expand HLV capability to NLS domain
- Fast crisis response GEO satellites
  - Launch-On-Schedule & on-orbit spares
- 2x the payload & 1/2 trip time of ballistic asteroid, comet missions
- High power platform for space tests

Program Milestones

- Program start, *Jan 91*
- Go/No Go, Dec 91
- Systems Requirements Review, Sep 92
- Go/No Go, Dec 92
- Preliminary Design Review, Jul 93
- Critical Design Review, Jun 94
- Flight Qualification Review, Mar 96
- Launch, *Sep 96*
- Flight Op.s end, Jun 97

Subsystem Status

- Ammonia arcjet
  - RFP release end Nov 92
  - 1460 continuous hours demonstrated, 50% x needed
  - 707 on/off cycles demonstrated, 30% x needed
  - At 10 kW, Isp > 620 s, efficiency > 33%
- Photovoltaic array
  - RFP release end Nov 92
  - BOL= 10.2 kW, EOL>= 6.8 kW
  - Sp power >= 40W/kg
- Diagnostics
  - Prioritized
  - Health & status, engineering & scientific data

Subsystem Status
Cont.'d

- High Power Testbed
  - Objective: Simulate ELITE power distribution system with arcjet load
  - Approach: Solar array simulator, peak pwr tracking, pwr distribution system, arcjet
  - Rationale: Eliminate risks of end-to-end system
  - Testing: Nov 92 at PL/Edwards AFB
- Spacecraft Bus (TRW UTB)
  - Modular, improved performance, reduced weight, reduced cost
  - ELITE adaptations include structure, thermal control, avionics

Partners

- Other government & industry partners likely

Organization	Bus	Propul	P/V	Diag	GN&C	Experiment	Cost
TRW	X				X		
Air Force		X	X	X			
NASA Code R			X			Xe Ion, hydrogen arcjet	
NASA Cose S							X
AFSPACECOM							X
General Dynamics			X			Cryo H2 storage	

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6Oct92

Payloads

- Global Surv Sat Tech Demo, Eagle Dancer (PL)
  - New, 5-10kW, maneuvering?, #1 AFSPACECOM
- High Temperature Superconductivity Space Experiment (NRL)
  - Needs bus, operate in Van Allen belts, #2 Tri Service SERB
- High power ELITE spacecraft bus (JPL)
  - Candidate for Discovery program
  - The body for EP planetary spacecraft
  - 2x payload mass in 1/2 time to comets, asteroids
- High power ion engine experiment (JPL/LeRC)
  - Qualify engine for VESTA/CLIPPER
- Radiation Hardened S/C Microelectronics (NRL & PL)
  - Van Allen belt exposure? #6 AFSPACECOM
- Space Surveillance Initiative (PL)
  - New, maneuvering need? #9 AFSPACECOM
- Hydrogen arcjet and cryo storage (Gen Dynamics, LeRC)
  - Experiment or primary propulsion

Summary

- ELITE system demo traceable to operational solar electric vehicle
- CRDA assures transfer of advanced technology to industry
- Orbit transfer, maneuvering, planetary vehicles greatly reduce the cost of future missions
- ELITE & CRDA offers opportunity benefiting DOD, NASA and industry interests

LIQUID MOTION IN A ROTATING TANK EXPERIMENT (LME)

Southwest Research Institute

Program Manager: D. M. Deffenbaugh

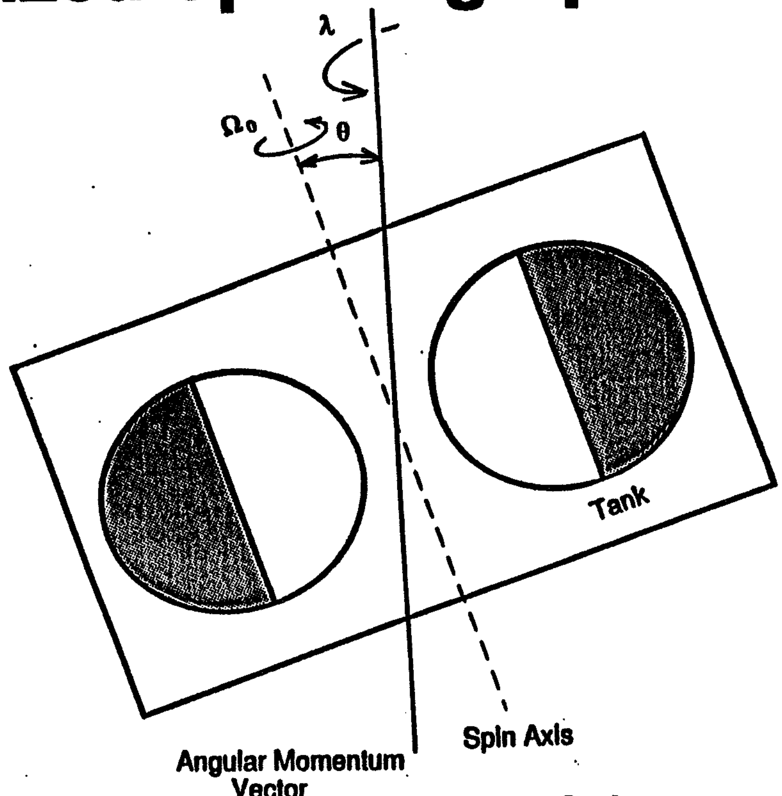
Principal Investigator: F. T. Dodge

NASA Technical Monitor: J. Didion (NASA-GSFC)

Problem Description

- Many spacecraft spin to obtain gyroscopic stability or to moderate solar heating effects
- A spinning spacecraft has a natural "wobbling" or nutational motion
- The nutation produces an oscillatory motion of the liquid in the spacecraft tanks
- The dynamic effects of the liquid motion can detrimentally affect the stability and control of the spacecraft
- These problems are aggravated by the lack of both knowledge and analytical models about liquid motions in spinning tanks

Generalized Spinning Spacecraft



Nutation frequency of the spacecraft is:

$$\lambda = \Omega_o \left| 1 - \left(I_{spin} / I_{transverse} \right) \right|$$

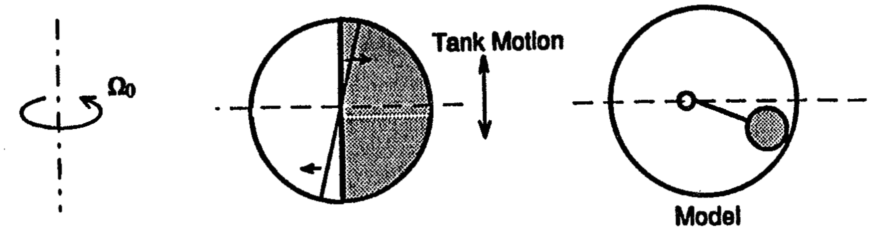
LME Objective

- There is a demonstrated need for improved understanding of the dynamics of liquids in the tanks of spinning spacecraft.
- Objectives of LME are:
    - acquire representative data to improve ground-test scaling procedures for spacecraft design
    - obtain scientific understanding to formulate and validate better analytical models of liquid motions in spinning tanks

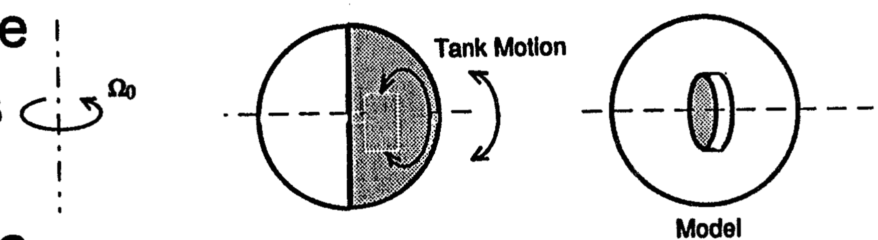
Liquid Oscillations in Spinning Tanks

Spinning-tank liquid motions are of two kinds:

- Free surface waves similar to liquid sloshing
  - need a free surface
  - mechanical analog is a pendulum
  - natural frequency is more than 2Ω



- Inertial (or internal) waves
  - no counterpart in a non-spinning tank
  - does not a free surface
  - natural frequency less than 2Ω
  - mechanical analog is a rotor

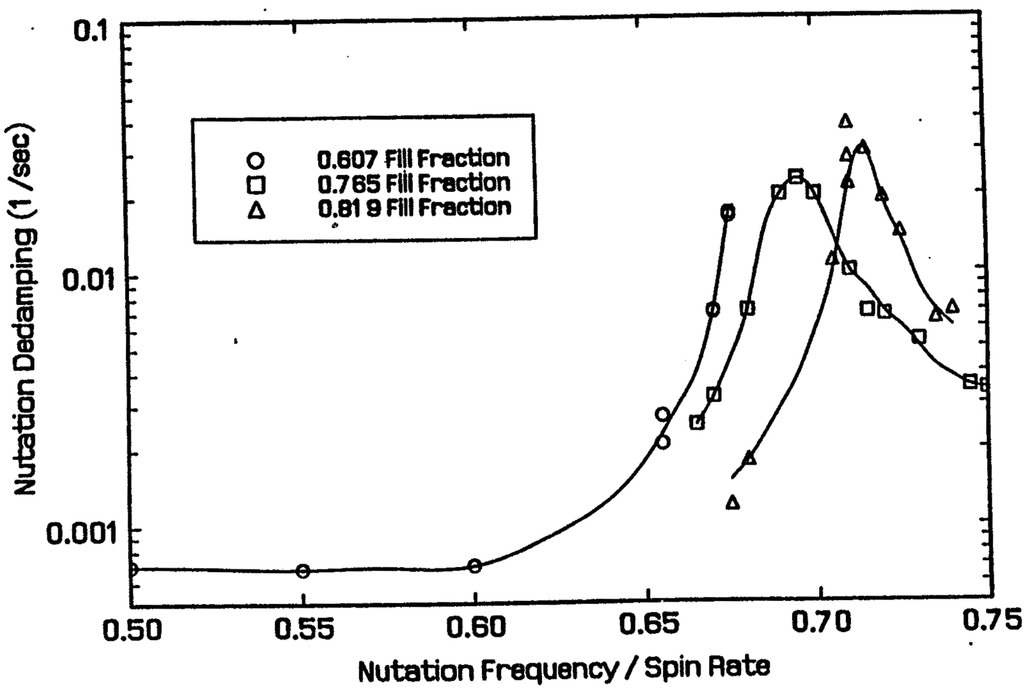


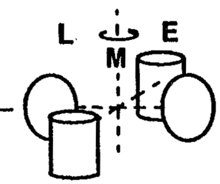
Importance of Problem

- The liquid inventory of spacecraft is a significant fraction of the total mass (over 50% for a spacecraft with a liquid apogee engine)
- Transient and oscillatory motions of such large amounts of liquid easily influence the spacecraft attitude control system when:
  - the liquid natural frequency is near the nutation frequency or control frequency of the spacecraft, or
  - the spacecraft is susceptible to a "flat" spin instability

Importance of Problem (cont'd)

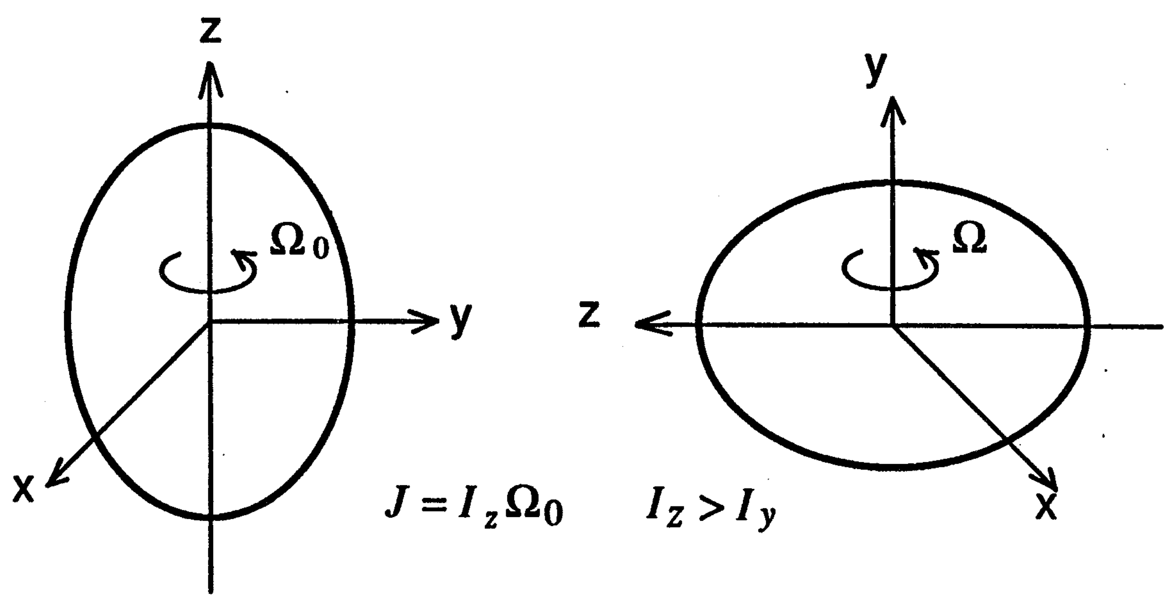
The graph shows liquid resonance effects obtained during in-space experimentation of INTELSAT IV. The resonance frequencies are near the nutation frequency. Ground testing could not and did not reveal the presence of these resonances.





Importance of Problem (cont'd)

- When resonances do not occur, energy dissipated by viscous liquid motions can cause serious problems
  - spacecraft kinetic energy is $J^2/2I_s$



- Since $I_z < I_y$, the spacecraft will eventually spin about the y axis - unless nutation is controlled

Limitations of Ground Testing

- Two general kinds of ground testing are in use
  - spin table (free-floating air bearing)
  - drop test
- Spin table tests must use unrealistically high spin rates to overcome gravitational acceleration
  - energy dissipation rates much greater than flight
  - inertial resonances are over-damped
  - any effect of surface tension cannot be studied
  - results must be scaled to flight conditions
    - correlations cannot be validated
- Drop tests must use small models
  - short test times (less than two seconds)

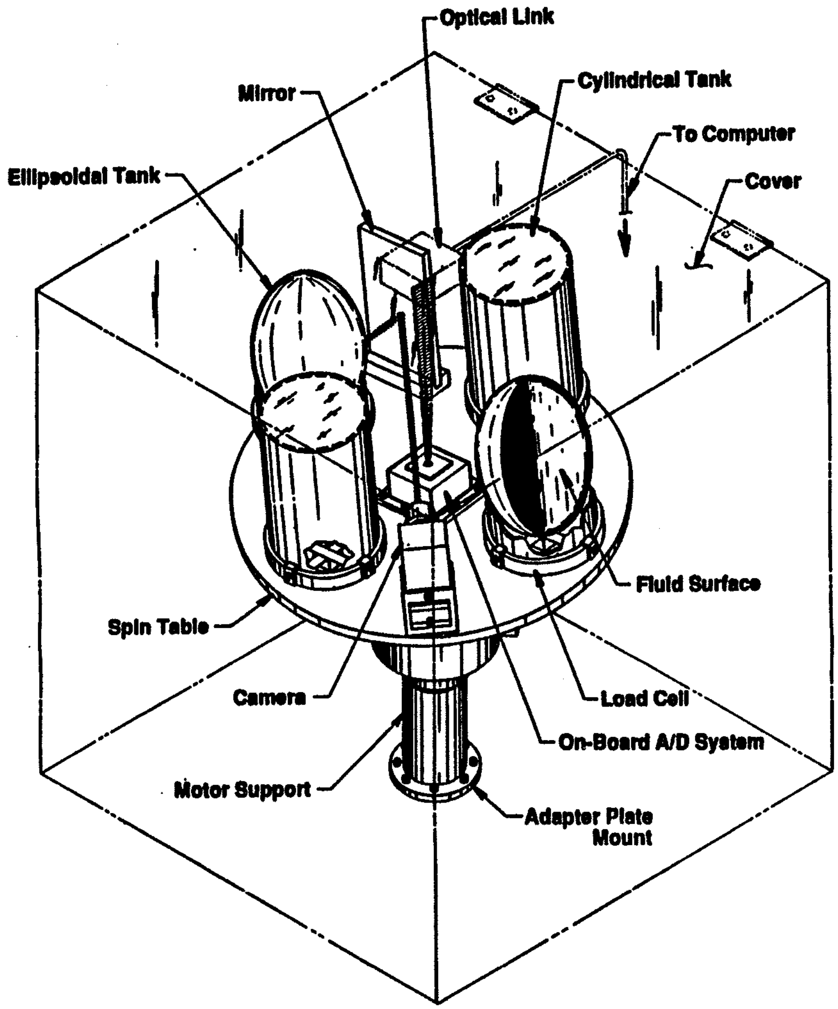
Need for LME Space Experiment

Space tests overcome **all** ground-test limitations

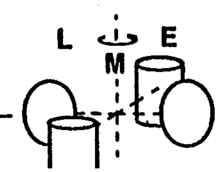
- eliminate unwanted and spurious effects of gravity
  - give correct geometric configuration
  - eliminate once-per-nutation cycle free surface excitation
- allow low spin rates to be used
  - reduce energy dissipation rate to flight-like values
  - eliminates over-damping of inertial oscillations
  - can investigate importance of surface tension
  - makes visual observations possible
- permits sufficiently long test times

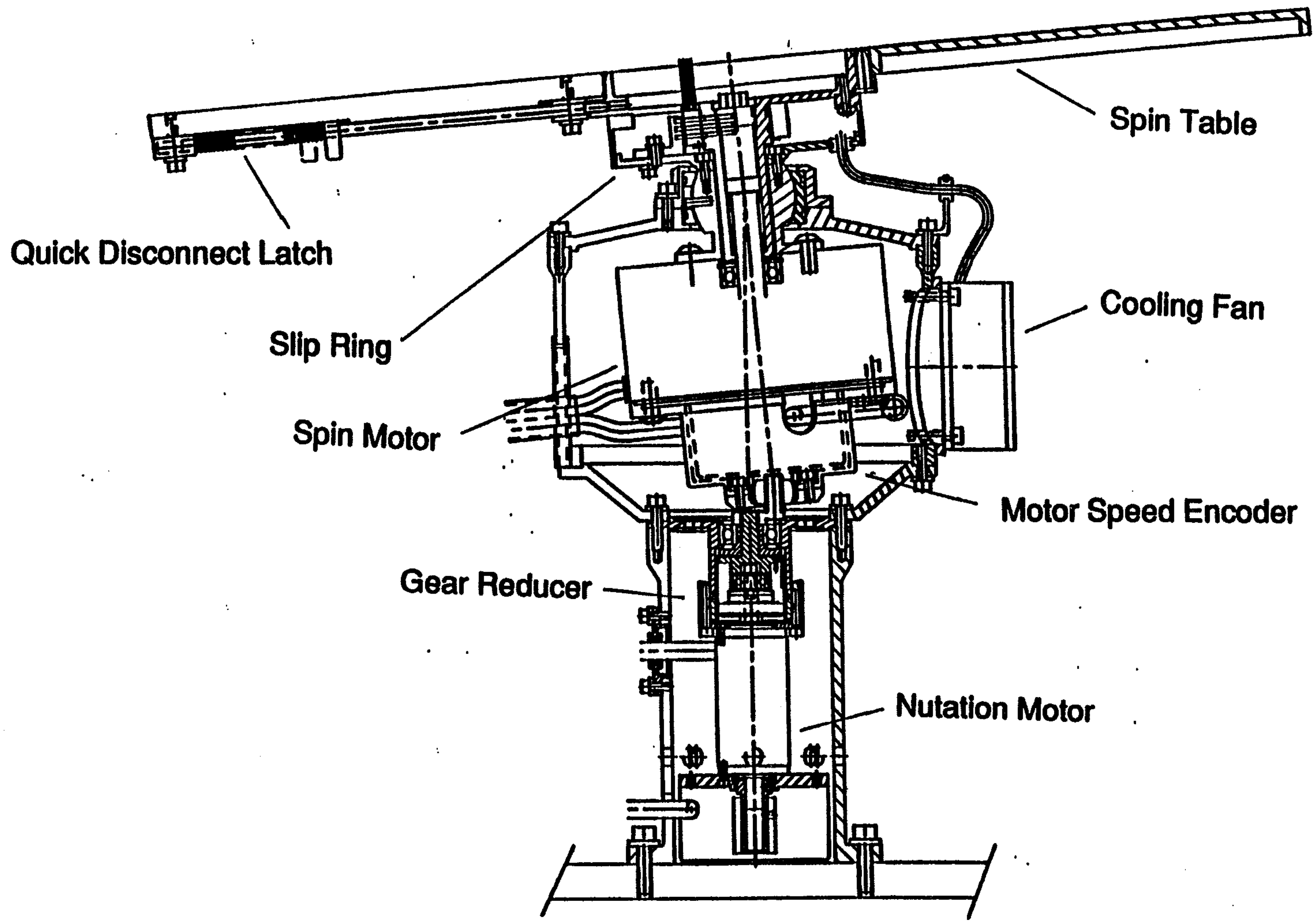
LME Description

- Basic experiment design is a spin table that will subject a set of 4 model tanks containing liquid to realistic nutation motion
  - hardware is mounted on a double adapter plate
  - experiment computer is located in an adjacent locker
- Nutation frequency is variable for any spin rate
  - spin table driven by two independently-controllable brushless DC motors
- Tank shape, liquid fill level, and liquid properties are the primary test variables
  - 2 sets of tanks are required (changed in flight)
  - all 4 tanks of a set are tested simultaneously



Isometric of LME Hardware Configuration

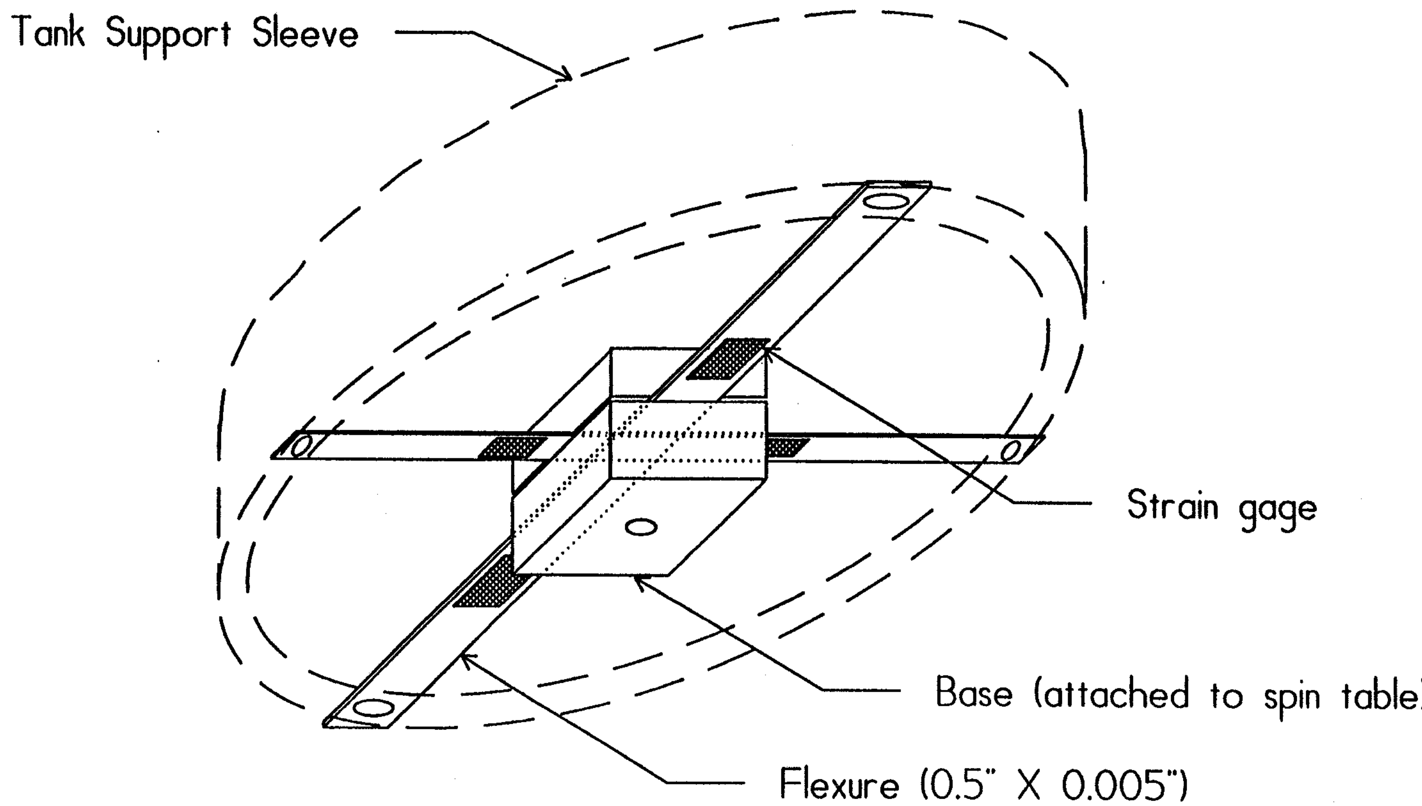




Schematic of LME Spin Table and Drive Motors

Load Cells

- Oscillatory torque magnitudes to be measured range from 1.7×10^{-4} in-lb to 0.025 in-lb

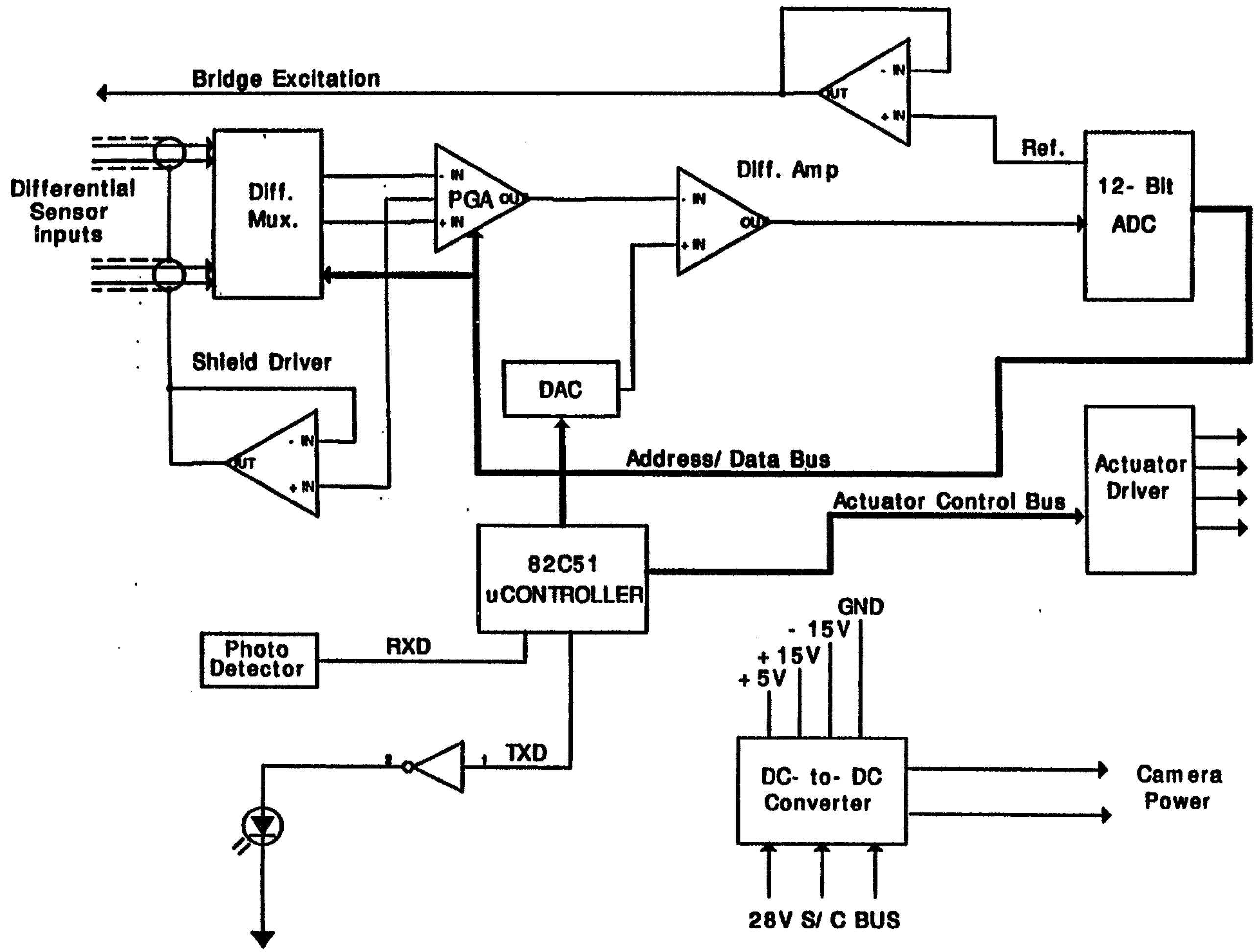


Schematic of Load Cell

Experiment Control and Data Acquisition

LME electronic subsystem consists of:

- 2 motor speed controllers
- Data acqusition system
    - printed wiring board mounted on spin table
    - signals transmitted optically to expmnt. computer
- Dedicated experiment processor (SC-4)
    - located in 3rd locker
- Data mass storage device (WORM optical disk)
- Operator interface computer
    - GRID computer supplied by NASA

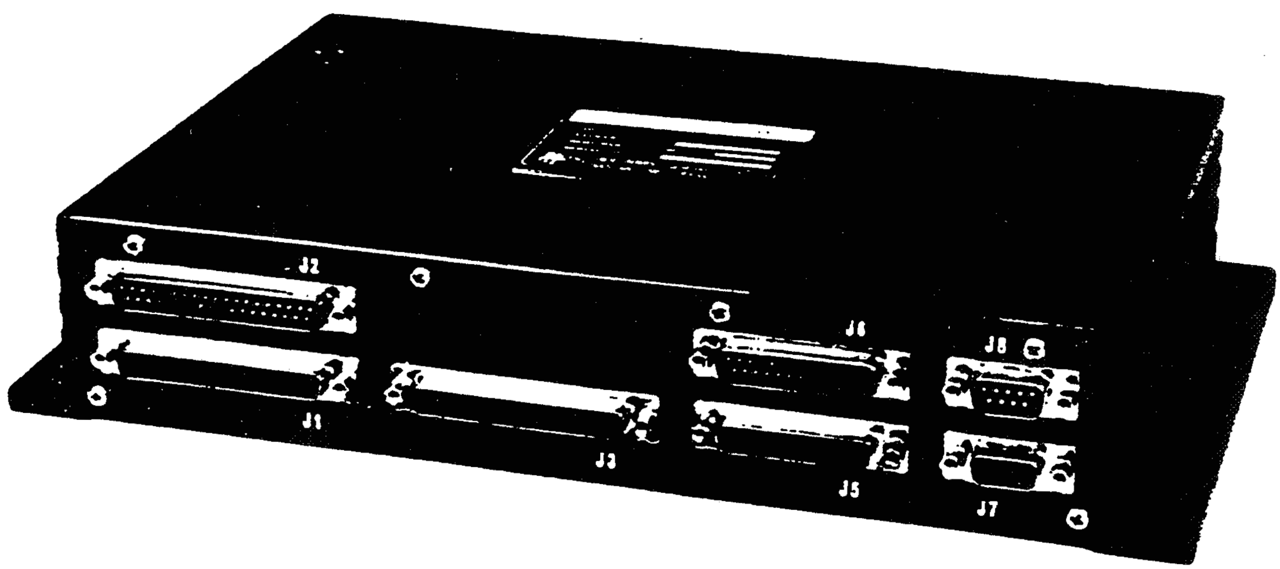


LME Spin Table Electronics

L M E

SPECIFICATION
FOR THE
SC-4 SINGLE-BOARD SPACECRAFT COMPUTER

Central Processor	80C186/80C187 16 Bit
Clock Frequency	10 MHz
Operating System	MS-DOS and VRTX Compatible
Onboard Memory	
RAM	*512K Bytes w/EDC*
EEPROM	*256K Bytes w/EDC*
UVPROM	*64K Bytes w/EDC*
Hardware Vectored Interrupts	16 User Configurable
Timer/Event Counters	8, Software Configurable
Input/Output Capability	
Parallel I/O	*16 Input, 16 Output*
Analog Input	*32 Channels, 12-bit Resolution*
Analog Output	*4 Channels, 12-bit Resolution*
RS-422 Serial I/O	*2 Channels*
SCSI Interface	*1 Port*
Software Controlled Power Switch	*4 Each*
Mass Storage	24M Bytes, Read/Write Battery-Backed
Expansion	Internal Daughterboard Connector
Size	9.25 x 12 x 2.25 in.
Weight	5 Lb. (Approximate)
Power	28V @ 5W (Approximate)



Southwest Research Institute
Instrumentation and Space Research Division
P. O. Drawer 28510
San Antonio, Texas 78228-0510
(512) 522 3477

LME Variable and Response Parameters

- Parameters to be varied in LME tests:
  - tank shape and liquid fill level
  - liquid viscosity and surface tension
  - tank motion
    - nutation frequency swept from 1 rpm to twice the spin rate for each test
- Primary measured response parameters
  - resonant conditions (natural frequencies)
  - liquid torque as function of spin rate and nutation
  - phase angle between torque and nutation
  - energy dissipation rate (computed from torque)

Proposed Test Matrix

Spin Rate, rpm	Nutation Freq., rpm	Liquid Visc., cp	Centrifugal Accel., g's	Bond No.
4	1 - 8	1	0.0027	1.2
16	1 - 32	1	0.033	15
20	1 - 40	1	0.067	30
4	1 - 8	10	0.0027	1.2
16	1 - 32	10	0.033	15
20	1 - 40	10	0.067	30

- Total test time, including calibration runs and repeats, is estimated to be 6 to 8 hours
- Mission specialist will verify test setups and initiate each test

Summary

- The "Liquid Motion in Rotating Tank Experiment" (LME) will investigate and quantify liquid motions occurring in spin-stabilized spacecraft:
  - acquire representative data to validate ground-test scaling procedures
  - obtain scientific understanding to formulate better analytical models
- LME eliminates the limitations of ground testing
- LME design is nearing the end of Phase B
  - breadboard hardware model has been completed
  - load cells have been fabricated and tested
  - experiment computer has been flight qualified
  - other electronics have been breadboarded

Vented
Tank
Resupply
Experiment

NASA Lewis
Research Center
Contract NAS3-25977

October 6, 1992

Vented Tank Resupply Experiment (VTRE) for In-Space Technology Experiments Program (IN-STEP)

Program Overview Presentation

Presented at:

NASA/DOD
Flight Experiments Technical
Interchange Meeting

Presented by:

William J. Bailey
Program Manager

MARTIN MARIETTA

N93-28711

Vented Tank Resupply Experiment

VTRE CONTRACT SUMMARY

PROGRAM DESCRIPTION	DEVELOP, DESIGN, BUILD & PROVIDE FLIGHT AND POST FLIGHT SUPPORT FOR A SHUTTLE HITCHHIKER EXPERIMENT TO INVESTIGATE AND DEMONSTRATE VENTED TANK RESUPPLY AND DIRECT TANK VENTING IN SPACE
CONTRACT	NAS3-25977
CUSTOMER	NASA LEWIS RESEARCH CENTER AL SEIGNEUR - PROJECT MANAGER
DESCRIPTION OF WORK ØB	DEVELOP THE CONCEPTUAL DESIGN FOR A SPACE SHUTTLE FLIGHT EXPERIMENT TO INVESTIGATE VENTED TANK RESUPPLY IN LOW-G. DEVELOP THE RATIONALE, OBJECTIVES & JUSTIFICATION, DEFINE THE DESIGN, ANALYSIS AND TESTING REQUIRED, AND DEVELOP A COMPLETE AND DETAILED MANAGEMENT PLAN FOR THE PHASE C/D OPTION. PRESENT AND ADVOCATE THIS PLAN BEFORE NASA MANAGEMENT.
DESCRIPTION OF WORK ØC/D	CONTINUE THE PHASE B EFFORT TO DESIGN, BUILD, TEST, AND DELIVER THE THE VTRE FLIGHT EXPERIMENT. SUPPORT THE PREPARATIONS FOR FLIGHT AND OPERATION OF THE EXPERIMENT AND REDUCE, ANALYZE, INTERPRET AND DISSEMINATE THE RESULTS OF THE EXPERIMENT.
PERIOD OF PERFORMANCE	JUNE 1991 - JULY 1992 (PHASE B) AUGUST 1992 - MAY 1996 (PHASE C/D)
PROGRAM MANAGER	WILLIAM J. BAILEY
CONTRACT VALUE	\$ 508 K (PHASE B) \$3,773 K (PHASE C/D) \$4,281 K (TOTAL PROGRAM)

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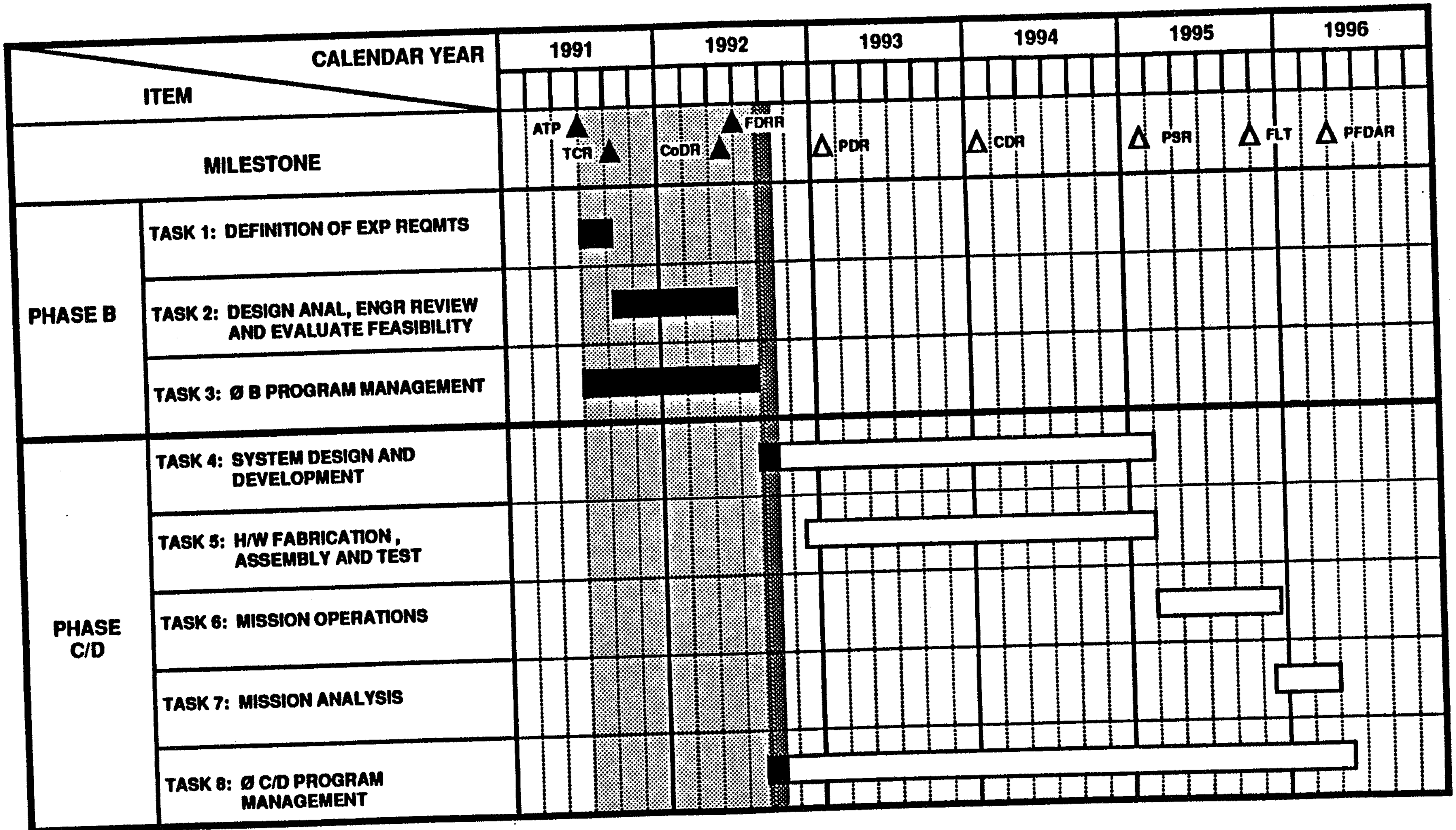
VTRE TOP-LEVEL PROGRAM SCHEDULE

Vented Tank Resupply Experiment



ATP: AUTHORITY TO PROCEED
TCR: TECHNICAL CONCEPT REVIEW
C oDR: CONCEPTUAL DESIGN REVIEW
PDR : PRELIMINARY DESIGN REVIEW
FDRR: FLIGHT DEVELOPMENT READINESS REVIEW
CDR: CRITICAL DESIGN REVIEW
PSR: PRESHIP REVIEW
PFDAR: POST FLIGHT DATA ANALYSIS REVIEW
FLT: FLIGHT

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Vented Tank Resupply Experiment

VTRE TOP LEVEL SET OF TECHNICAL OBJECTIVES

VTRE Top-Level Technical Objectives

1.0* Vented Tank Fill

Evaluate the filling of a vented tank without the loss of liquid at:

- various inflow rates
- different drag vectors
- empty and different initial fill conditions
- different initial tank pressures

2.0* Direct Tank Venting

Investigate direct tank venting without loss of liquid with:

- different vent rates
- different drag vectors
- different fluid states
- for two tanks with different tank sizes/shapes

3.0* Capillary Liquid Retention Capability

Demonstrate that the ullage can be maintained near the vent during a low level disturbance

4.0* Capillary Positioning Device Liquid Recovery

Demonstrate the recovery capability of the capillary positioning system to position the ullage near the vent after a major disturbance

* Test categories refer to Technology Requirements Document (TRD) test set definition which also establishes the priority of testing

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Vented Tank Resupply Experiment

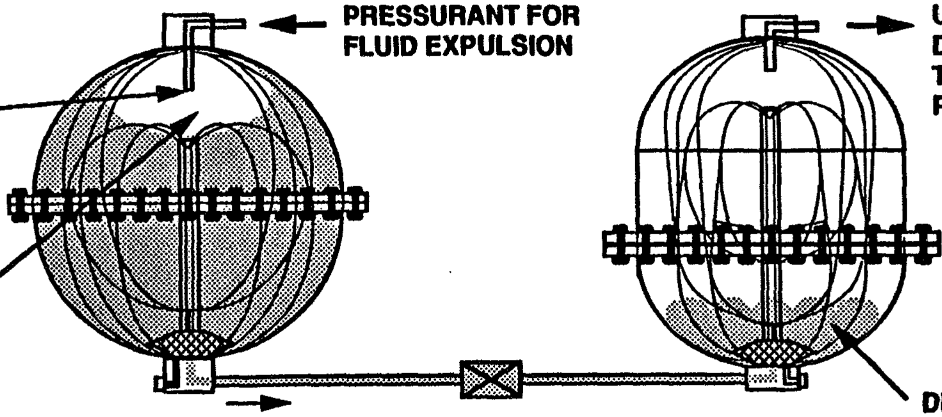
VTRE TECHNICAL OBJECTIVE No 1
TRANSFER WHILE VENTING

DEMONSTRATE STABLE INFLOW DURING TRANSFER TO 95% FILL LEVEL

PRESSURANT FOR FLUID EXPULSION

ULLAGE VENTING DURING TANK FILL TO MAINTAIN DESIRED PRESSURE

VENT OUTLET LOCATED IN CENTER OF NON-WICKING REGION FOR BOTH TANKS



EARLY IN THE FILL PROCESS SHOWN

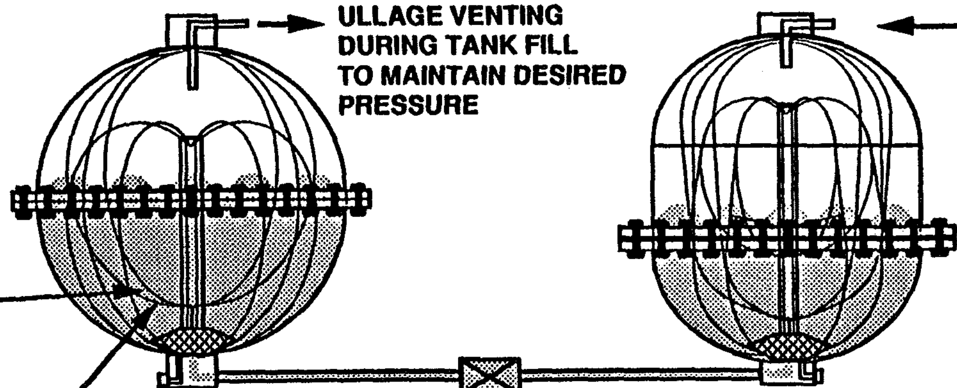
VANE CONFIGURATION MUST PROVIDE PREFERRED LOCATION FOR ALL LIQUID EXCEPT THE 5% ULLAGE REGION FOR BOTH TANKS

TEST TANK B TO TEST TANK A

DETERMINE THE ABILITY OF CAPILLARY VANE POSITIONING DEVICES TO HOLD THE INFLOW LIQUID INPLACE

ULLAGE VENTING DURING TANK FILL TO MAINTAIN DESIRED PRESSURE

PRESSURANT FOR FLUID EXPULSION



MIDWAY THROUGH THE FILL PROCESS SHOWN

CIRCUMFERENTIAL FLOW PATH WITHIN VANES PROVIDED TO ALLOW REDISTRIBUTION OF NON-UNIFORM INLET FLOW

TEST TANK A TO TEST TANK B

DETERMINE THE ABILITY OF CAPILLARY VANE POSITIONING DEVICES TO HOLD THE INFLOW LIQUID INPLACE

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CAPILLARY WICKING PRESSURE MUST EXCEED DYNAMIC PRESSURE OF INFLOWING LIQUID

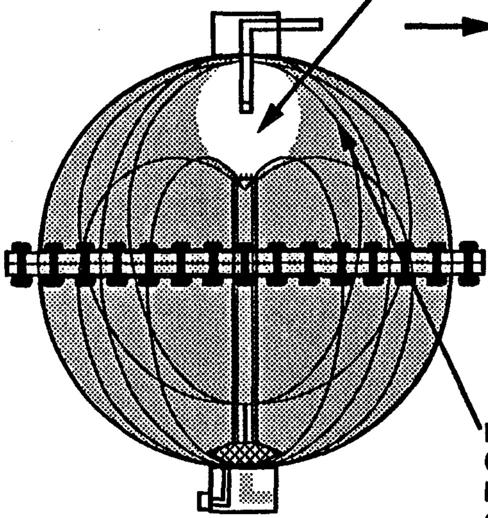
VTRE TECHNICAL OBJECTIVE No 2
DIRECT TANK VENTING

Vented Tank Resupply Experiment

VENT THE TANK WITHOUT ANY LIQUID LOSS WHILE AT THE 95% FILL LEVEL

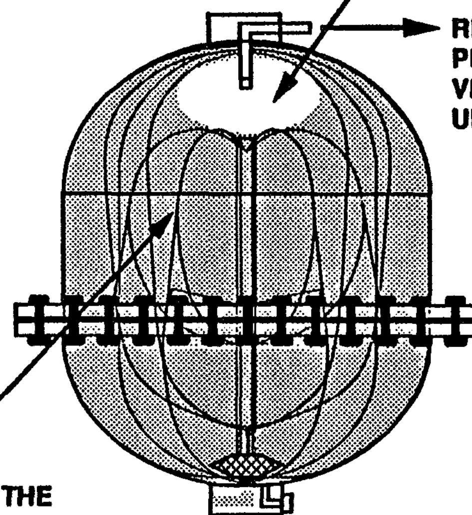
PROVIDE A WICKING VOLUME THAT WILL CONTROL THE LOCATION OF ALL LIQUID, LEAVING THE ULLAGE VOLUME (AT IT'S MINIMUM) AT THE A PREFERRED POSITION FROM WHICH TANK VENTING CAN BE ACCOMPLISHED

LARGER/SMALLER ULLAGE REGIONS THAN THE 95% FULL CONFIGURATION WILL ALSO OCCUPY THE REGION OF THE VENT PORT



REDUCE TANK PRESSURE BY VENTING TANK ULLAGE

RANGE OF FLOW THAT WILL BE PROVIDED WILL ALLOW FOR ASSESSMENT OF RATE EFFECT ON BUBBLE PHENOMENA



REDUCE TANK PRESSURE BY VENTING TANK ULLAGE

DETERMINE CAPABILITY OF THE CAPILLARY VANE DEVICE TO LIMIT LIQUID DISPLACEMENT OUT THE OPEN VENT BY PROVIDING A WICKING GRADIENT WITHIN THE VANE STRUCTURE THAT WILL PROPELL GAS BUBBLES TOWARD THE PREFERRED ULLAGE LOCATION

TEST TANK B

TEST TANK A

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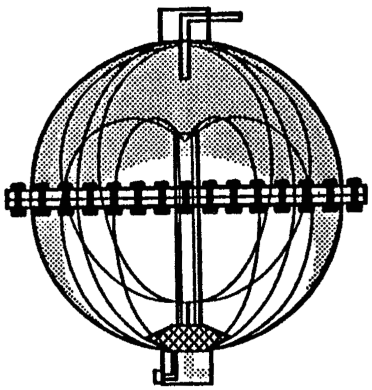
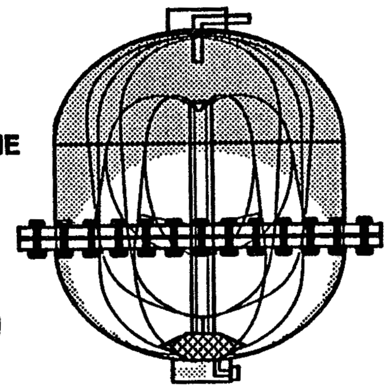
- GATHER BUBBLE GROWTH AND TRANSPORT DATA
- CHARACTERIZE EFFECTS DUE TO DRAG, VARYING FILL LEVELS AND TANK GEOMETRY

VTRE TECHNICAL OBJECTIVE No 3 & 4
CAPILLARY VANE DEVICE CAPABILITY

Vented Tank Resupply Experiment

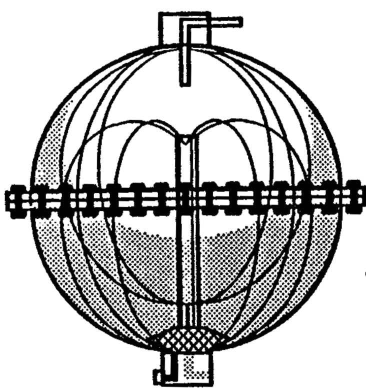
LIQUID RECOVERY CAPABILITY

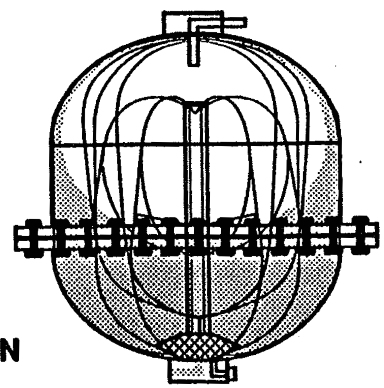




CAPILLARY VANE DEVICE WILL RELOCATE LIQUID AFTER AN INDUCED UPSET ACCELERATION

TEST START

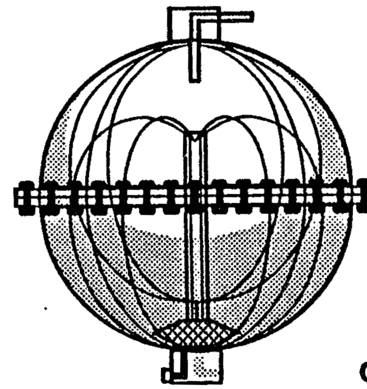




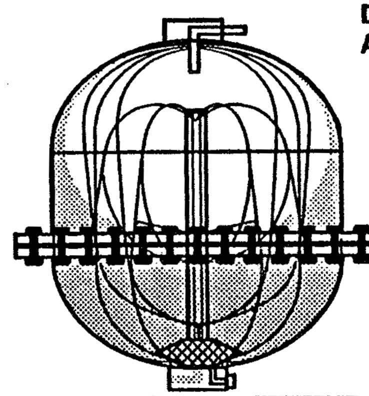
TEST END AFTER ACCELERATION REMOVED

DETERMINE REFILL TIME OF VANE DEVICE

LIQUID RETENTION CAPABILITY



CAPILLARY VANE DEVICE WILL HOLD LIQUID AGAINST A DISTURBING ACCELERATION

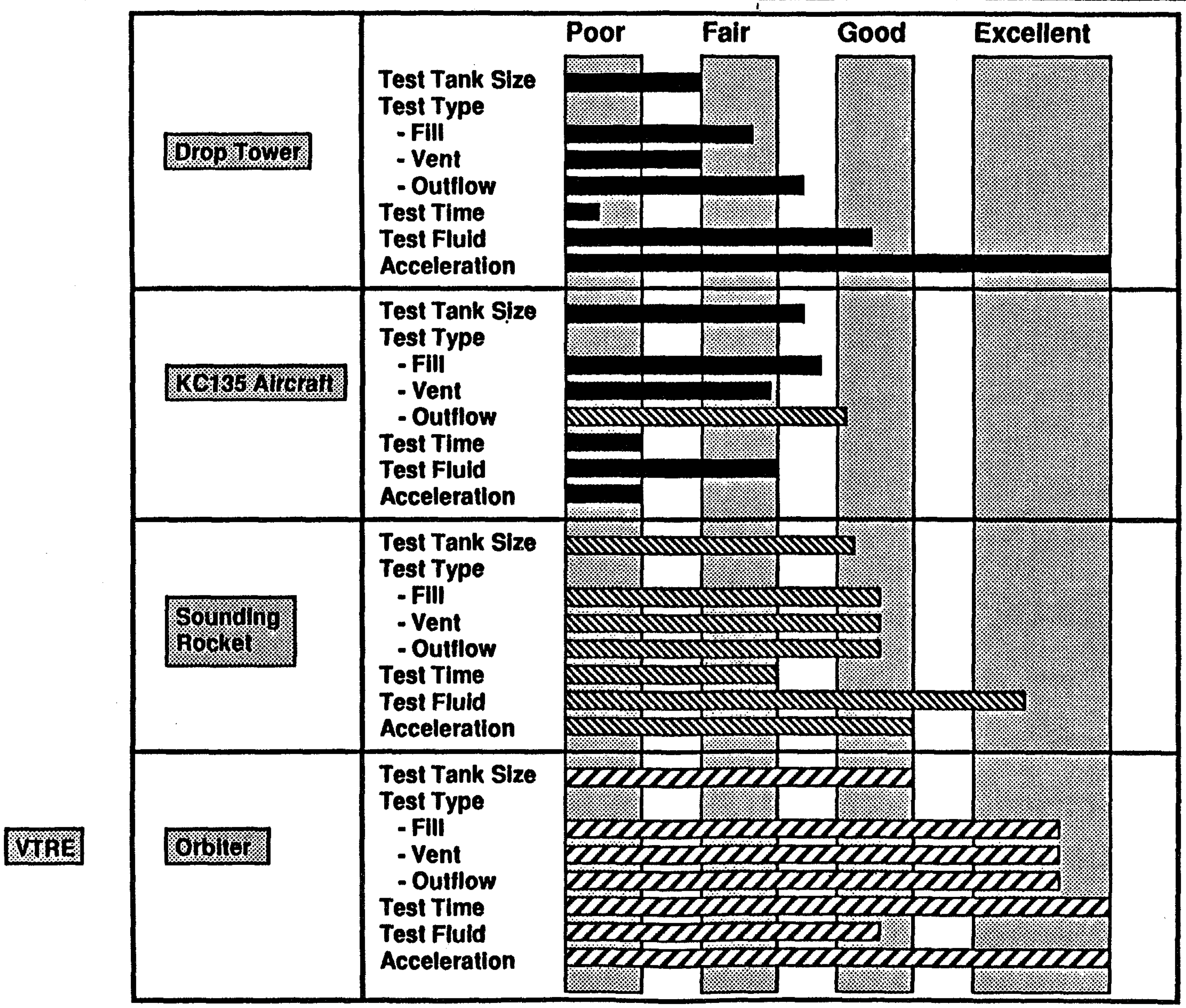


DETERMINE LIQUID RETENTION CAPABILITY IN ADVERSE, LOW-LEVEL G FIELD

THESE SECONDARY TESTS DO NOT IMPOSE ANY ADDITIONAL HARDWARE DESIGN REQUIREMENTS ON THE VANES OR THE TANKS

VALUE ADDED OF VTRE TO VANE TECHNOLOGY DATA BASE

Vented Tank Resupply Experiment



Potential that could be realized (no testing accomplished to date)

Potential to be realized by VTRE

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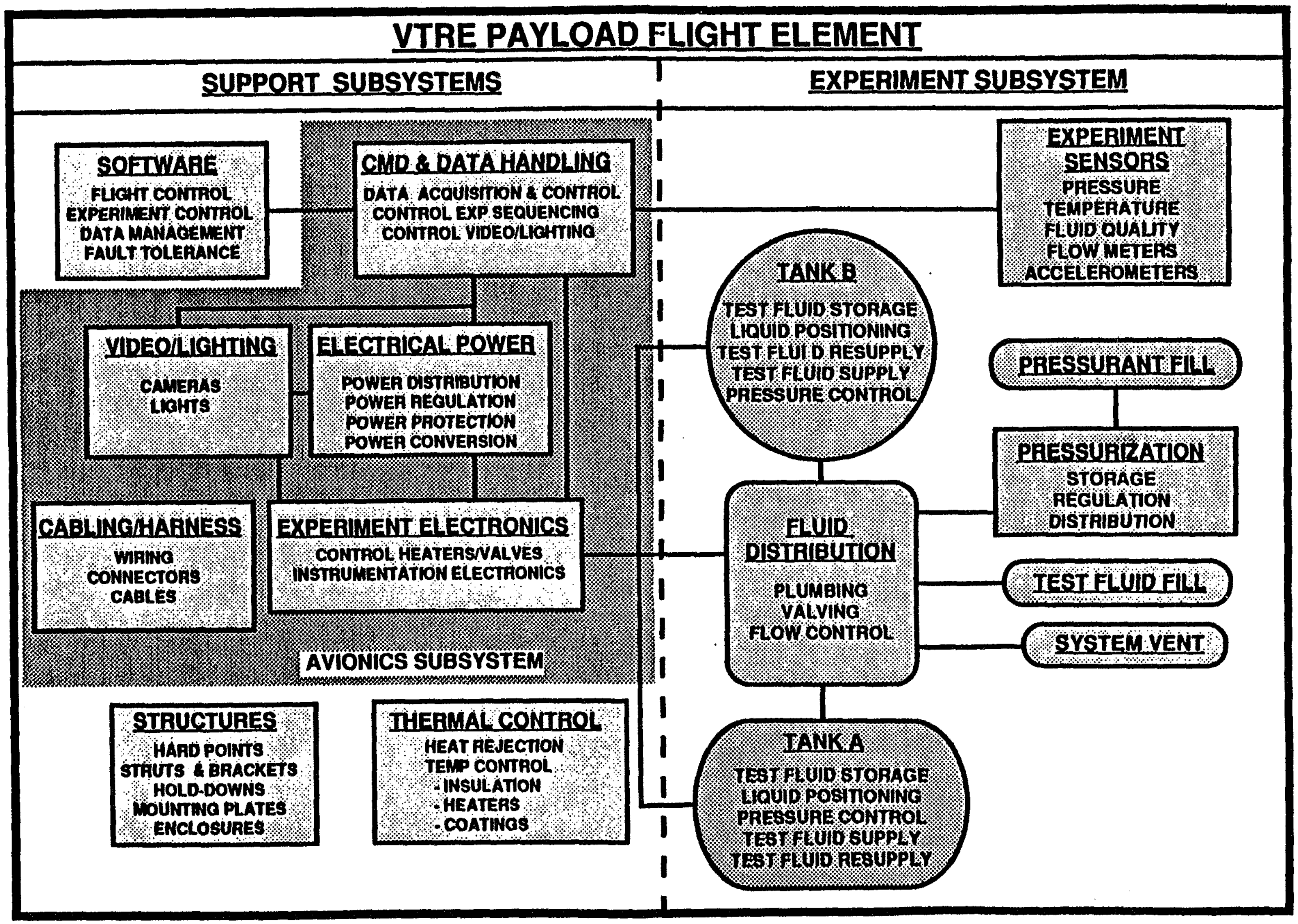
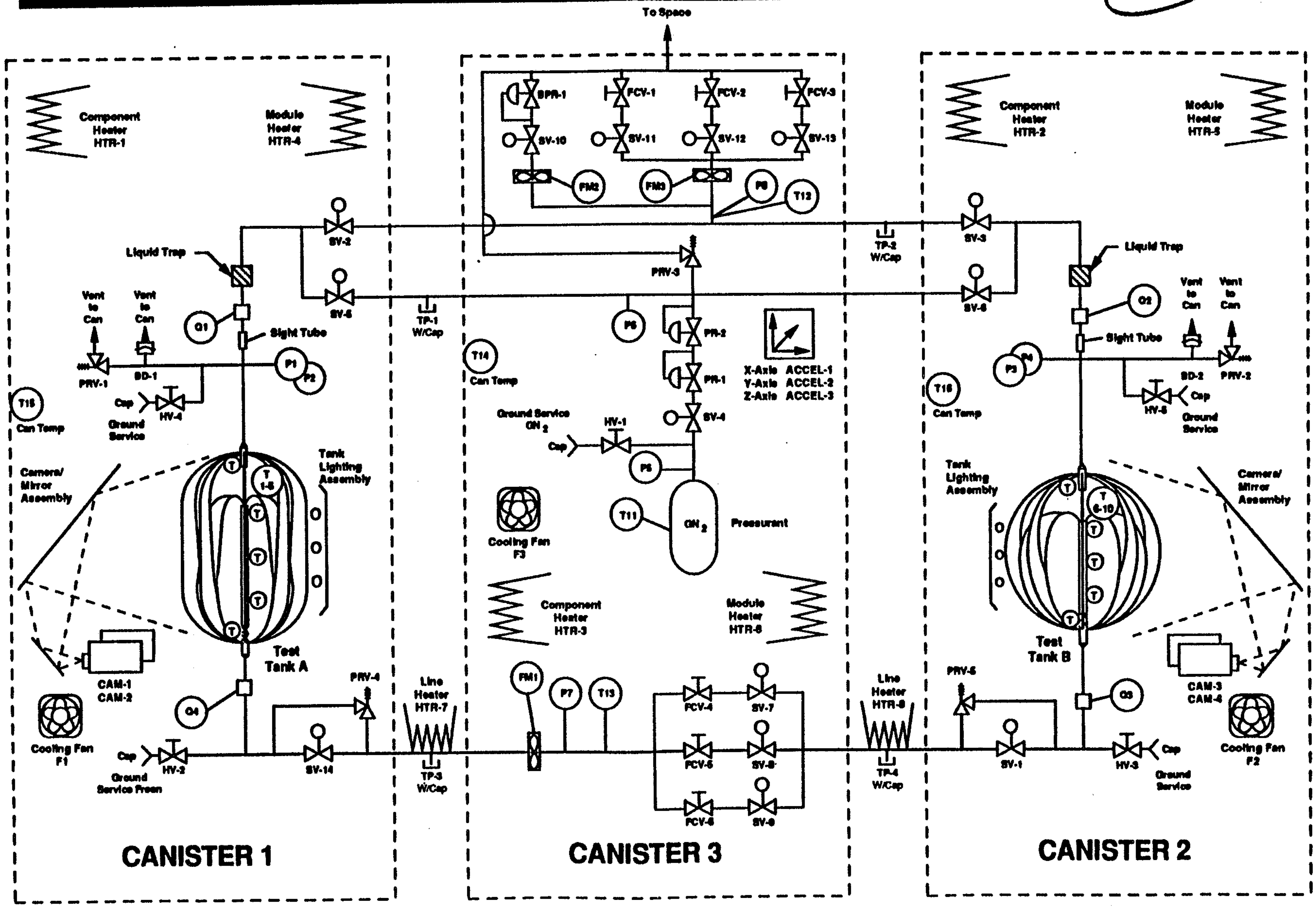
VTRE FUNCTIONAL BLOCK DIAGRAM

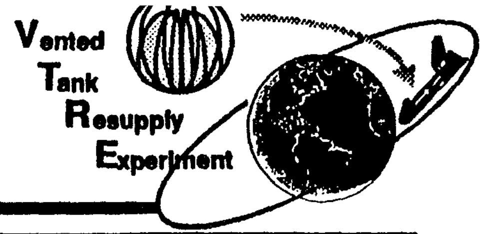
Vented Tank Resupply Experiment



Vented Tank Resupply Experiment

EXPERIMENT SUBSYSTEM SCHEMATIC



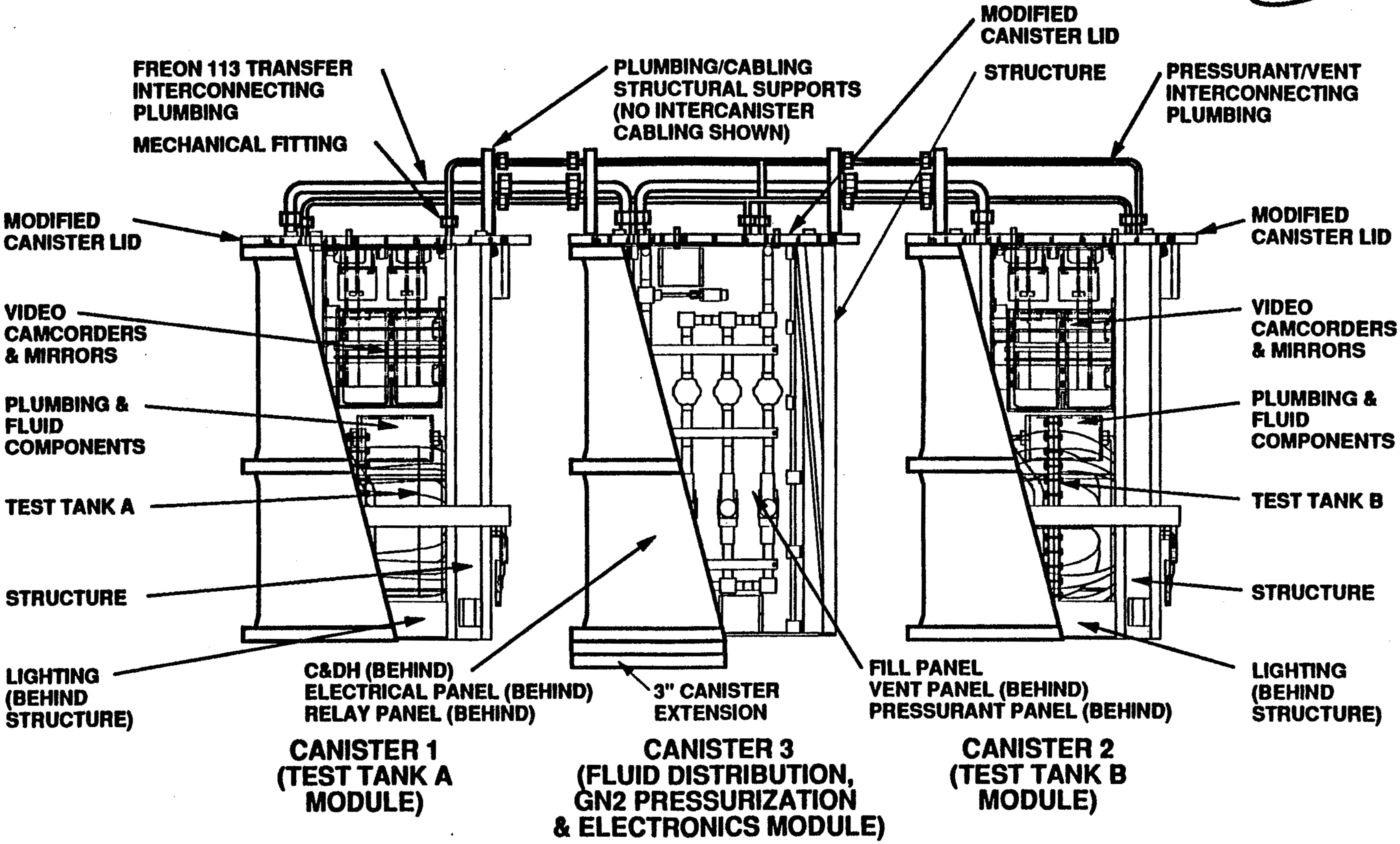


VTRE TEST DATA REQUIREMENTS

INSTRUMENT ID	RANGE	SAMPLE PERIOD (SEC)	NUMBER SENSORS	LOCATION	ACCURACY (+-)
EXPERIMENT SUBSYSTEM					
TEMPERATURE (R)					
TANK FLUID, T1-5	400-600	10 or 60	5	TANK A	1
TANK FLUID, T6-10	400-600	10 or 60	5	TANK B	1
PRESSURANT GAS, T11	400-600	10	1	PRESSURANT	1
VENT GAS, T12	400-600	1 or 10	1	CAN C	1
TRANSFER LINE FLUID, T13	400-600	10	1	CAN C	1
PRESSURE (PSIA)					
TANK A, P1,P2	0-35	1 or 60	2	TANK A	0.2
TANK B, P3,P4	0-35	1 or 60	2	TANK B	0.2
PRESSURANT, P5	0-3000	10	1	PRESSURANT	20
REGULATED PRESSURANT, P6	0-35	10	1	CAN C	0.2
TRANSFER LINE, P7	0-35	1	1	CAN C	0.2
VENT LINE, P8	0-35	1	1	CAN C	0.2
FLOWRATE					
TRANSFER LINE, F1 (GPM)	0.1-10	1	1	CAN C	0.1
HIGH FLOW VENT GAS, F2 (FT^3/MIN)	0.08-1.25	1	1	CAN C	0.1
LOW FLOW VENT GAS, F3 (FT^3/MIN)	0.008-0.2	1	1	CAN C	0.0002
QUALITY GAUGING					
TANK A VENT GAS, Q1	0.0-1.0	1	1	TANK A	0.05
TANK B VENT GAS, Q2	0.0-1.0	1	1	TANK B	0.05
TRANSFER LINE FLOW, Q3	0.0-1.0	1	1	TANK B	0.05
TRANSFER LINE FLOW, Q4	0.0-1.0	1	1	TANK A	0.05
MISCELLANEOUS					
TANK A INTEGRATED LIQUID (FT^3)	0-.84	1	1	CAN A	0.1
TANK B INTEGRATED LIQUID (FT^3)	0-.84	1	1	CAN B	0.1
ACCELERATION (G'S)	+1 to -1	1	3	CAN C	1.00E-05
TIME (SEC)	0-7 DAYS	1	1	N/A	N/A

VTRE FLIGHT CONFIGURATION IN 3 MODIFIED HITCHHIKER CANISTERS

Vented Tank Resupply Experiment



- CANISTERS LIDS REQUIRE MODIFICATION TO ACCOMMODATE ELECTRICAL/MECHANICAL PENETRATIONS
- INTERCONNECT PLUMBING REQUIRES THERMAL PROTECTION
- INTERCONNECT PLUMBING AND CABLING REQUIRES STRUCTURAL SUPPORT TO THE CANISTER LIDS

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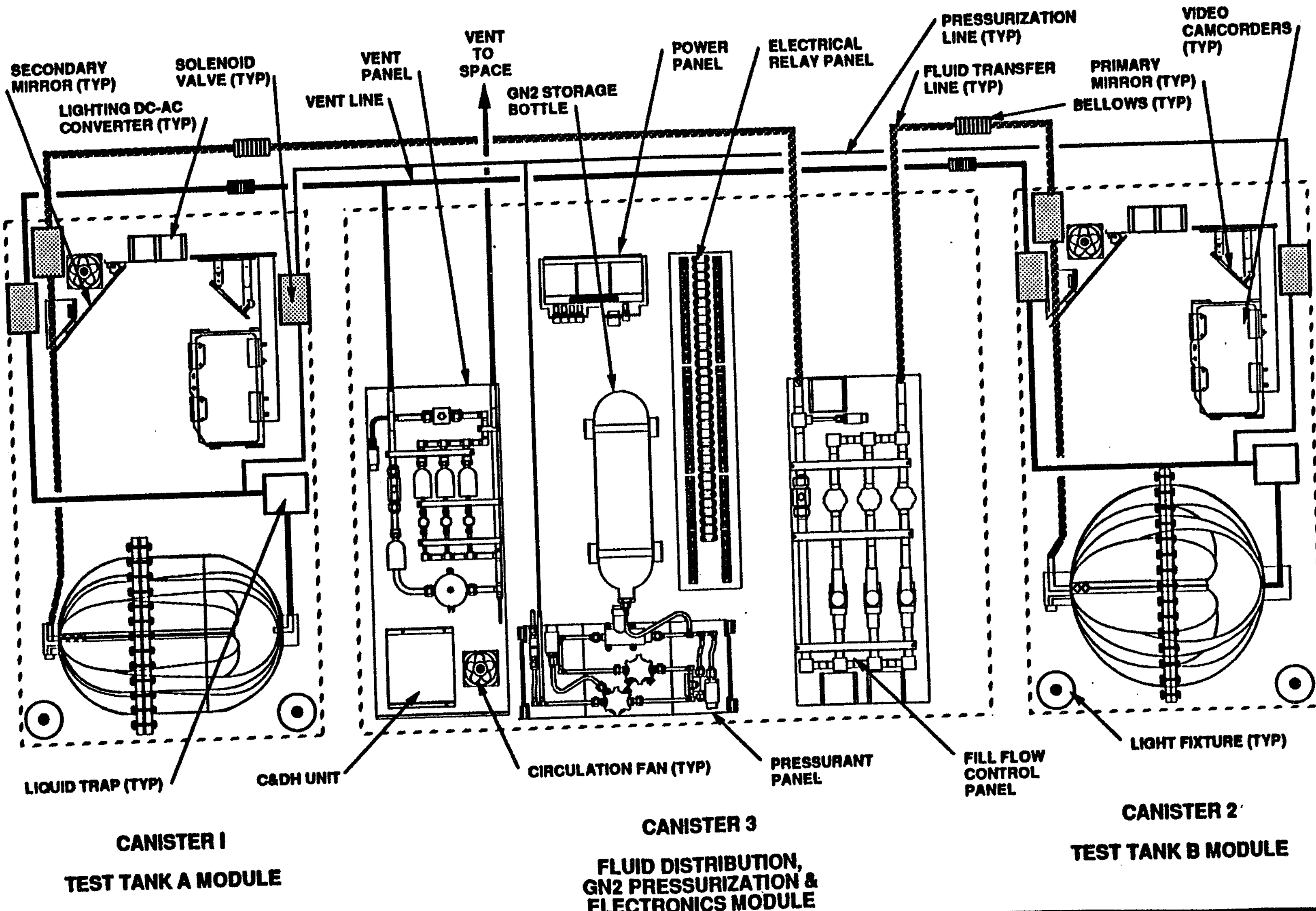
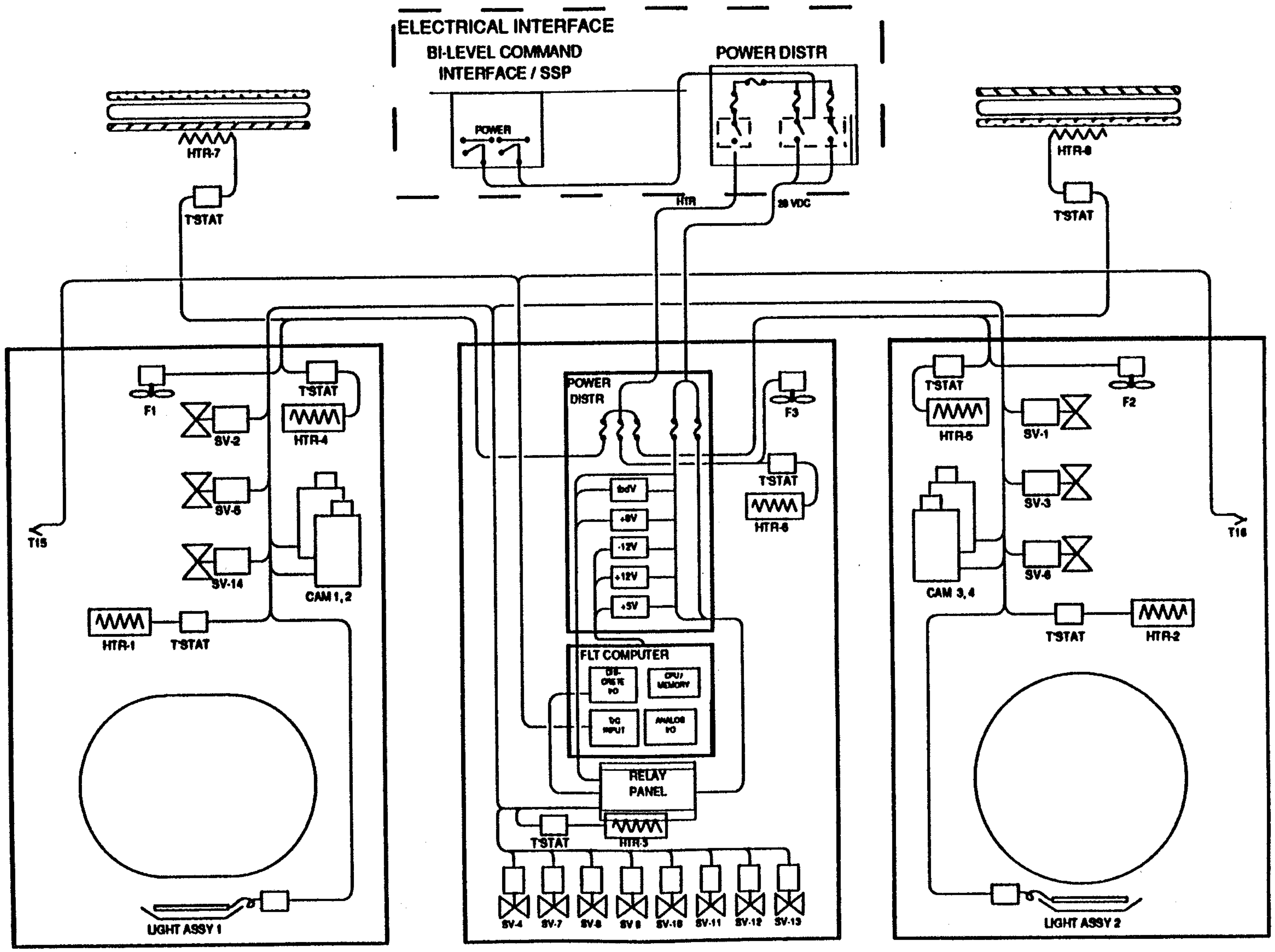
VTRE ELEMENTS INTERRELATIONSHIP



VTRE ELECTRICAL POWER SUBSYSTEM (EPS) SCHEMATIC

Vented Tank Resupply Experiment



CANISTER 1 CANISTER 3 CANISTER 2

Vented Tank Resupply Experiment

VTRE POWER REQUIREMENTS

SUBSYSTEM	# OF UNITS	POWER PER UNIT (W)	AVERAGE LOAD (W) OPERATING 8 HRS MAX	AVERAGE LOAD (W) IDLE	PEAK LOAD (W)	NOTES	HERITAGE
ELECTRICAL POWER SUBSYSTEM			16	16	16		
POWER DISTRIBUTION UNIT	1	3	3	3	3		
DC/DC CONVERTER	1	13	13	13	13	WITH 80% EFFICIENCY	ADTECH
EXPERIMENT SUBSYSTEM			90.2	3	90.2		
PRESSURE TRANSDUCERS	7	0.6	4.2	3	4.2	CONTINUOUSLY ON	DRUCK
FLOWMETER (INCLUDING ELECTRONICS)	3	1	3	0	3		FLOW TECH
LIQUID/GAS SENSORS	4	1	4	0	4		
SOLENOID VALVES - LATCHING	5	0	0	0	0		
SOLENOID VALVES - LOW POWER	4	1	4	0	4	* 18 W INRUSH, 1 W HOLD	
SOLENOID VALVES - HIGH POWER	5	25	75	0	75		
COMMAND & DATA SUBSYSTEM			21.8	21.8	21.8		
COMPUTER & I/O UNITS	1	21.8	21.8	21.8	21.8		PRO-LOG
THERMAL			0	67	180		
CANISTER 1 MAKEUP HEAT	1	60	0	31	60	AVERAGE OF 35 WATTS	
CANISTER 2 MAKEUP HEAT	1	60	0	31	60	DISSIPATION IN EACH	
CANISTER 3 MAKEUP HEAT	1	60	0	5	60	CANISTER, MINIMUM	
VIDEO SUBSYSTEM			51.5	0	103	MAX 2 CAMERAS ON	
VIDEO CAMCORDER	4	5.5	5.5	0	11	NORMALLY ONLY 1	SONY
LIGHTING SYSTEM (INCL POWER CONVERSION, 2 LAMPS EACH)	2	46	46	0	92	50% EFFICIENT POWER SUPPLY	SFMD
EXPERIMENT TOTALS			179.5	107.8	411		
HITCHHIKER AVIONICS			112	112	225		
HH- INTERNAL ELECTRONICS	1	100	100	100	125		HH
SUPPLEMENTAL HEATERS	2	50	12	12	100	12 W FOR FANS	HH
VTRE TOTALS			291.5	219.8	636		

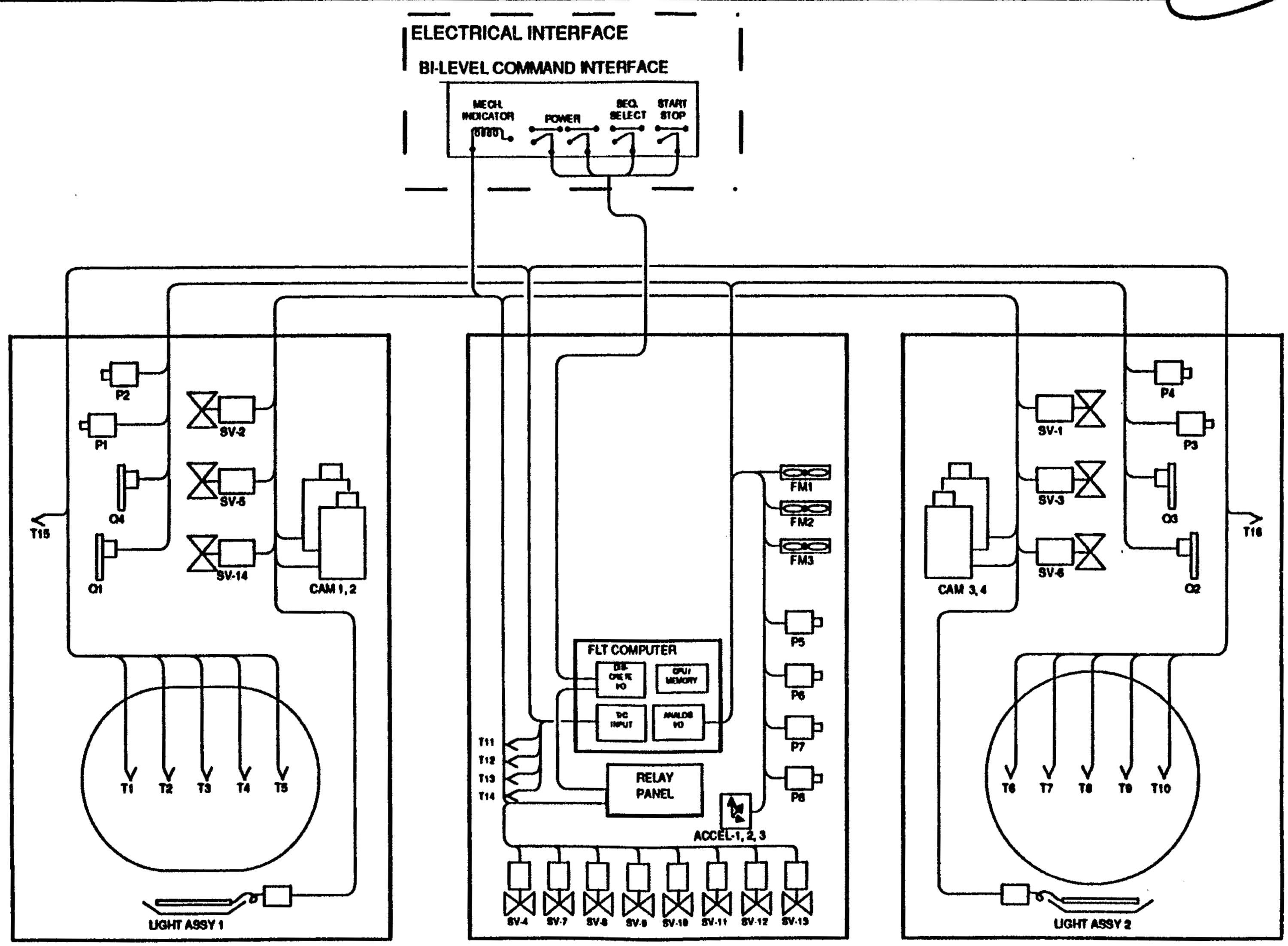
OPERATING POWER = 1.53 KWH OVER 8 HOURS
PLUS 4.31 KWH IDLE POWER OVER 36 HOURS FOR MINIMUM MISSSION
PLUS 16.29 KWH IDLE POWER OVER 136 HOURS FOR MAXIMUM MISSSION

TOTAL = 5.84 KWH FOR EXPERIMENT OVER MISSION
TOTAL = 17.82 KWH FOR EXPERIMENT OVER MISSION
TOTAL = 10.24 TO 32.22 KWH IF INCLUDE HH AVIONICS

NOTE : EXPERIMENT CAN BE POWERED DOWN BEFORE AND AFTER TEST SEQUENCES

VTRE COMMAND & DATA HANDLING SUBSYSTEM (C&DHS) SCHEMATIC

Vented Tank Resupply Experiment



CANISTER 1 CANISTER 3 CANISTER 2

Vented Tank Resupply Experiment

VTRE PAYLOAD ASSEMBLIES

Assembly	Weight	Size X (in)	Size Y (in)	Size Z (in)	Mount	Power	FOV	Temp (C) STG	Temp (C) OP
Test Tank A Module	191.6 lbs	19.75	19.75	31.25	Modified Canister 1	84.7 w (36.2 w)	None	-15 to +42	
Test Tank B Module	109.4 lbs	19.75	19.75	31.25	Modified Canister 2	84.7 w (36.2 w)	None	-15 to +42	
Fluid Distribution, GN2 Pressurization & Electronics Module	139.6 lbs	19.75	19.75	31.25	Modified Canister 3*	73.6 w (47.4 w)	None	-4 to +42	

All weights include 20% dry weight margin

Operating - no heaters on (Idle - heaters may be on, fans will be on)

* May require 3' canister extension

MARTIN MARIETTA

Vented Tank Resupply Experiment

VTRE WEIGHT SUMMARY

PAYLOAD DRY WEIGHT	295.9
CONSUMABLES	85.5
PAYLOAD (W/O HH AVIONICS OR PLATES)	381.4
MARGIN OF 20% ON ALL BUT FLUID ITEMS	59.2
LAUNCH WEIGHT WITH MARGIN	440.6
HITCHHIKER GAS CAN (3 REQUIRED)*	420.0
HITCHHIKER AVIONICS + PLATE	210.0
TOTAL IF CHARGED W/ HH AVIONICS	1070.6

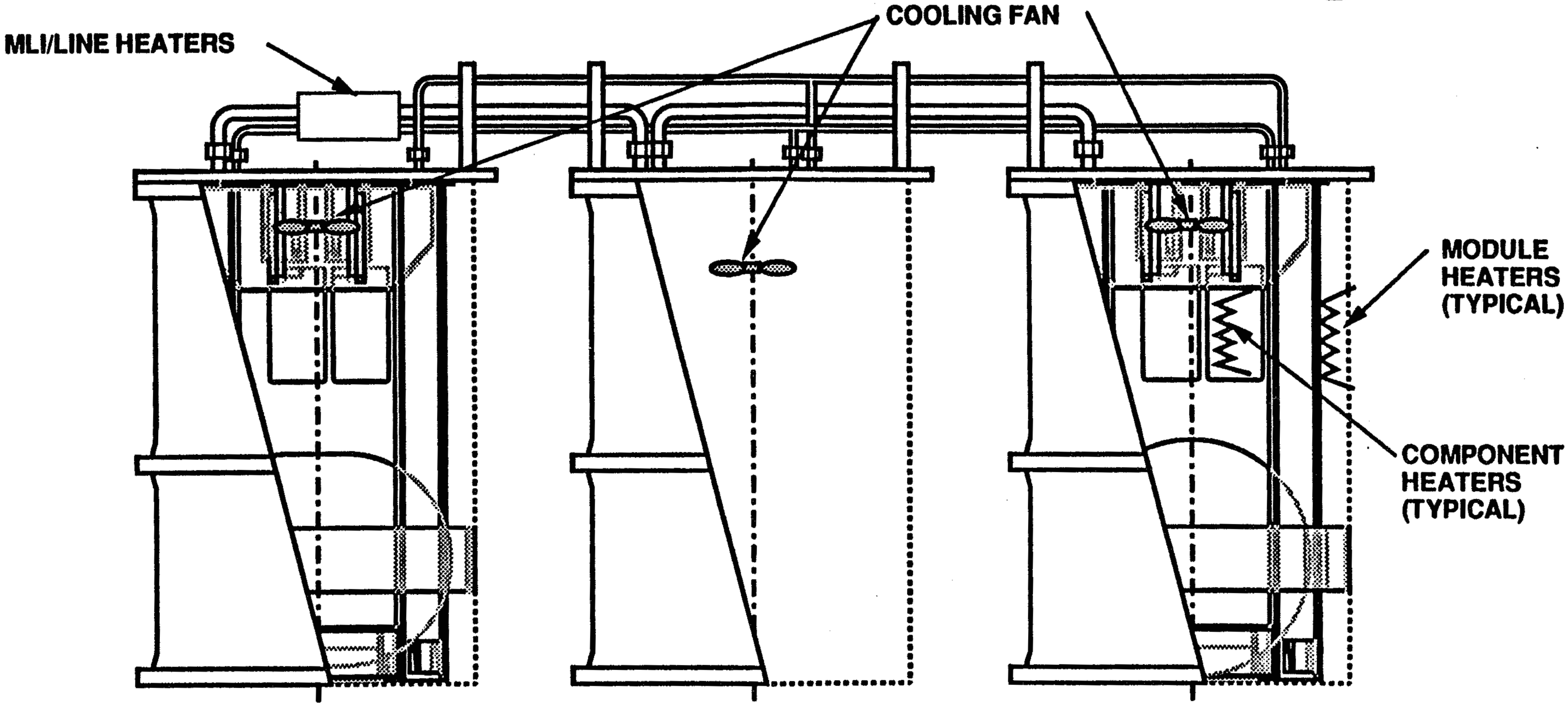
COMPARISON WITH CARRIER CAPACITIES

	TANK/VIDEO SYSTEMS		AVIONICS/PLUMBING	HH AVIONICS
MASSES (LBM) W/ MARGIN	TANK A 191.6	TANK B 109.4	139.6	210.0
GAS CAN LIMITS	200.0*	200.0*	200.0*	210.0

* GAS CAN LISTED WITH 200 LBM MAX CARRYING CAPABILITY, BUT STRUCTURE CAN HOLD 400 LBM WITH 140 REQUIRED FOR GAS CAN ALLOWING A 260 LBM PAYLOAD IN GAS CAN

VTRE THERMAL CONTROL APPROACH USING HITCHHIKER CANISTERS

Vented Tank Resupply Experiment



CANISTER 1 (TEST TANK A MODULE)

CANISTER 3 (FLUID DISTRIBUTION, GN2 PRESSURIZATION & ELECTRONICS MODULE)

CANISTER 2 (TEST TANK B MODULE)

- CANISTER 1 & 2 CONTAINS CAMERAS, LIGHTS, VALVES AND ELECTRONICS THAT WILL BE THERMALLY MANAGED BY CONDUCTION/CONVECTION
- CANISTER 3 CONTAINS VALVES, MICROPROCESSOR, POWER AND ELECTRONICS THAT WILL BE THERMALLY MANAGED BY CONDUCTION/CONVECTION
- INTERCONNECT PLUMBING REQUIRES THERMAL PROTECTION (MLI AND HEATERS)
- CANISTER COOLING FAN APPROACH REQUIRES THAT CANISTERS BE PRESSURIZED (15 PSIG GN2)
- ALL CANISTERS CONTAIN COMPONENT & MODULE HEATERS FOR COLD CASE MANAGEMENT

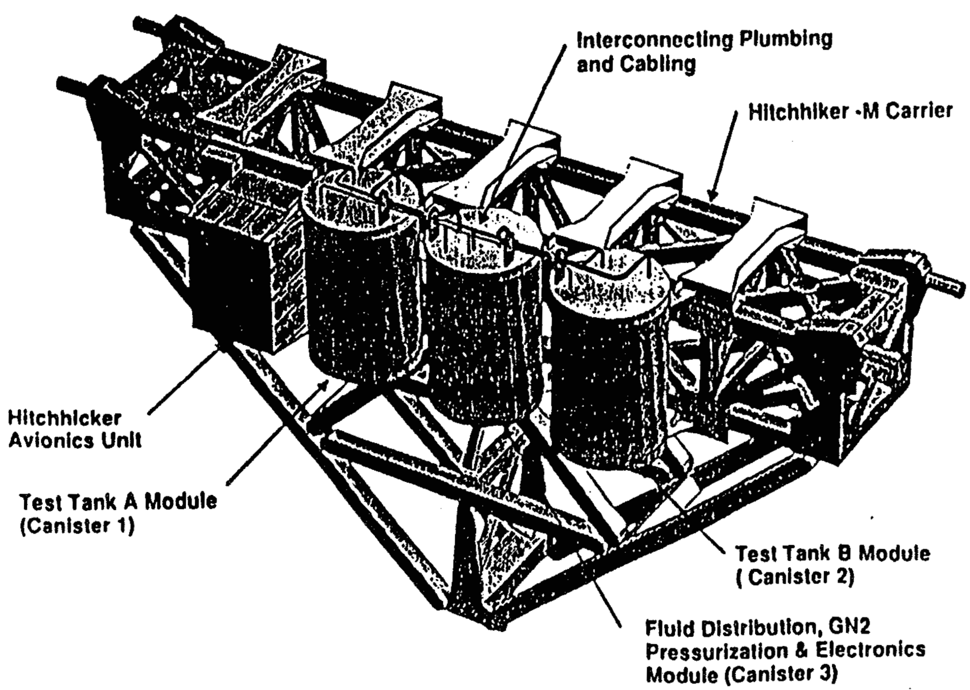
CARRIER APPROACH

Vented Tank Resupply Experiment

- USING THREE HITCHHIKER CANISTERS THAT WILL HAVE THE TOP MOUNTING PLATES MODIFIED TO ALLOW FOR FLUID AND ELECTRICAL CONNECTIONS BETWEEN THE UNITS

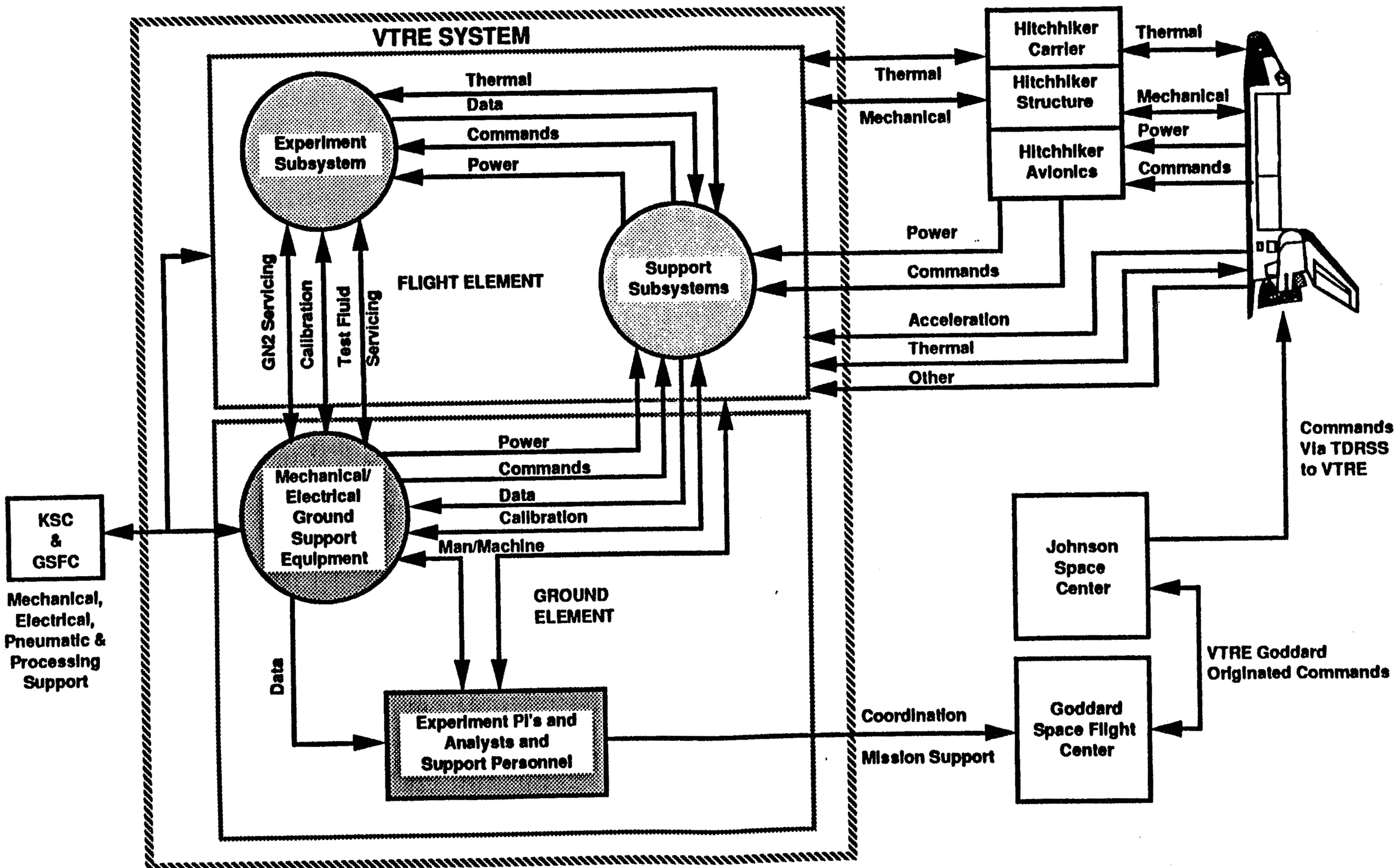
- CANISTERS WILL BE MOUNTED SIDE BY SIDE ON A HH-M MPESS BEAM TO BE FLOWN AS PART OF A MIXED EXPERIMENT

  - CONFIGURATION COULD ALSO BE FLOWN AS A HH-G PAYLOAD ATTACHED TO THE ORBITER SIDEWALL



VTRE ELEMENTS- INTERNAL AND EXTERNAL FUNCTIONAL INTERFACES

Vented Tank Resupply Experiment



Vented Tank Resupply Experiment

MISSION APPROACH

- VTRE TESTING TO BE INTEGRATED INTO MISSION TIMELINE IN A MANNER THAT WILL MINIMIZE THE ORBITER IMPACTS
  - MAJORITY OF TESTING CAN BE PERFORMED DURING SLEEP PERIODS
  - ONLY ASTRONAUT INVOLVEMENT WILL BE TWO THRUSTINGS AND ONE ATTITUDE MANEUVER
- EXPERIMENT CONSISTS OF 7 SEQUENCES THAT CAN BE CONTROLLED VIA THE GSFC POCC
  - COMMANDS ISSUED IN SIMILAR MANNER TO THOSE SET FROM SSP IN ORBITER
  - POWER ON/OFF, SEQUENCE START/STOP, AND SEQUENCE SELECT ARE ONLY REQUIRED COMMANDS

Vented Tank Resupply Experiment

EXPERIMENT REQUIREMENTS IMPACT ON ORBITER MISSION

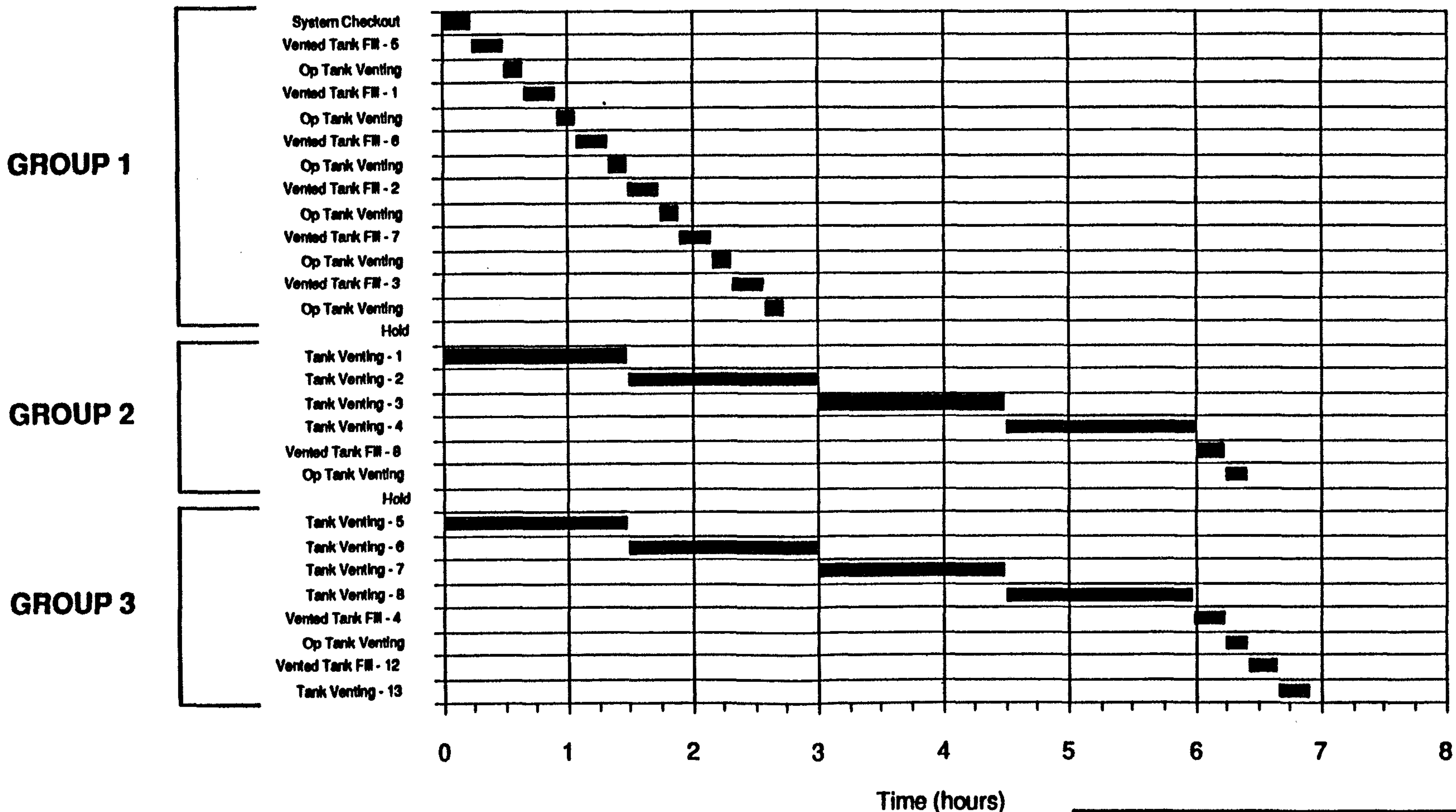
• ACCELERATION	SHUTTLE PRCS TO PROVIDE THRUST IN X AXIS USING THRUSTERS TO PROVIDE A PURE TRANSLATION IN X.
• ALTITUDE	STS ALTITUDE DEPENDENT ON PRIMARY PAYLOADS; NO VTRE REQUIREMENT.
• ATTITUDE	FOR VENTING TEST WITH UPSETTING ACCELERATION, SHUTTLE MUST BE PLACED IN HIGH DRAG CONFIGURATION WITH SHUTTLE Z AXIS ALIGNED WITH VELOCITY VECTOR
• MISSION DURATION	44 HOURS MINIMUM TO AS LONG AS 144 HOURS
• CREW INTERACTION	THRUSTER OPERATIONS; SHUTTLE ORIENTATION TO HIGH DRAG ORIENTATION; STANDARD SWITCH PANEL
• PAYLOAD WEIGHT	1070.6 LBS INCLUDING 3 GAS CANISTERS AND HH AVIONICS
• PAYLOAD POWER	191.5 W - AVE POWER DURING TEST; MINIMUM OF 5.84 KWH/FLIGHT TO MAXIMUM OF 17.82 KWH/FLIGHT (DEPENDENT ON LENGTH OF POWERED-UP HOLD PERIODS)
• EXPULSION RATE OF FREON IN P/L BAY	2.5 LBM/HR MAXIMUM VENT RATE
• TEST START	START ASAP AFTER P/L BAY DOORS OPEN & CREW AVAILABLE

MARTIN MARIETTA

Vented
Tank
Resupply
Experiment

EXPERIMENT GROUPING TIMELINE

- EXPERIMENT BROKEN INTO 7 GROUPINGS OF TESTS WITH HOLD PERIODS INBETWEEN (OTHER GROUPINGS OF TESTS ARE POSSIBLE AND WILL BE EVAUATED)

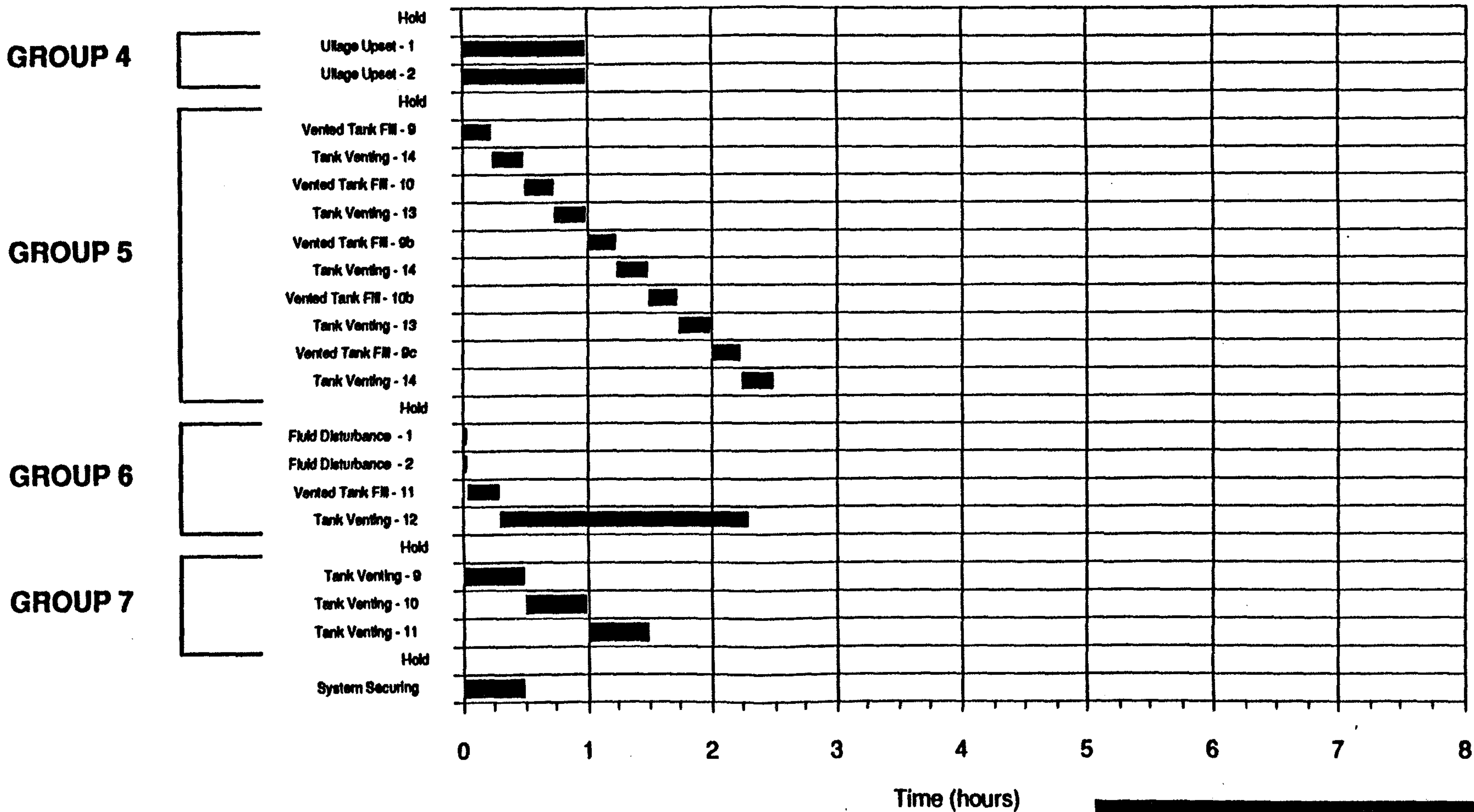


MARTIN MARIETTA

EXPERIMENT GROUPING TIMELINE (CONCLUDED)

Vented Tank Resupply Experiment

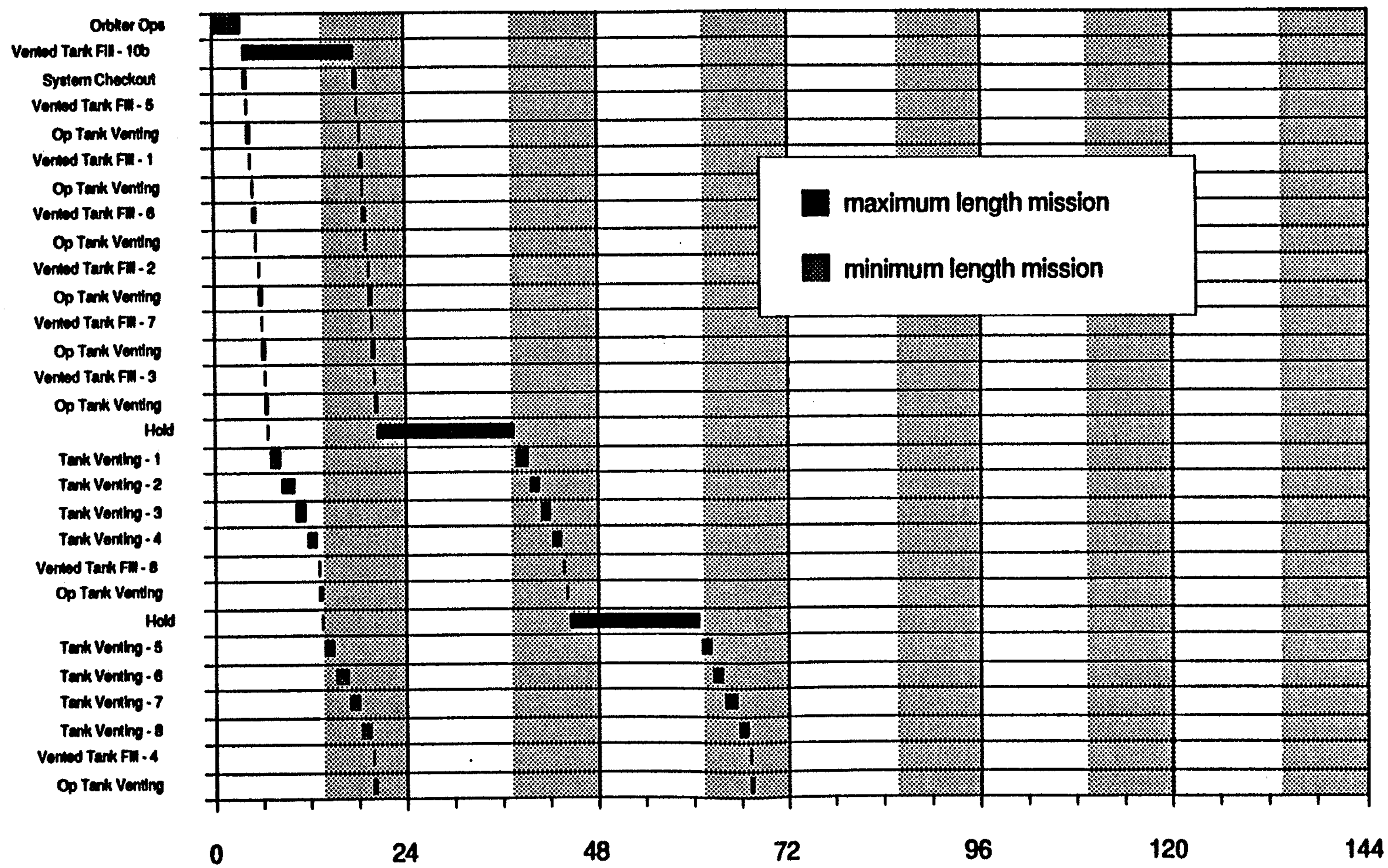
- EXPERIMENT BROKEN INTO 7 GROUPINGS OF TESTS WITH HOLD PERIODS INBETWEEN (OTHER GROUPINGS OF TESTS ARE POSSIBLE AND WILL BE EVAUATED)



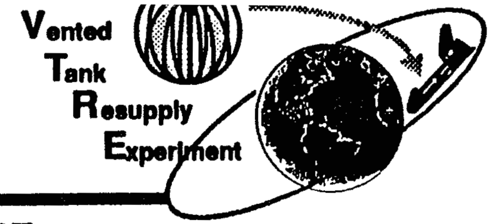
FULL EXPERIMENT TIMELINE

Vented Tank Resupply Experiment

- TIMELINE GENERATED FROM DATABASE WHICH TRACKS ENTIRE FLOW OF VTRE TESTING
  - GENERATED TWO TIMELINES THAT BOUND MISSION REQUIREMENTS



FULL EXPERIMENT TIMELINE (CONCLUDED)



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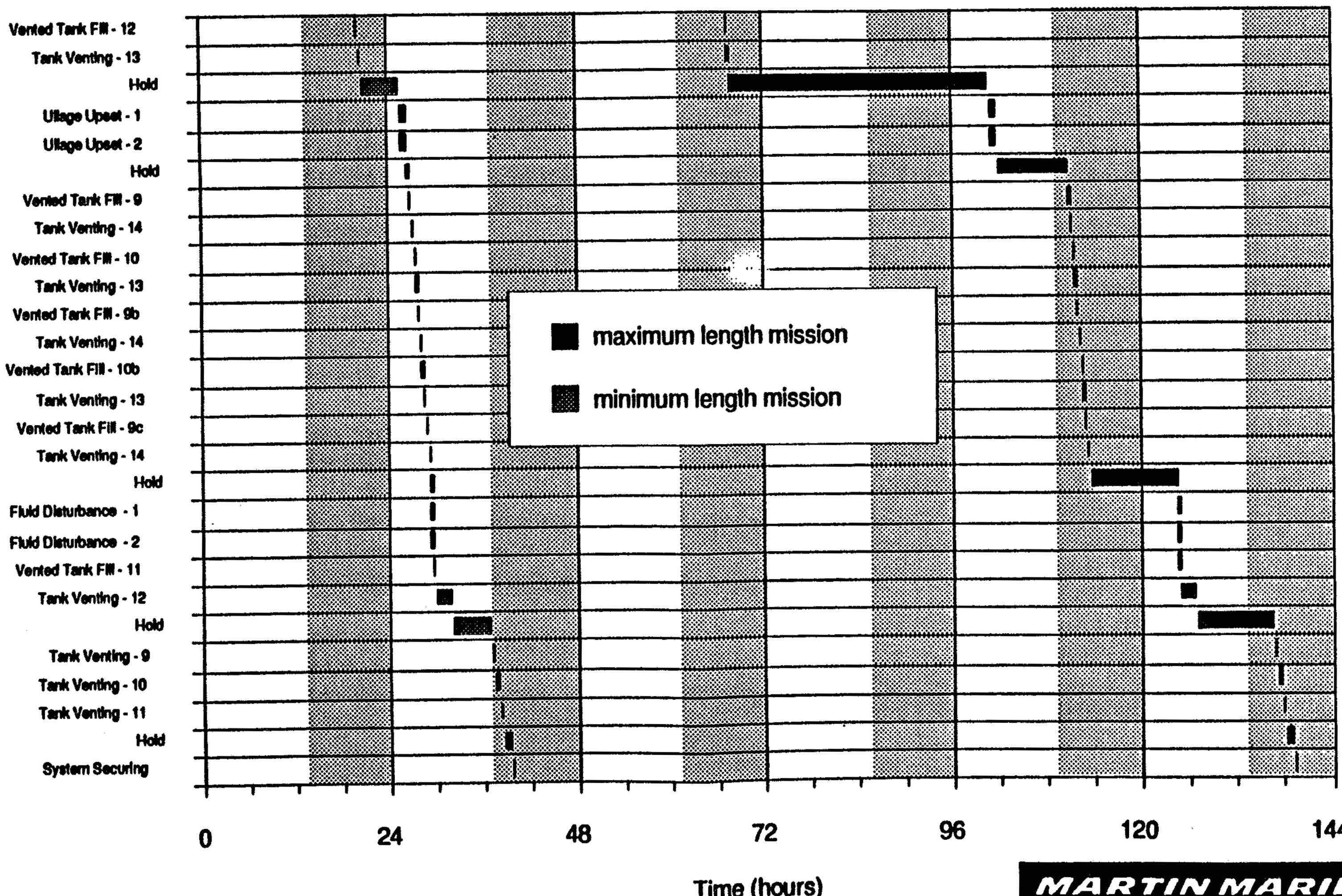


VTRE TOP ISSUES & MAJOR AREAS WHICH DRIVE THE DESIGN, COST & SCIENCE FIDELITY OF THE PROGRAM

Vented Tank Resupply Experiment

- CARRIER SELECTION (GSFC HAS BASELINED THE HH-M CARRIER)
  - MEETING 35 - 50 HZ NATURAL FREQUENCY REQUIREMENT
  - NEGOTIATING MASS/CG CONSTRAINTS AND RESULTING CONFIGURATION ON THE CARRIER
  - MANIFESTING / CARRIER AVAILABILITY
  - THREE INTERCONNECTING CANISTER ARRANGEMENT

- FREE DRIFT OF ORBITER & DESIRED THRUSTING FOR FLUID SETTLING
  - ALL DRIFTING ACCELERATION REGIMES
  - INTERFACE POSITION AND MAINTENANCE OF STABILIZED INTERFACE

- SCALING OF EXPERIMENT DESIGN
  - TANK SIZES / EXPERIMENT WEIGHT
  - TANK SHAPES
  - ORBITER ORIENTATION
  - VANE CONFIGURATION

- STS SAFETY REQUIREMENTS
  - MORE EXTENSIVE SCRUTINY THAN SIMPLE GAS CAN EXPERIMENTS
  - VENTING FREON 113 IN THE BAY

- AMOUNT OF CONSUMABLE VIDEO TAPE FOR DATA COLLECTION
  - IMPACT ON DESIRED EXPERIMENT SET
  - ALLOCATION OF EXPERIMENTS TO THE RESOURCE

- EXPERIMENT RISK
  - DESIGN APPROACH IS SINGLE STRING, EXCEPT WHERE SAFETY DICTATES OTHERWISE
  - AUTONOMOUS TEST WITH NO GROUND MONITORING
  - USE OF "INEXPENSIVE" NOT PREVIOUSLY FLIGHT QUALIFIED HARDWARE

- USE OF HH CANISTERS
  - CONTAINMENT
  - USE OF HIGH STRUCTURAL SAFETY FACTORS / MARGIN
  - LID MODIFICATION IMPACT
  - IMPACT OF FREON SERVICING

- PROTECTING TEST FLUID FROM THERMAL HEATING

- THERMAL MANAGEMENT USING PRESSURIZED CANISTER AND CIRCULATION FANS

TANK PRESSURE CONTROL EXPERIMENT (TPCE)

Flight Experiments Technical Interchange Meeting
October 6, 1992

Mike Bentz
Boeing Defense & Space Group
Seattle, WA

OVERVIEW

- Tank Pressure Control Experiment (TPCE) is a small self-contained STS payload
- Objective is to test jet mixer for cryogenic fluid pressure control
- Flown on STS-43 in August 1991
- Demonstrated reliable pressure control with low-energy mixer
- Reflight scheduled for late October on STS-52

PROJECT ORGANIZATION

- Sponsored by In-Space Technology Experiments Program (In-STEP)
- Managed by NASA Lewis Research Center
- Design, fabrication, flight data analysis by Boeing Defense & Space Group
- STS integration managed by NASA Goddard Space Flight Center

PROJECT PHILOSOPHY

- Quick-response, relatively low-cost experiment
- GAS carrier chosen for ease of integration, manifesting
- Class D Modified approach used for hardware development
  - minimum cost
  - commercial-grade components
  - reduced product assurance requirements (except safety)
  - extra system-level testing to assure of flight readiness

RISK MANAGEMENT APPROACH

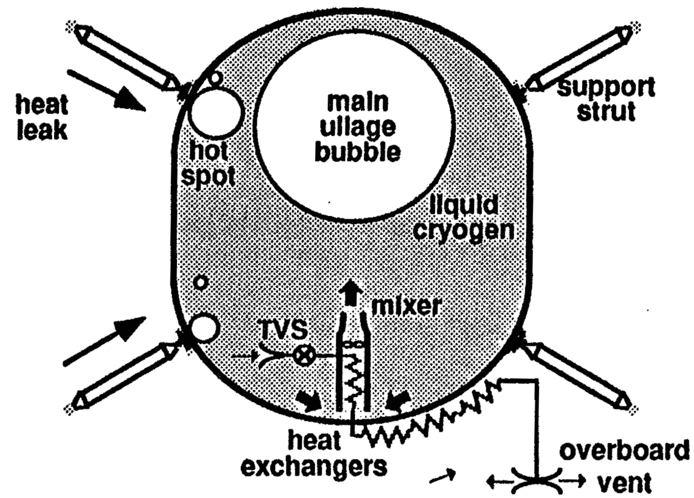
- Designed with redundancy where most beneficial
- Designed for minimal requirements on Orbiter and crew
- Based designs and components, where possible, on those used on prior payloads
- Tested prototype in low-g on Lewis Learjet Microgravity Test Facility
- Performed five complete Mission Simulation tests prior to delivery

PROBLEM AND OBJECTIVES

Problem:

Storage of cryogens for long durations in low-g requires efficient and reliable control of tank pressure.

Active jet mixing is leading candidate for pressure control but energy addition results in boiloff penalty. Low-energy mixing requires in-space test.



Objectives:

- Determine jet mixing effectiveness in realistic low-g environment, as measured by ability of jet to:
  - penetrate vapor bubbles and reach all tank regions
  - reduce pressure in minimum time / minimum energy
  - equilibrate fluid temperatures
- Provide data for development of analytical models

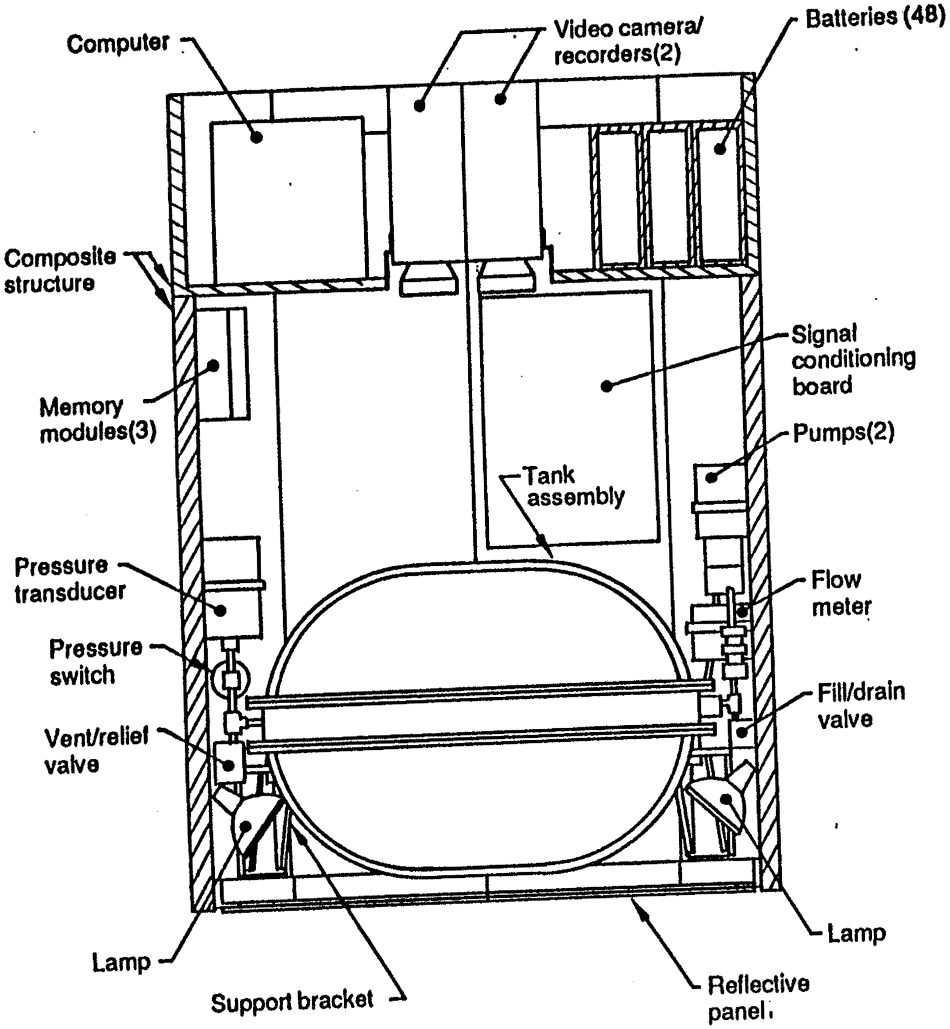
APPROACH

- Refrigerant 113 simulates cryogens
- 0.5 cu-ft tank filled to 83% level
- Pressure raised by heating, then reduced by mixing
- 38 test runs to determine effect of flow rate, acceleration environment, heater location
- Packaged as an autonomous STS payload using GAS carrier

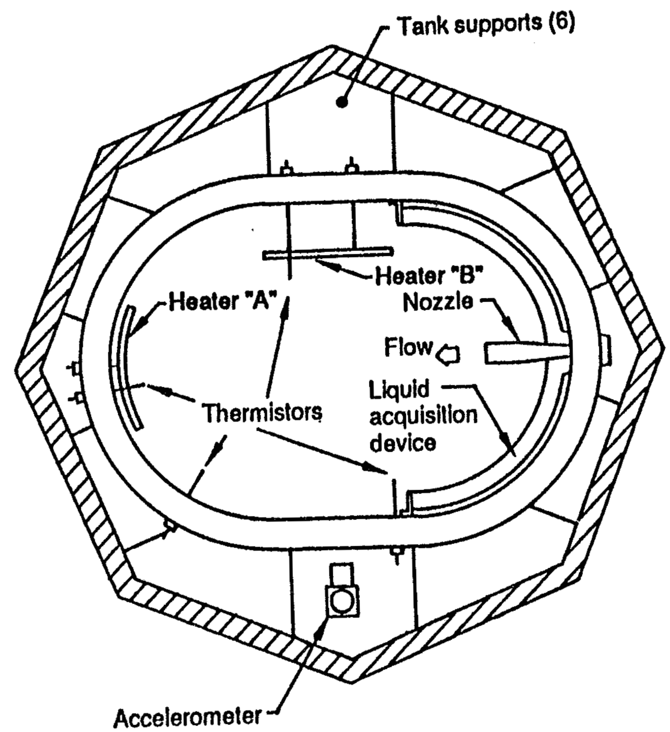
TANK PRESSURE
CONTROL
EXPERIMENT

PAYLOAD CONCEPT

BOEING



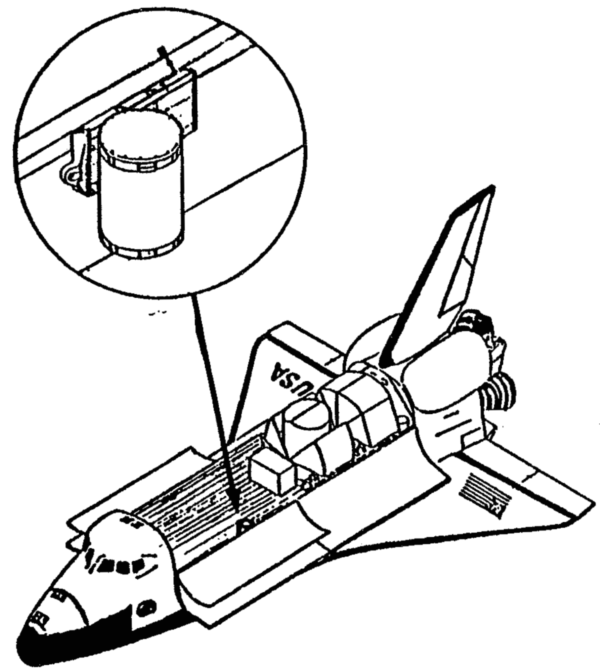
Cutaway View



Note: For clarity, some components are not shown in their true orientations

STS MISSION PLAN

- Secondary payload using Get-Away Special carrier
- Payload size: <200 lbm, 5.0 cu-ft
- Tank major axis aligned with Orbiter X-axis, mixer nozzle at aft end
- OMS burns will settle liquid at mixer end
- Tail-first Orbiter orientation during first sleeping period (8 hours)
- Payload activation by baroswitch during launch
- Test duration: approximately 27 hours



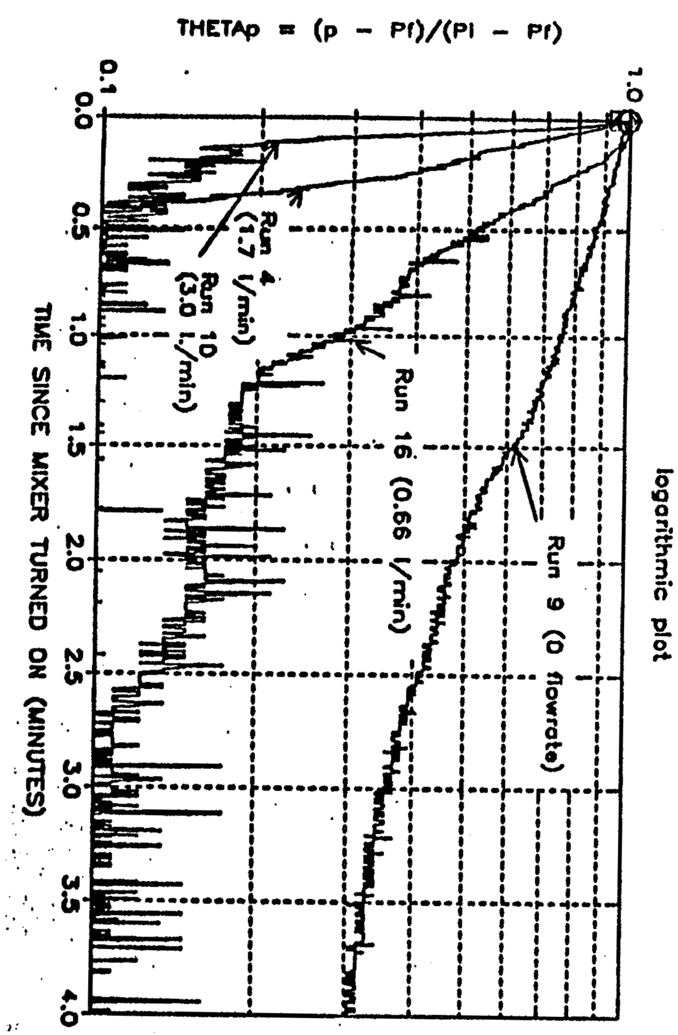
SUMMARY OF DATA - VIDEO

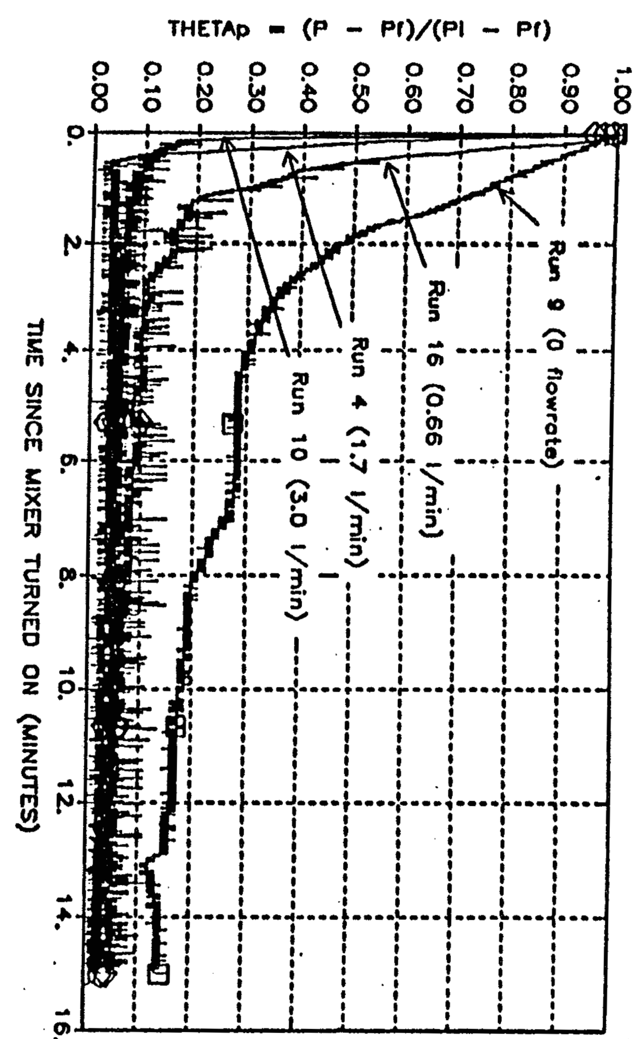
- Effects of acceleration environment
- Heating (pressure rise) phase
- Self-mixing behavior
- Mixing flow patterns

TANK PRESSURE
CONTROL
EXPERIMENT

NORMALIZED PRESSURE HISTORIES - FOUR TYPICAL RUNS

BOEING

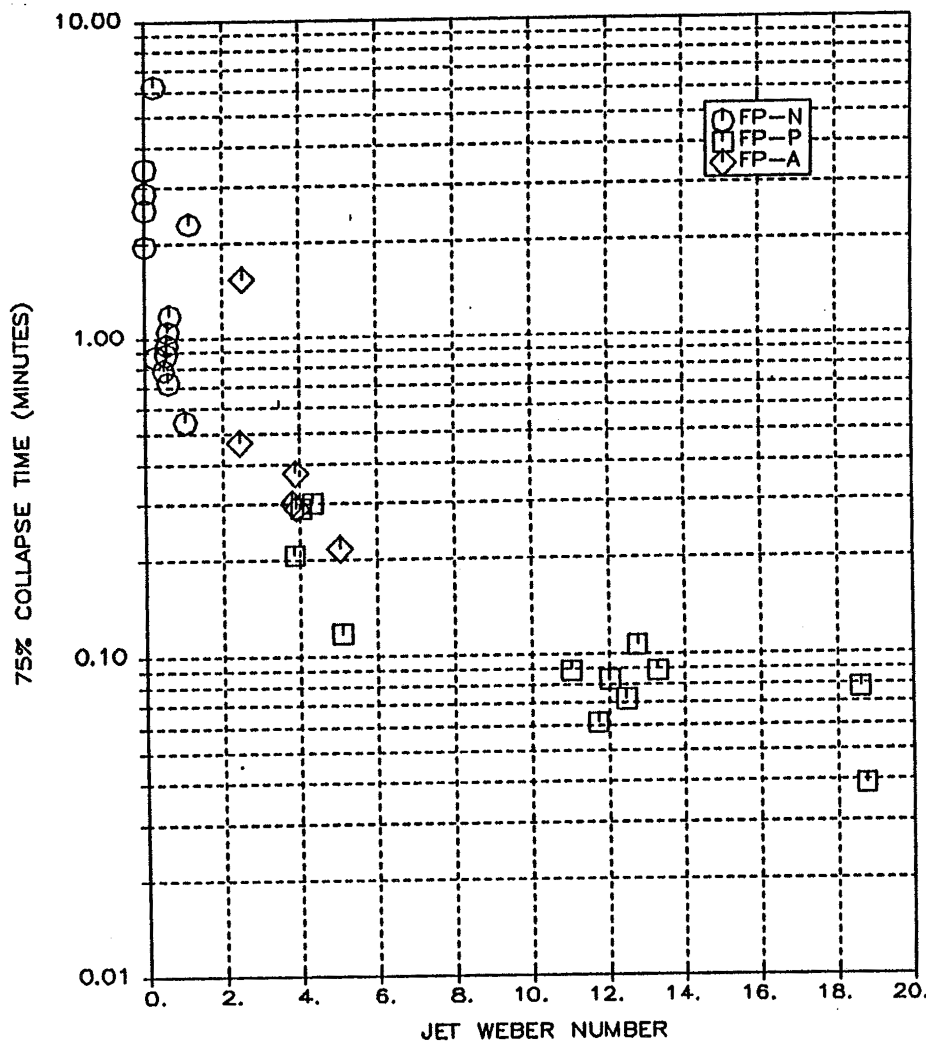




TANK PRESSURE
CONTROL
EXPERIMENT

PRESSURE REDUCTION TIME VERSUS WEBER NO. & FLOW PATTERN

BOEING



RESULTS AND CONCLUSIONS

- All objectives met or exceeded, no failures
- Data support thesis that low-energy jet is efficient pressure control device
  - Moderate velocities cause complete circulation, rapid pressure drop
  - Low velocities also cause reliable pressure drop with ~80% less energy added
  - Identified ranges of dimensionless jet momentum to be avoided
  - Generated large amount of digital and video data to support model development
- Identified potentially significant pressure rise phenomena requiring further study
- Payoff: - Cryogen pressure control shown to be manageable problem
  - Boiloff mass due to mixing can be reduced to insignificant level

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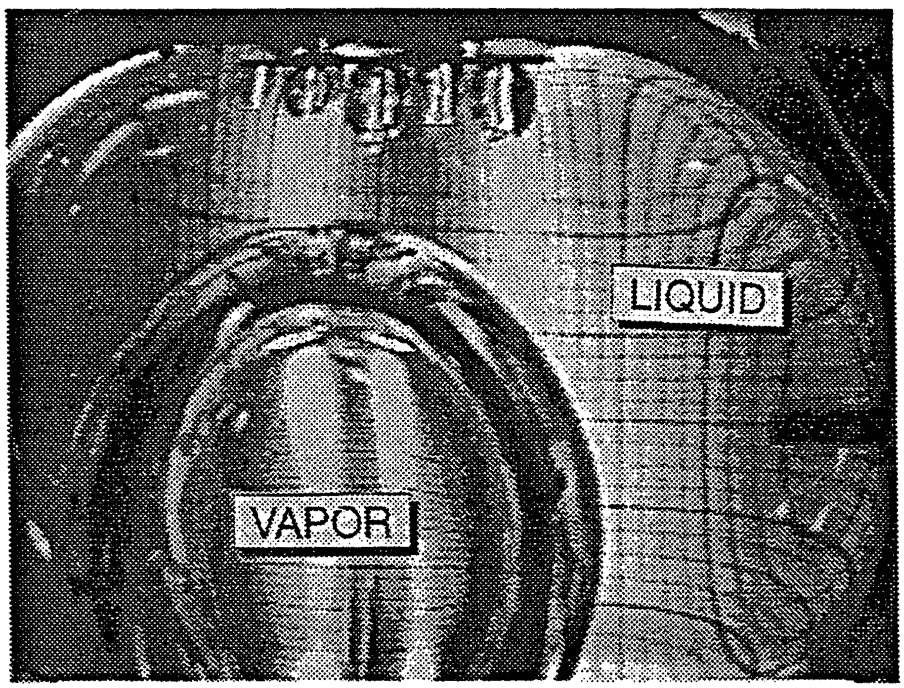
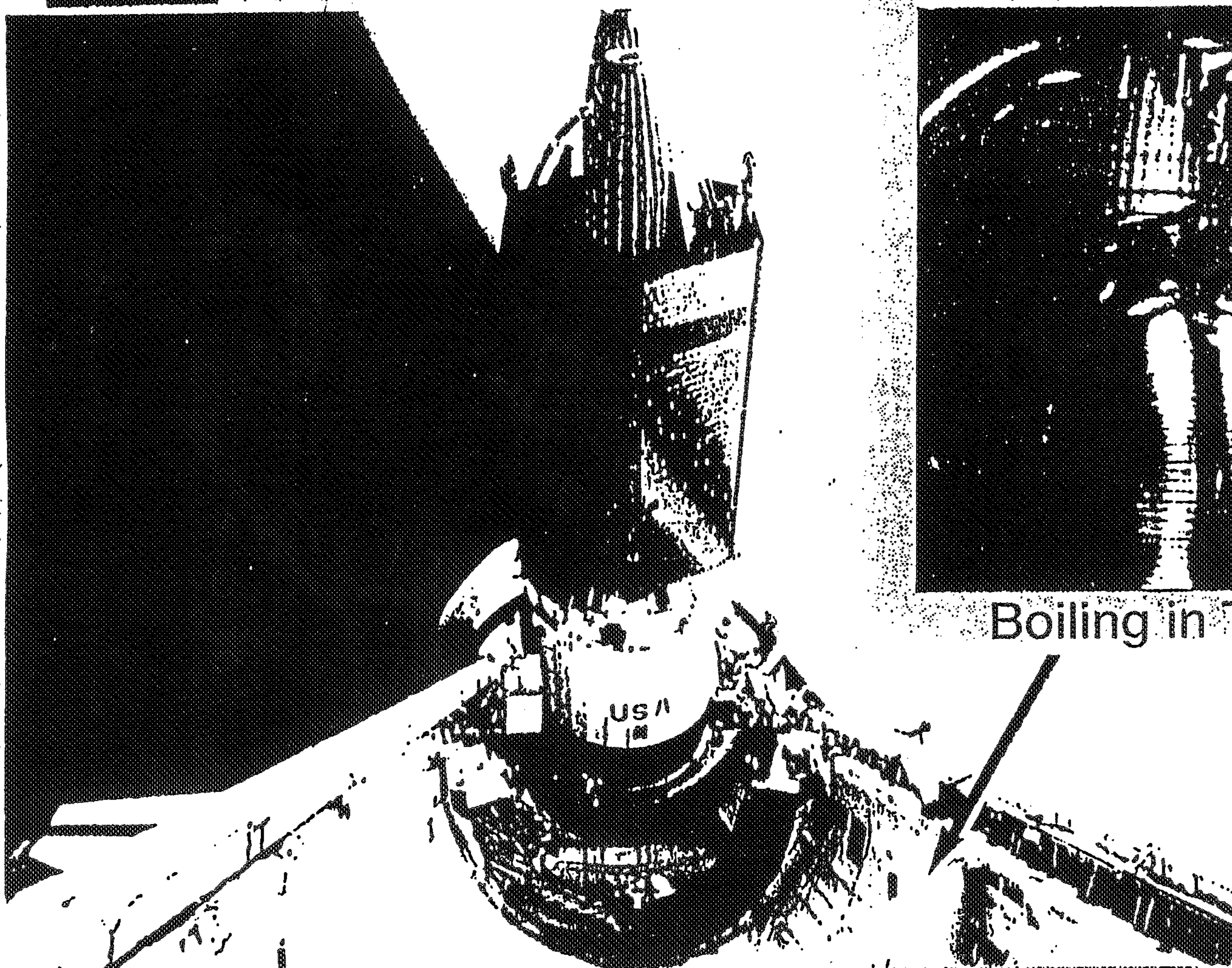
Tank Pressure Control Experiment/Thermal Phenomena
TPCE/TP

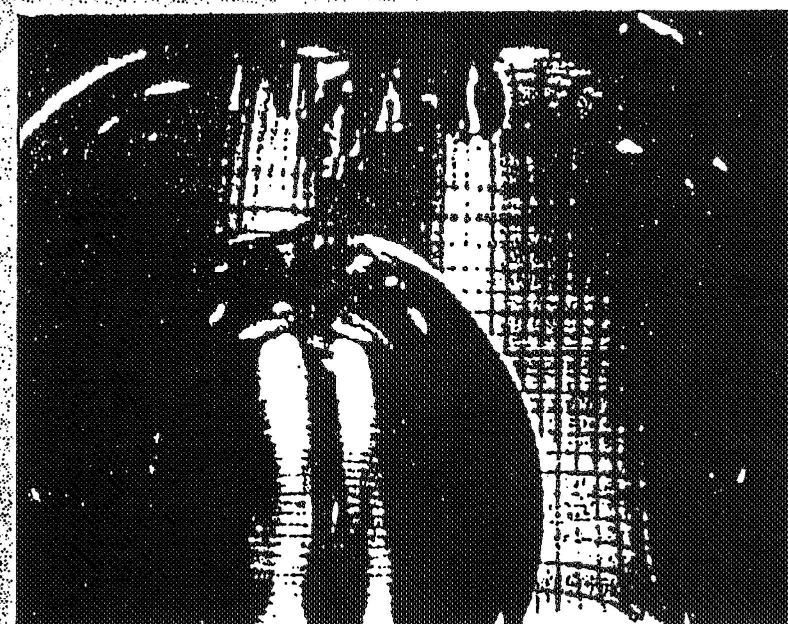
M. M. Hasan and R. H. Knoll
NASA Lewis Research Center



Flight Experiments Technical Interchange Meeting
October 5-9, 1992
Monterey, California

TPCE/TP is a reflight of TPCE which flew successfully on STS-43 (August 1991)





Boiling in TPCE freon tank

CD-92-61081

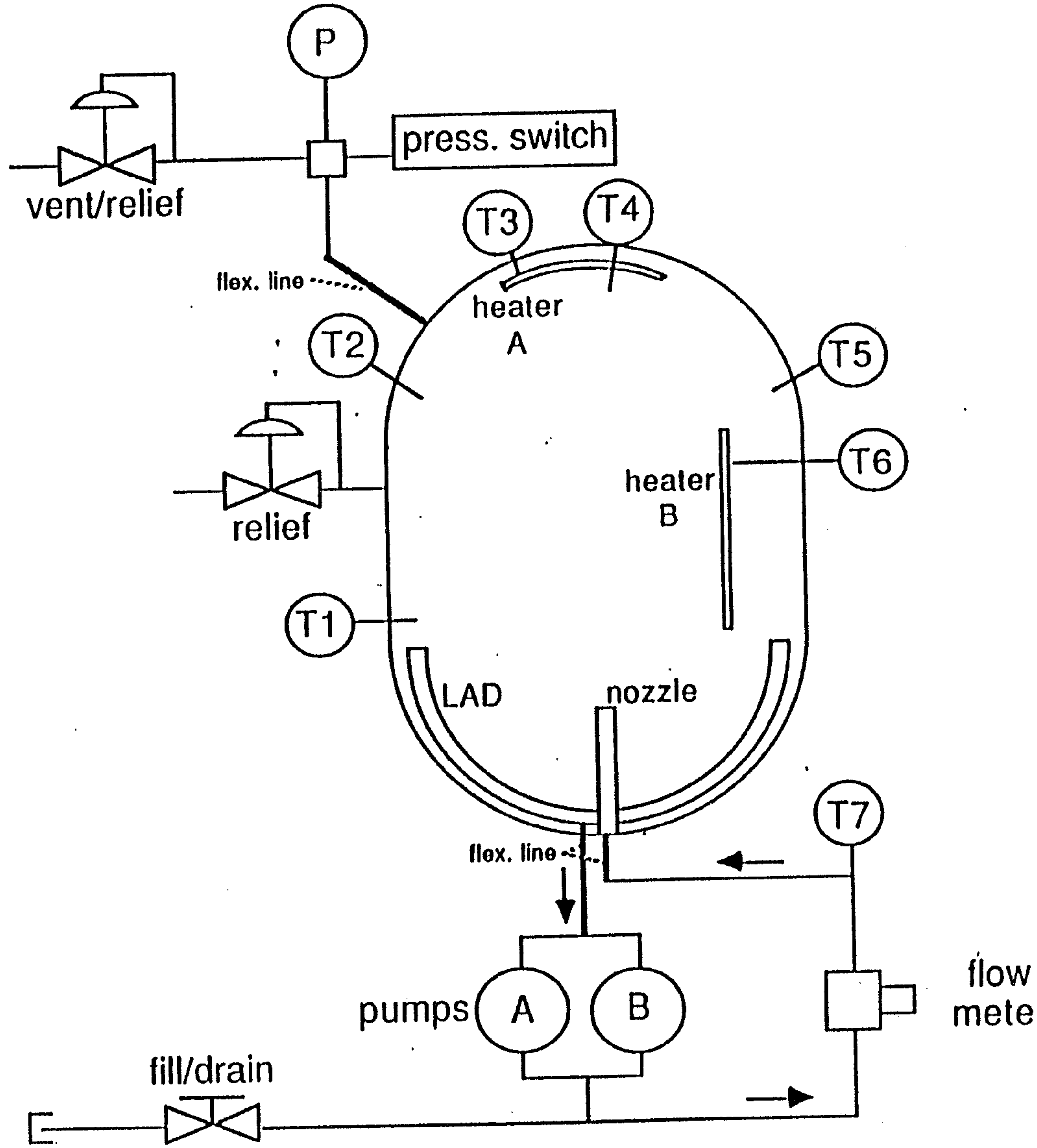
TANK PRESSURE CONTROL EXPERIMENT:THERMAL PHENOMENA

M. M. Hasan and R. H. Knoll
NASA Lewis Research Center
Cleveland, Ohio

Abstract

The "Tank Pressure Control Experiment/Thermal Phenomena (TPCE/TP)" is a reflight of the tank pressure control experiment (TPCE), flown on STS-43 in a standard Get-Away special (GAS) container in August 1991. The TPCE obtained extensive video and digital data of the jet induced mixing process in a partially filled tank in low gravity environments. It also provided limited data on the thermal processes involved. The primary objective of the reflight of TPCE is to investigate experimentally the phenomena of liquid superheating and pool nucleate boiling at very low heat fluxes in a long duration low gravity environment. The findings of this experiment will be of direct relevance to space based subcritical cryogenic fluid system design and operation.

SCHEMATIC OF TANK PRESSURE CONTROL EXPERIMENT (TPCE) HARDWARE



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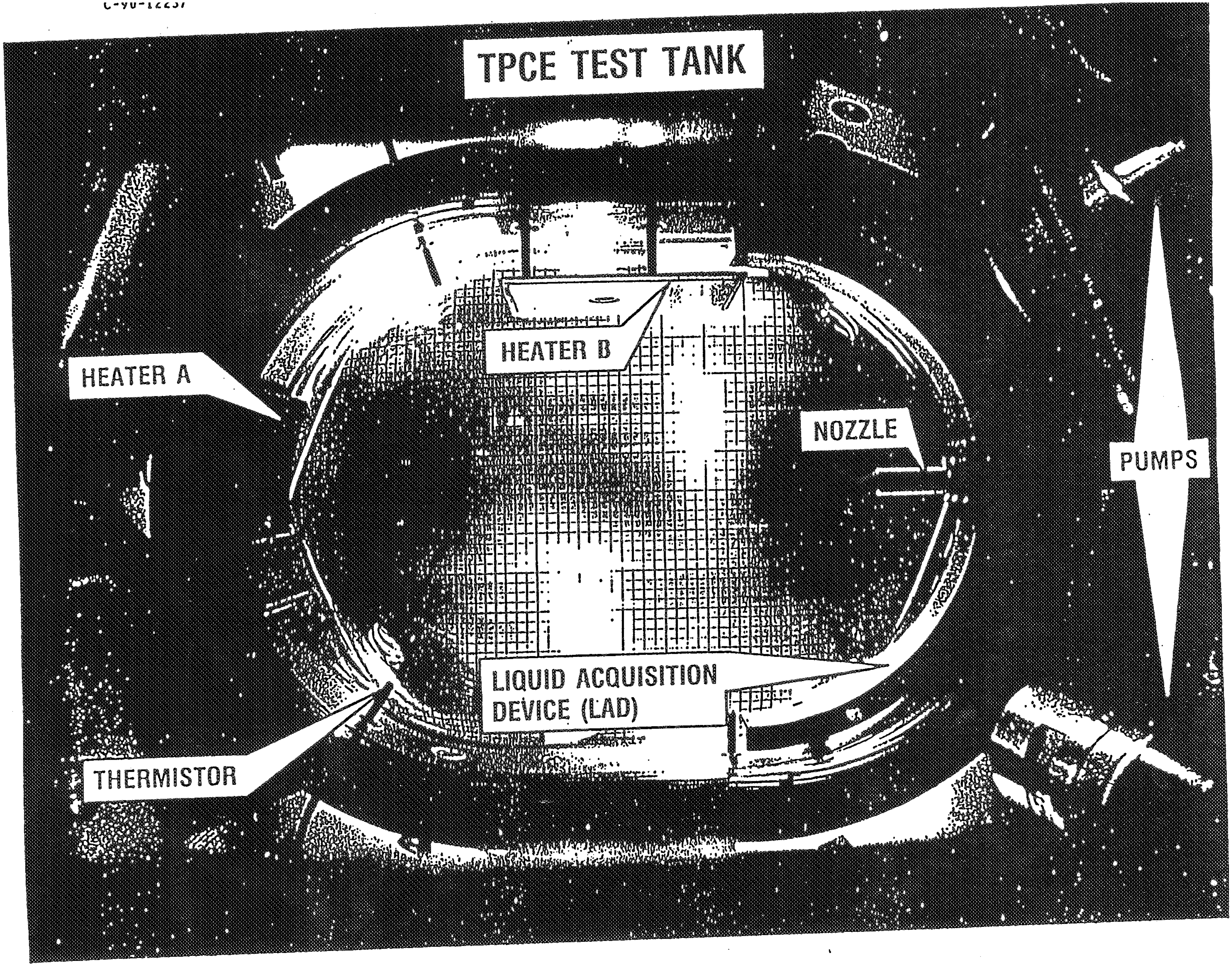
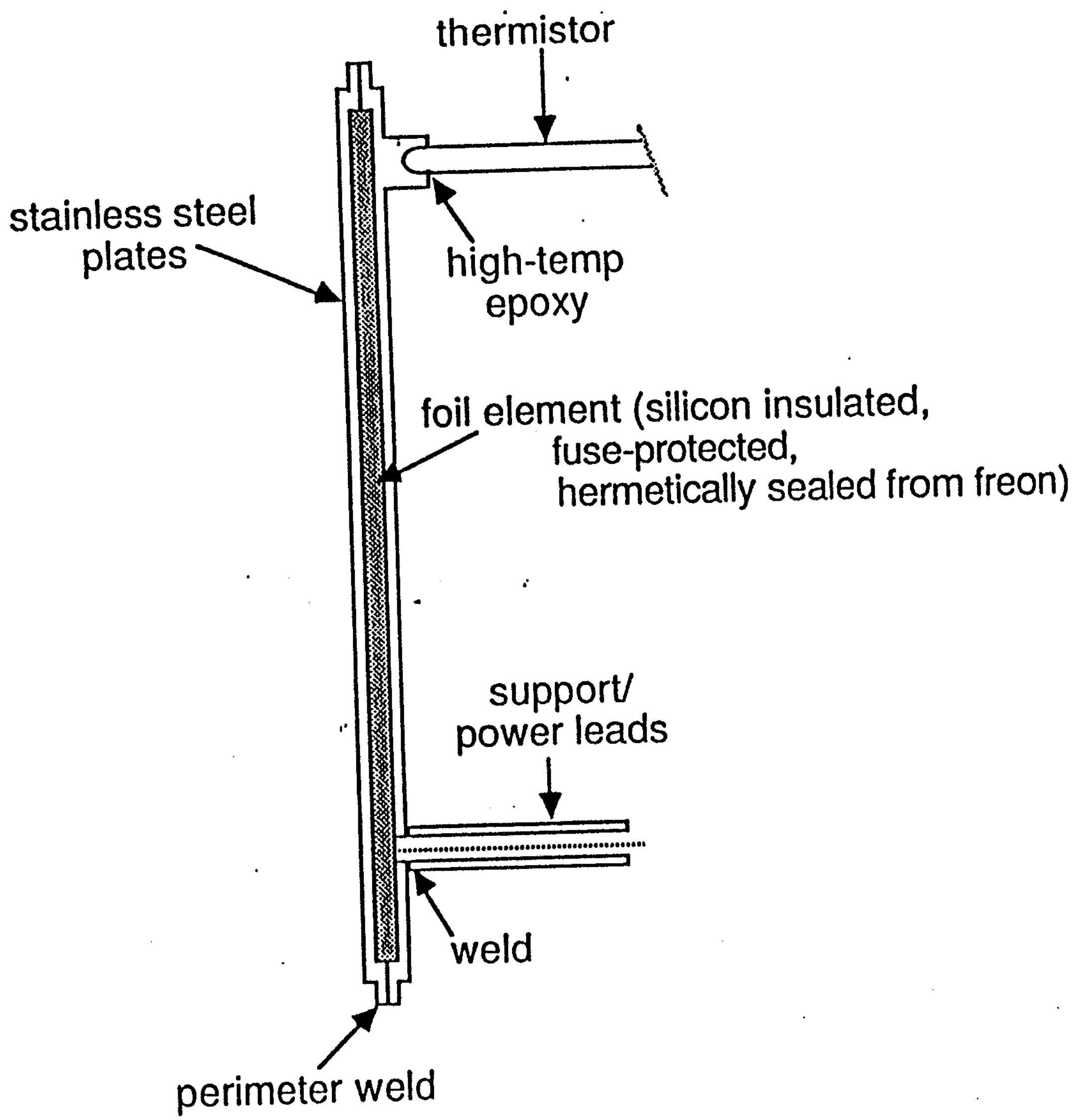

TPCE TEST TANK
HEATER B
HEATER A
NOZZLE
PUMPS
LIQUID ACQUISITION
DEVICE (LAD)
THERMISTOR

HEATER ASSEMBLY



OBSERVATION OF BOILING PROCESS IN LOW-G

- FLUID: FREON 113
- HEATER: FLAT HEATING SURFACE(10.16cm x 6.35cm)
- HEAT FLUX: 0.1 TO 0.15 Watt/cm^2
- LIQUID SUBCOOLING: 3 TO 4^o c
- LIMITED VIDEO OBSERVATION OF BUBBLE INCEPTION, GROWTH AND DEPARTURE
- EXTENDED (10 min.) LOW-G DATA

SOME OBSERVATIONS OF TPCE DATA DURING HEATING PHASE

• FLIGHT DATA SHOW SIGNIFICANT PRESSURE FLUCTUATION

- AN INITIAL PERIOD OF ABOUT 1 TO 2 MINUTES RESULTED IN NO PRESSURE CHANGE, FOLLOWED BY PRESSURE SPIKES OF VARYING MAGNITUTES AND THEREAFTER SOMEWHAT REGULAR PRESSURE FLUCTUATION.

- HEATER SURFACE TEMPERATURE EXCEEDED THE FLUID SATURATION TEMPERATURE BY ABOUT 3 TO 8C. A SIGNIFICANT DIFFERENCE BETWEEN GROUND AND FLIGHT DATA.

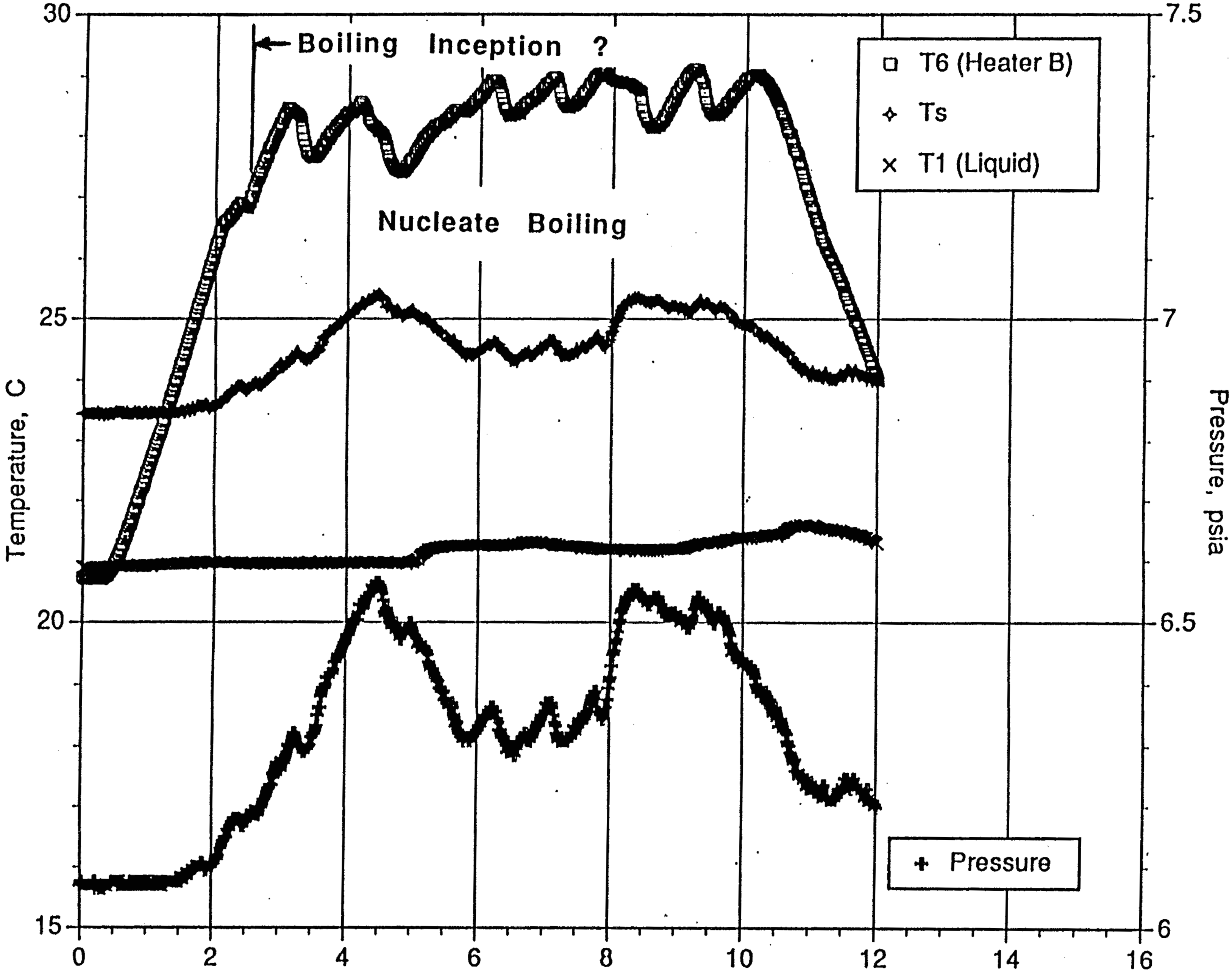
-HEATER TEMPERATURE AND TANK PRESSURE HISTORIES SUGGEST THAT BOILING OCCURRED AND NUCLEATE BOILING CONTINUED DURING THE HEATING PERIOD. VIDEO CAMERA SHOWS SOME LIMITED EVIDENCE OF THIS BUT MISSED INTERESTING PHASE BETWEEN 2 AND 10 MINUTES.

• IN SOME TEST RUNS WITH "HEATER-A" ON THE TANK PRESSURE DID NOT CHANGE FOR ABOUT SIX MINUTES OF HEATING.

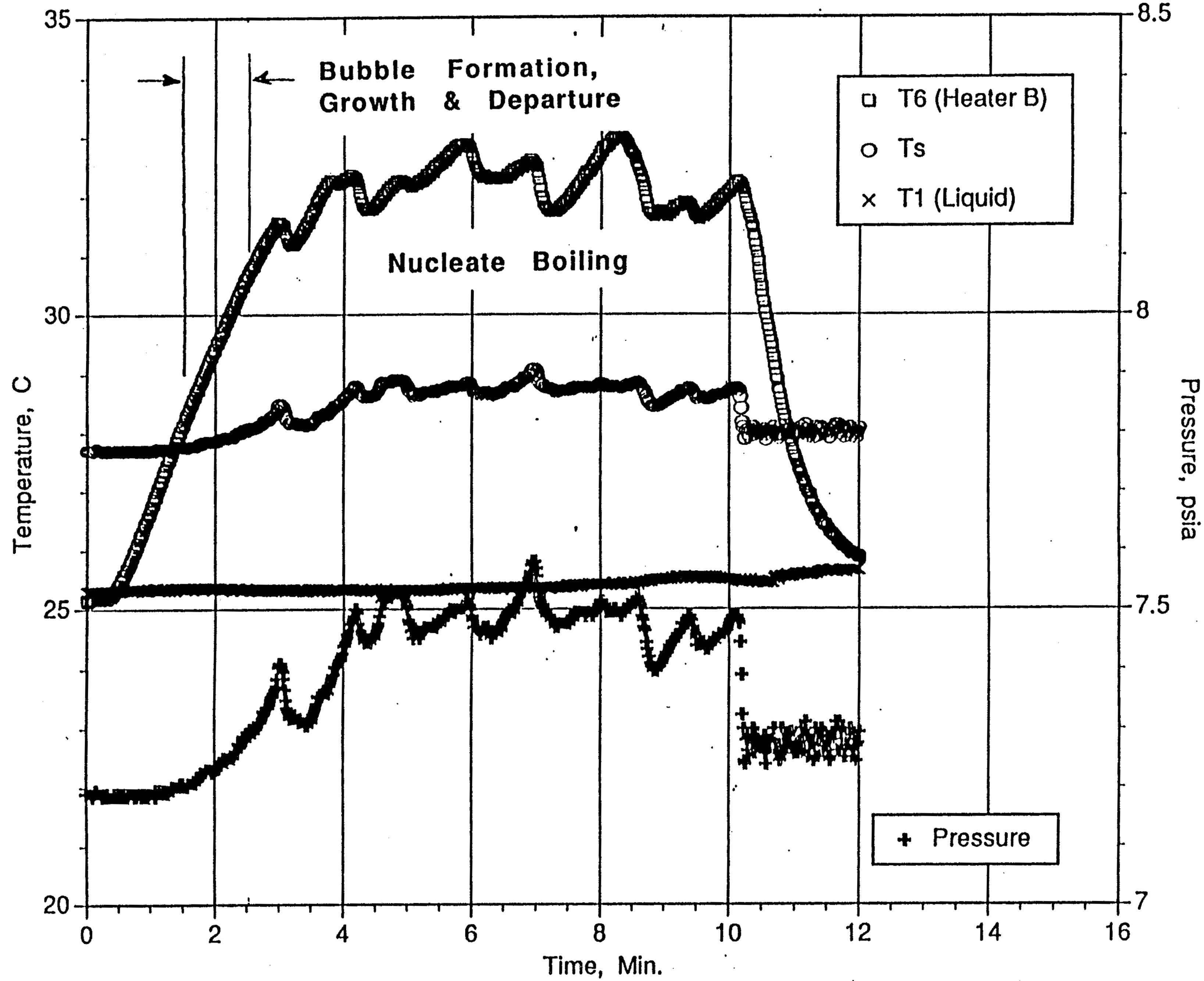
-HEATER SURFACE TEMPERATURE EXCEEDED THE SATURATION TEMPERATURE BY 12 TO 14C.

- THE SUPERHEATING PERIOD WAS FOLLOWED BY EITHER VIOLENT BOILING OR FLASHING OF SUPERHEATED LIQUID. THIS IS EVIDENCED BY PRESSURE SPIKES OF ABOUT 1.4 TO 2.0 PSI IN LESS THAN 0.3 SECOND.

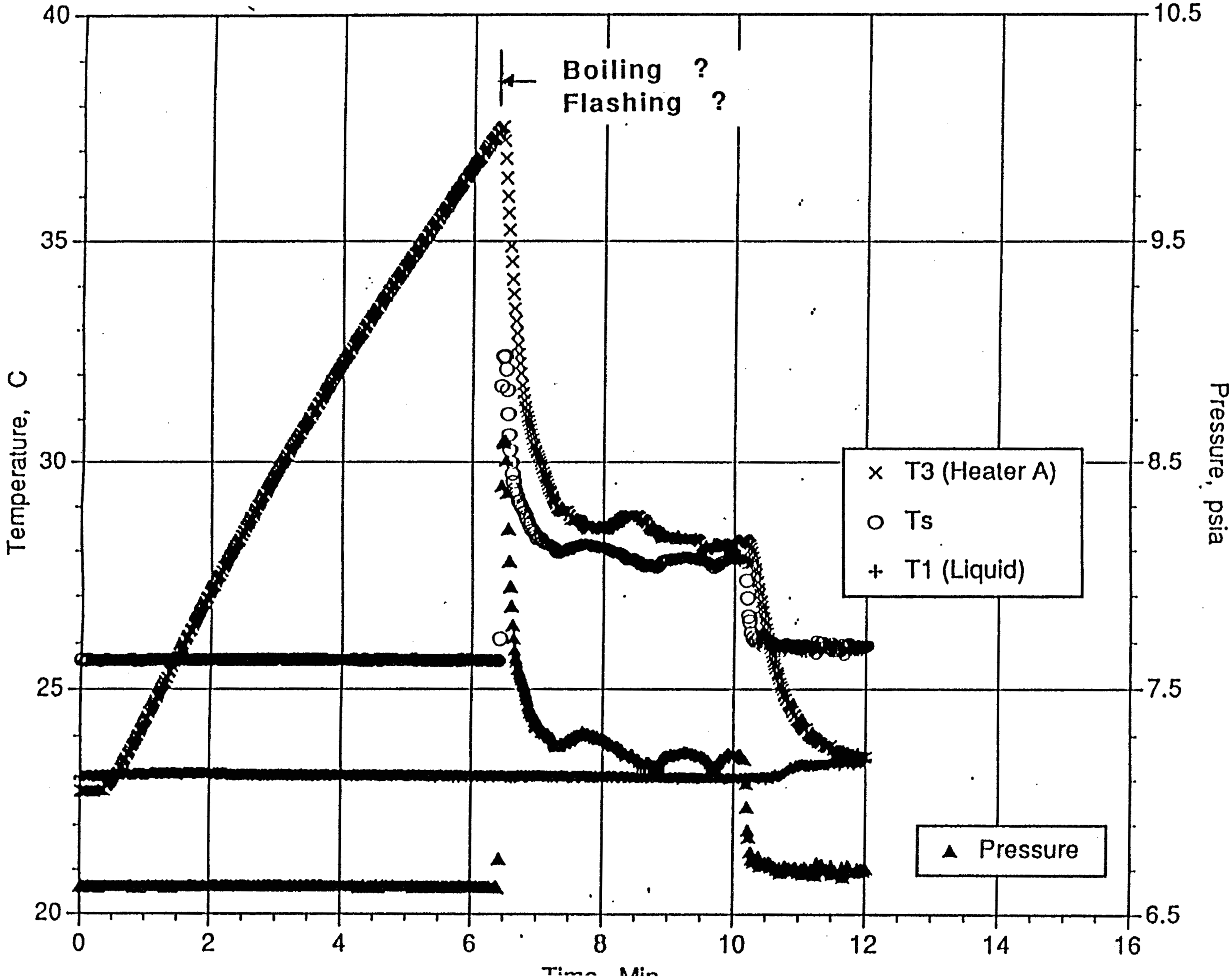
Heater and Liquid Temperatures and Tank Pressure as a Function of Time for
Run No. 03 with Heat Flux Equal to 0.14 watt/cm2



Heater and Liquid Temperatures and Tank Pressure as a Function of
Time for Run No. 37 with Heat Flux Equal to 0.10 watt/cm2



Heater and Liquid Temperatures and Tank Pressure as a Function of Time
for Run No. 13 with Heat Flux Equal to 0.12 watt/cm2



TPCE CONCLUSIONS

- **FLIGHT RESULTS:**
  - VIDEO AND DIGITAL DATA ON JET INDUCED MIXING PROCESS FOR PRESSURE CONTROL, POOL BOILING, AND LIQUID SUPERHEAT PHENOMENA

- **FINDINGS:**
  - FLUID MIXING AND VAPOR CONDENSATION IN LOW-G ARE ACHIEVED WITH CONSIDERABLY LOWER JET FLOW RATE THAN IN 1-G
  - LOW-G DATA MAY LEAD TO METHOD TO EXTRAPOLATE LOW-G PERFORMANCE FROM 1-G DATA
  - PRESSURE SPIKES OCCUR IN LOW-G DUE TO HIGH LIQUID SUPERHEAT AT LOW HEAT FLUX. THIS REPRESENTS A POTENTIALLY HAZARDOUS CONDITION FOR LONG TERM STORAGE OF SUBCRITICAL CRYOGENIC FLUID IN SPACE; FURTHER STUDY/EXPERIMENT IN LONG DURATION LOW GRAVITY ENVIRONMENT IS NEEDED

OBJECTIVES OF REFLIGHT OF TPCE (TPCE/TP)

- FOR A GIVEN HEAT FLUX AND LIQUID SUBCOOLING APPROACHING ZERO
  - OBTAIN EXPERIMENTAL DATA ON MAXIMUM SUSTAINABLE LIQUID SUPERHEAT
  - OBSERVE AND CHARACTERIZE THE LOW-G BOILING PROCESS

- DETERMINE SELF-PRESSURIZATION RATE OF NEARLY SATURATED LIQUID

- DETERMINE PRESSURE DECAY RATE AND CHARACTERIZE DIRECT CONTACT CONDENSATION OF VAPOR FOR LOW JET FLOW RATE MIXING PROCESS
  - DETERMINE QUANTITATIVE CRITERIA OF LOW FLOW RATE

RATIONALE FOR REFLIGHT

- HIGH LIQUID SUPERHEATING OBSERVED AT SUCH LOW HEAT FLUX IS OF PARTICULAR IMPORTANCE TO CRYOGENIC FLUID STORAGE AND TRANSFER. UNDERSTANDING OF LIQUID SUPERHEATING PROCESS WILL HELP TO:

  - ELIMINATE HAZARDOUS CONDITION DURING STORAGE

  - DEVELOP ALTERNATE METHOD FOR TANK PRESSURIZATION

- A VISUAL OBSERVATION OF BOILING AND LIQUID SUPERHEATING PHENOMENA COUPLED WITH TEMPERATURE AND PRESSURE MEASUREMENT WILL PROVIDE USEFUL INFORMATION TO MODEL THESE PROCESSES

EXPERIMENTS WITH EXTENDED HEATING PERIOD AND COMPLETE VIDEO COVERAGE ARE REQUIRED

RATIONALE FOR REFLIGHT (CONTINUED)

- TPCE USED NONCONDENSIBLE PRESSURANT (HELIUM) TO AVOID PUMP CAVITATION PROBLEM

  - RESULTED IN LIQUID SUBCOOLING OF ABOUT 5 ^{o}C

  - ANTICIPATED TO HAVE NEGLIGIBLE EFFECT ON CONDENSATION RATE OF VAPOR; REFLIGHT WILL VERIFY

- SUBCRITICAL CRYOGENIC FLUID IN SPACE ENVIRONMENT WILL ALWAYS BE AT NEARLY SATURATED CONDITION; HOWEVER, EXTRAPOLATION OF TPCE RESULTS TO NEARLY SATURATION CONDITION IS NOT STRAIGHTFORWARD AND OBVIOUS

EXPERIMENTAL DATA WITH NO NONCONDENSIBLES IN THE ULLAGE AND LIQUID SUBCOOLING APPROACHING ZERO ARE REQUIRED

PROPOSED EXPERIMENT

- FLUID: FREON 113
- LIQUID SUBCOOLING: LESS THAN 2 ^{o}C
- HEATING TIME: 10 TO 40 MINUTES
- LIQUID FILL LEVEL: 85 PERCENT SAME TPCE
- VIDEO: DURING THE ENTIRE HEATING PHASE FOR SELECTED TESTS
- JET FLOW RATE: IN THE RANGE OF 0.0 TO 0.5 GPM
- SOME MIXING TESTS WITH NO HEATING
- MEASUREMENTS, ACCURACY AND DATA RATE: SAME AS TPCE

A TEST MATRIX INCORPORATING THE ABOVE CHANGES AND WITHIN THE CONSTRAINTS OF THE SAME TOTAL POWER DISSIPATION AND TOTAL VIDEO TIME IS SHOWN IN TABLE 1

TABLE 1

TEST MATRIX (TPCE/TP)

RUN NO.	HEATER	HEATING TIME:MIN	CAMERA	VIDEO TIME:MIN	JET FLOW RATE, GPM	MIXING TIME MIN	SETTLING TIME:MIN	PRINCIPAL OBJECTIVE/FOCUS
1	OFF	0.0	A	12.0	0-0.5	6	20	FLOW TRANSITION CRITERIA
2	B	10.0	A	12.0	0.1	10	20	BOILING, VAPOR CONDENSATION, MIXING, STRATIFICATION
3	B	10.0	A	12.0	0.2	10	20	
4	B	10.0	A	12.0	0.3	10	20	
5	OFF	0.0	B	12.0	0-0.5	6	20	FLOW TRANSITION CRITERIA
6	A	10.0	B	12.0	0.1	10	20	LIQUID SUPERHEATING, PRESSURE SPIKES, BOILING, CONDENSATION
7	A	10.0	B	12.0	0.2	10	20	
8	A	10.0	B	12.0	0.3	10	20	
9	OFF	0.0	A	12.0	0-0.5	6	20	FLOW TRANSITION CRITERIA
10	B	18.0	A	20.0	0.2	10	20	SAME AS 2 TO 4 FOR LONGER HEATING TIME
11	B	18.0	A	20.0	0.0	10	20	
12	A	18.0	B	20.0	0.3	10	20	SAME AS 6 TO 8, EFFECT OF SUB-COOLING ON VAPOR COND. RATE
13	A	18.0	B	20.0	0.1	10	20	
14	OFF	0.0	B	12.0	0-0.5	6	20	FLOW TRANSITION CRITERIA
15	A&B	18.0	A	20.0	0.4	10	20	BOILING AT LOWER HEAT FLUX
16	A&B	18.0	B	20.0	0.4	10	20	
17	A	40.0	OFF	---	0.4	30	20	HEATER ON DURING MIXING: HOMOGENOUS PRESSURE RISE
18	B	40.0	OFF	---	0.4	30	20	
19	A&B	40.0	OFF	---	0.4	10	20	LONGER HEATING WITH NO VIDEO
20	B	40.0	OFF	---	0.2	10	20	
21	A	40.0	OFF	---	0.3	10	20	

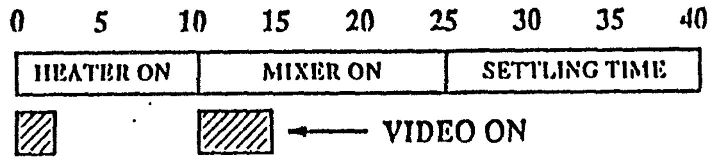
COMPARISONS OF TPCE & TPCE/TP TEST MATRIX'S & TIMELINES

TEST MATRIX

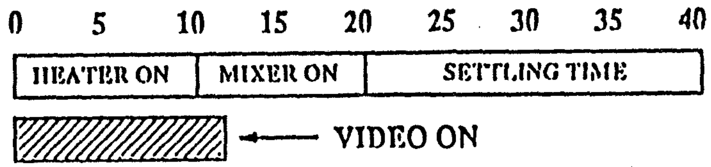
	TPCE	TPCE/TP
NUMBER OF TESTS	38	21
HEATING TIME	10 MIN	10-40 MIN
VIDEO DURING HEATING	1.75 MIN	10-18 MIN
MIXING RATE	0-1 GPM	0.1-0.4 GPM
FLOW TRANSITION TESTS	——	4 TESTS VARYING FLOW FROM 0-0.5 GPM IN 0.05 GPM INCREMENTS

TYPICAL TIMELINE

TPCE



TPCE/TP



SUMMARY

•FLIGHT RESULTS INCLUDE VIDEO AND DIGITAL DATA ON :

- THE JET INDUCED MIXING PROCESS FOR PRESSURE CONTROL

- THE POOL BOILING AND SUPERHEATING OF NEARLY SATURATED LIQUID SUBJECTED TO VERY LOW HEAT FLUXES

•TPCE/TP RESULTS ALONG WITH THE TPCE DATA WILL BE USED TO:

- CHARACTERIZE DIRECT CONTACT CONDESATION OF VAPOR FOR LOW JET FLOW RATE MIXING PROCESS

- CHARACTERIZE SOME ASPECTS OF POOL BOILING AND LIQUID SUPERHEATING

- ATTEMPT TO EXPLAIN THE OCCURANCE OF PRESSURE SPIKES (TPCE RESULTS) AND IDENTIFY LIMITING DESIGN CRITERIA(IF ANY) FOR A SUBCRITICAL CRYOGENIC FLUID STORAGE SYSTEM

- DEVELOP METHOD TO EXTRAPOLATE NORMAL GRAVITY RESULTS TO LOW GRAVITY CONDITION

ESEX

ELECTRIC PROPULSION SPACE EXPERIMENT

OBJECTIVE: DEMONSTRATE READINESS AND COMPATIBILITY OF NEAR TERM, HIGH POWER ARCJET PROPULSION

Answer operational challenges for users.

Give researchers a data point to correlate ground test data.

ATTD/ESEX PROGRAM GOALS

DEVELOP FLIGHT WEIGHT HARDWARE

INTEGRATE EXISTING SUBSYSTEMS

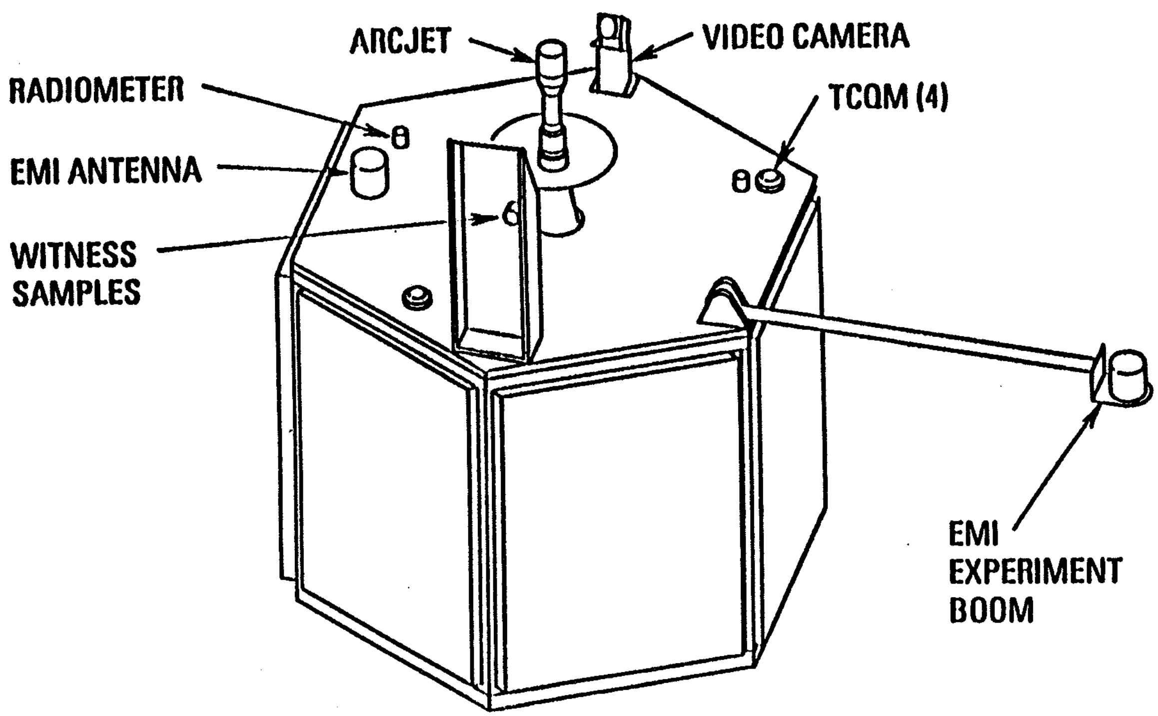
DEMONSTRATE ARCJET MATURITY

MEASURE INTEGRATED PERFORMANCE

ARCJET THRUSTER/SPACECRAFT COMPATIBILITY

- ZERO-G PLUME CONTAMINATION
- FREE SPACE ELECTROMAGNETIC INTERFERENCE
- THERMAL RADIATION EFFECTS

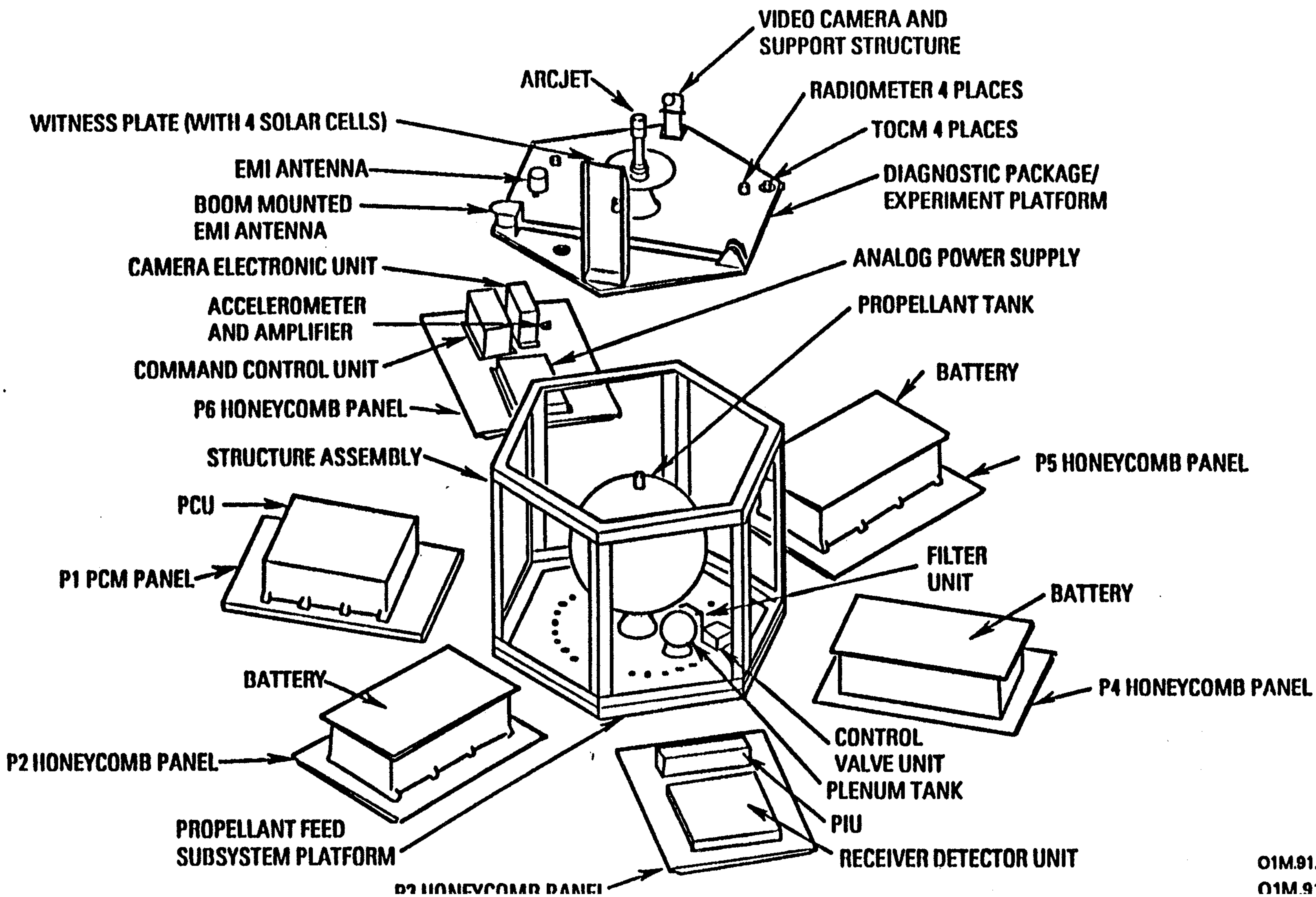
Initial Layout of Diagnostics Package Instruments



ATTD

System Design
Design Description

Arcjet ATTD Equipment



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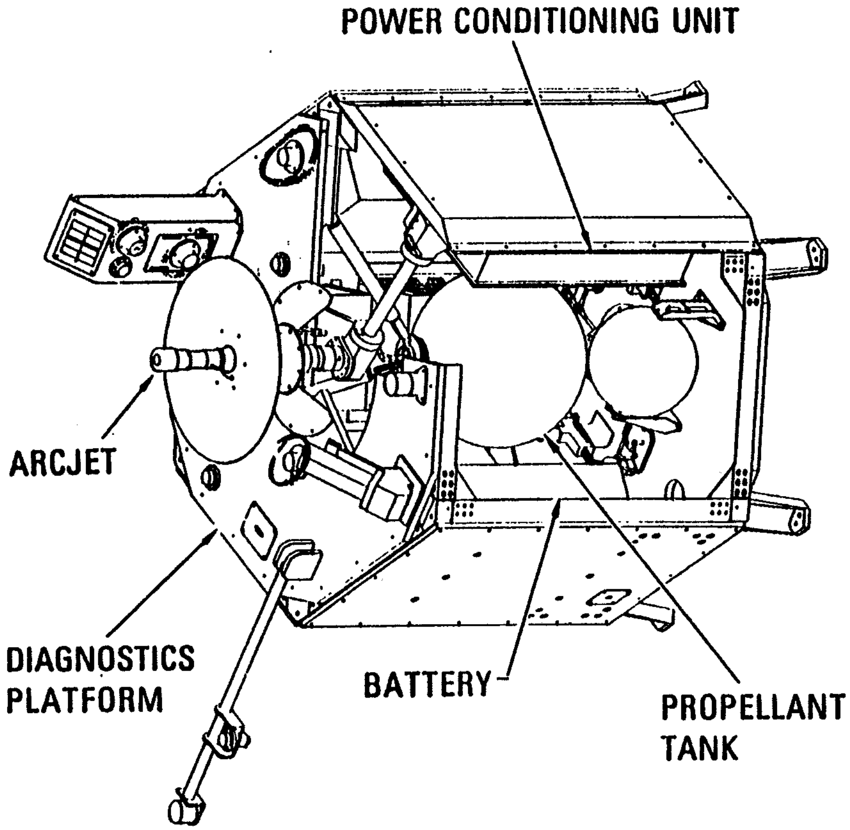
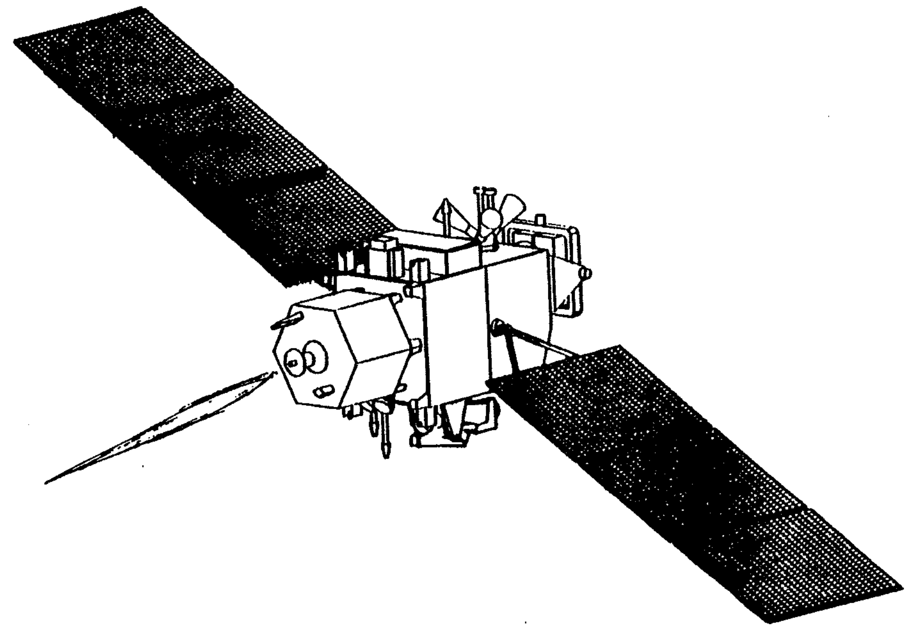
ATTD

Deployed Flight Configuration



ESEX



P91-1 ARGOS

01M.92.093.01-2

ATTD

Diagnostic Package

Requirements Versus Capabilities

Requirement	Source	Capability	Verification Method
Measure thruster performance	SOW 3.1.6		
Thrust		Accelerometer	Computed from measurement and mass
Mass flow		Mass flow meter } RRC	Measured
Thruster voltage and current		Voltmeter, ammeter } RRC	Measured
Measure plume contamination (function of radius and angle)	SOW 3.1.6	TQCM	Measurements at four positions
Determine plume effects on	SOW 3.1.6		
Solar arrays		Solar array witness rate	Measurements
Typical space vehicle structures			Provide flow field for further analysis and test
Refractory metals			
Th mal radiation flux (fun tion of radius and angle)	SOW 3.1.6	Radiometers	Measurements at four positions
Ele tromagnetic interference (function of radius and angle)	SOW 3.1.6	Radiated EMI experiment	Measurements at two pos ions
Document nozzle and plume luminescence	SOW 3.1.6	Video camera	Observation
Relate collected data to expected results at 30 m	SOW 3.1.6		Analysis

/002

ATTD/ESEX STATUS

ATTD CRITICAL DESIGN REVIEW - FIRST QUARTER FY93

INTEGRATION WITH P91-1 (ARGOS)*

LAUNCH - FIRST QUARTER FY96

* ARGOS - Advanced Research Global Observation Satellite

PRELIMINARY FLIGHT OPERATIONS

START AT 460 NM

10 FIRINGS - 15 MINUTES EACH

100 HRS BETWEEN FIRINGS

1/2 OF FIRINGS OVER AIR FORCE MAUI OPTICAL SITE (AMOS)

FINAL ALTITUDE 450 NM

DEVELOPING AN INFLATABLE SOLAR ARRAY

Patrick K. Malone, L'Garde, Inc.

Dr. Francis J. Jankowski, USAF Phillips Laboratory/VTPN

Geoffery T. Williams, L'Garde, Inc.

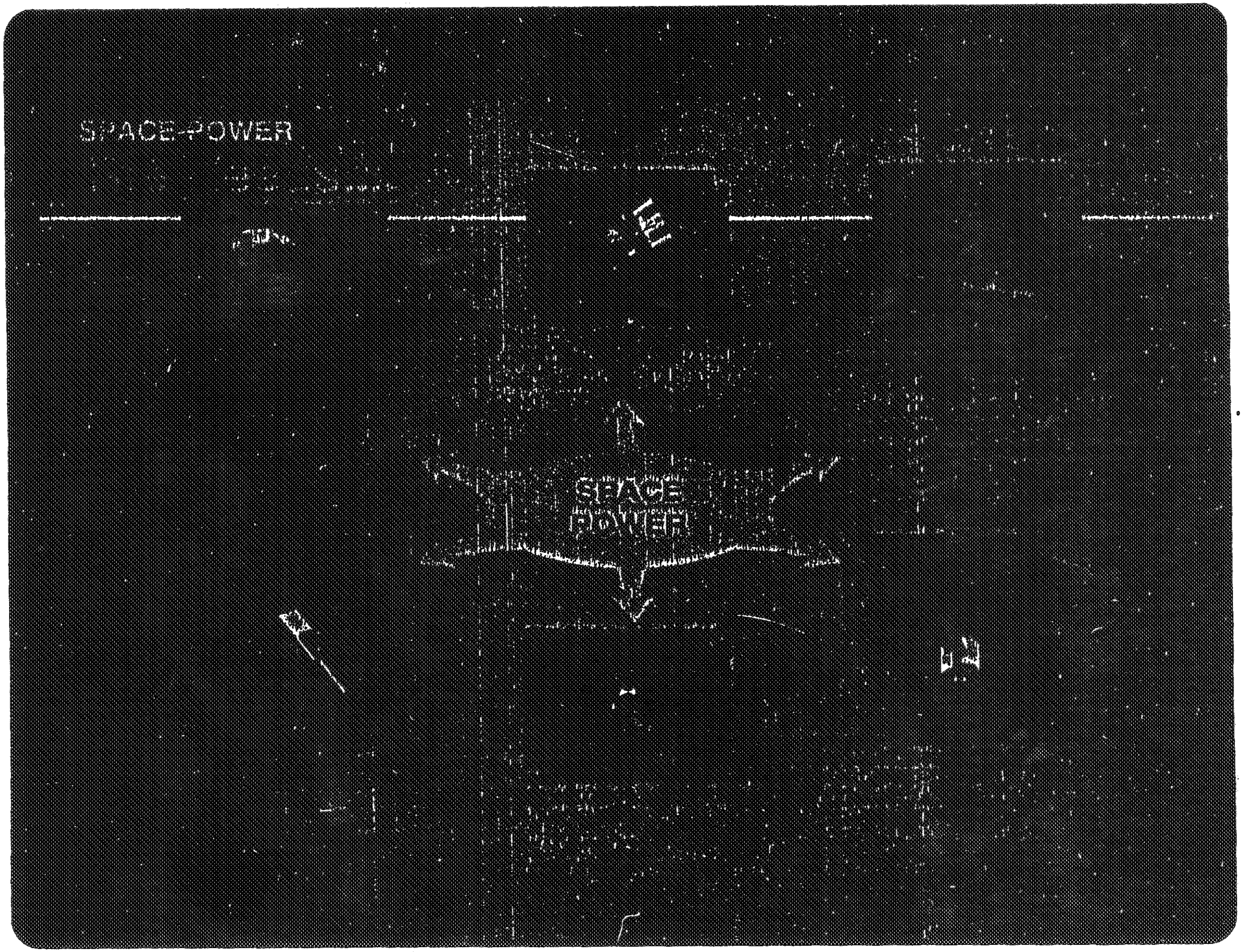
Dr. George J. Vendura, Jr., SUMM Associates

Presented By: Patrick K. Malone
October 6, 1992
NASA/DOD Flight Experiments Interchange Meeting

DEVELOPING AN INFLATABLE SOLAR ARRAY

ITSAT Design Goals

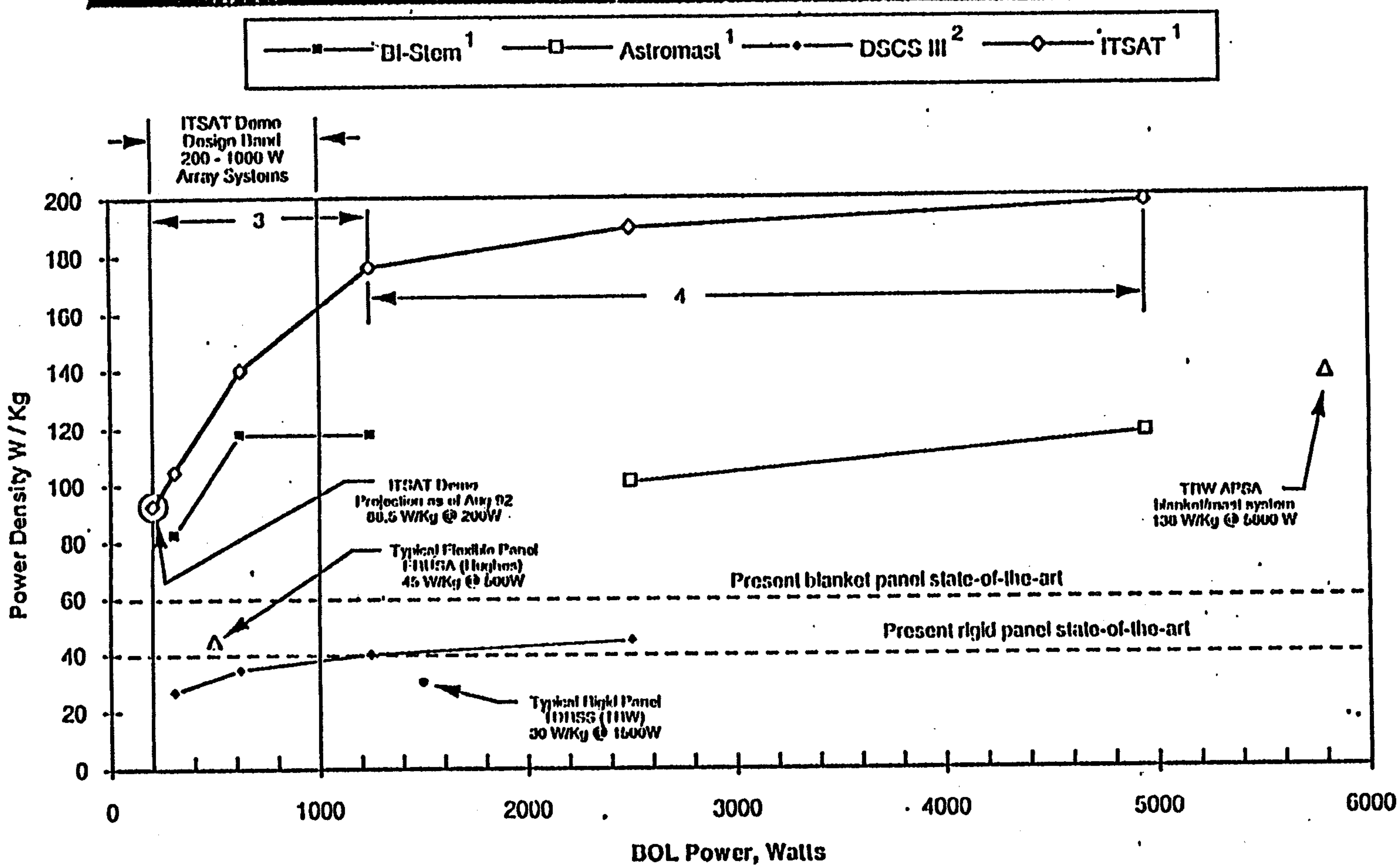
- 90 - 100 W/Kg Array System (200 W Wing)
- Design Orbit: 600 - 800 Km (Worst Case Inclination)
- 3 Year Life (200 W EOL)
- Scalability of design to 1000 Watt Wing
- Low Recurring Costs


SPACE-POWER
SPACE
POWER

DEPLOYMENT COMPARISON

ITSAT vs Bi-Stem vs Astromast DSCS III

(BOL/GEO/Crystal-Si)



1. L'Garde ITSAT with crystalline silicon cells on flexible blanket.
2. Rigid panel design.
3. ITSAT based on alimunum laminate.
4. ITSAT based on U-V cured resin structure.

VU-92-GW-022b

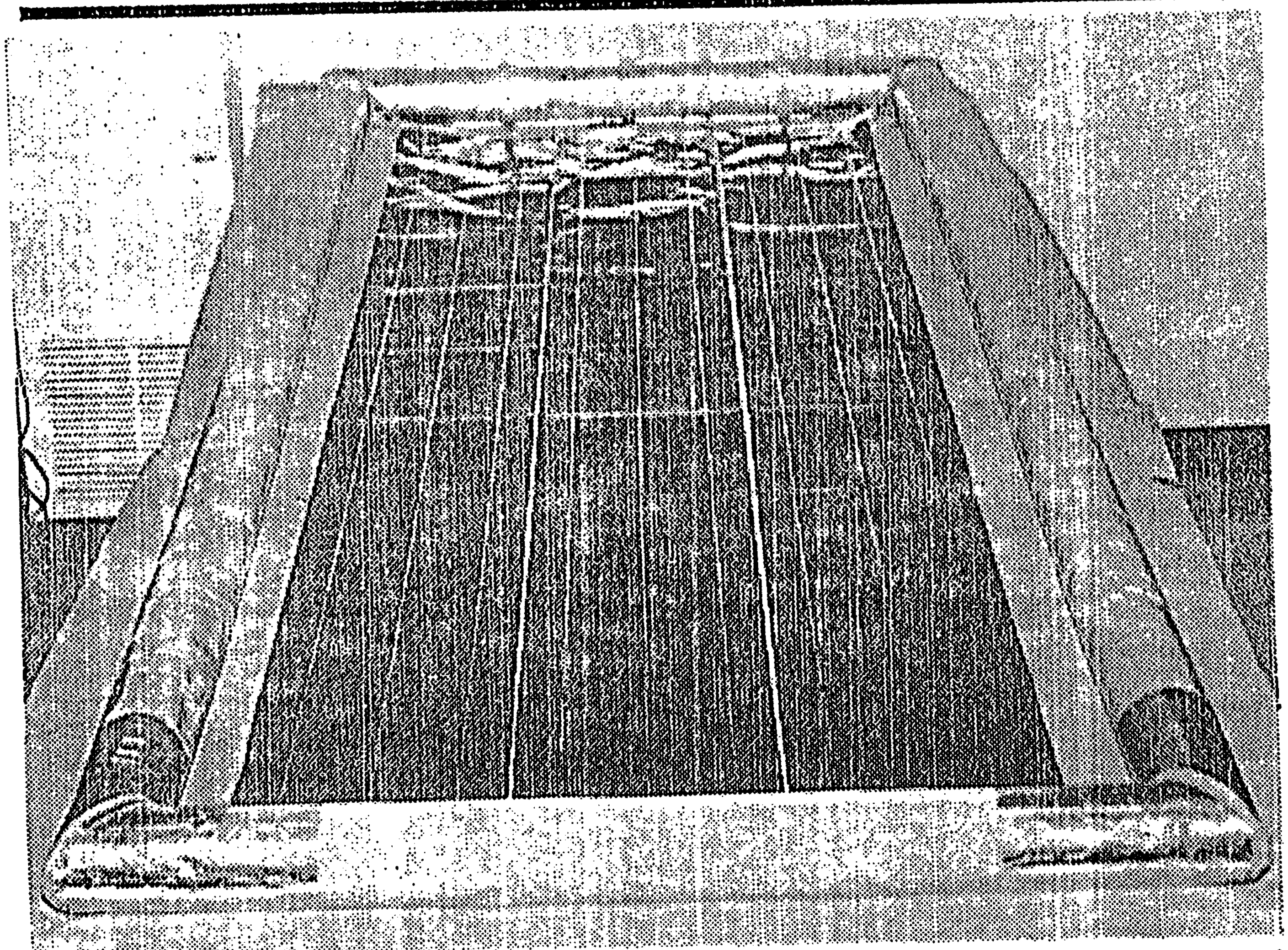
DEVELOPING AN INFLATABLE SOLAR ARRAY

Program Phases

- Three Phases
- Phase 1 Feasibility and Proof of Concept
- Phase 2 Update Ph 1 Design and Fabricate a Flight Qualified System
- Phase 3 Refurbish Ph 2 System and Conduct a flight test

(Program status: approximately mid-term of phase 2)

PROTO TYPE UNIT



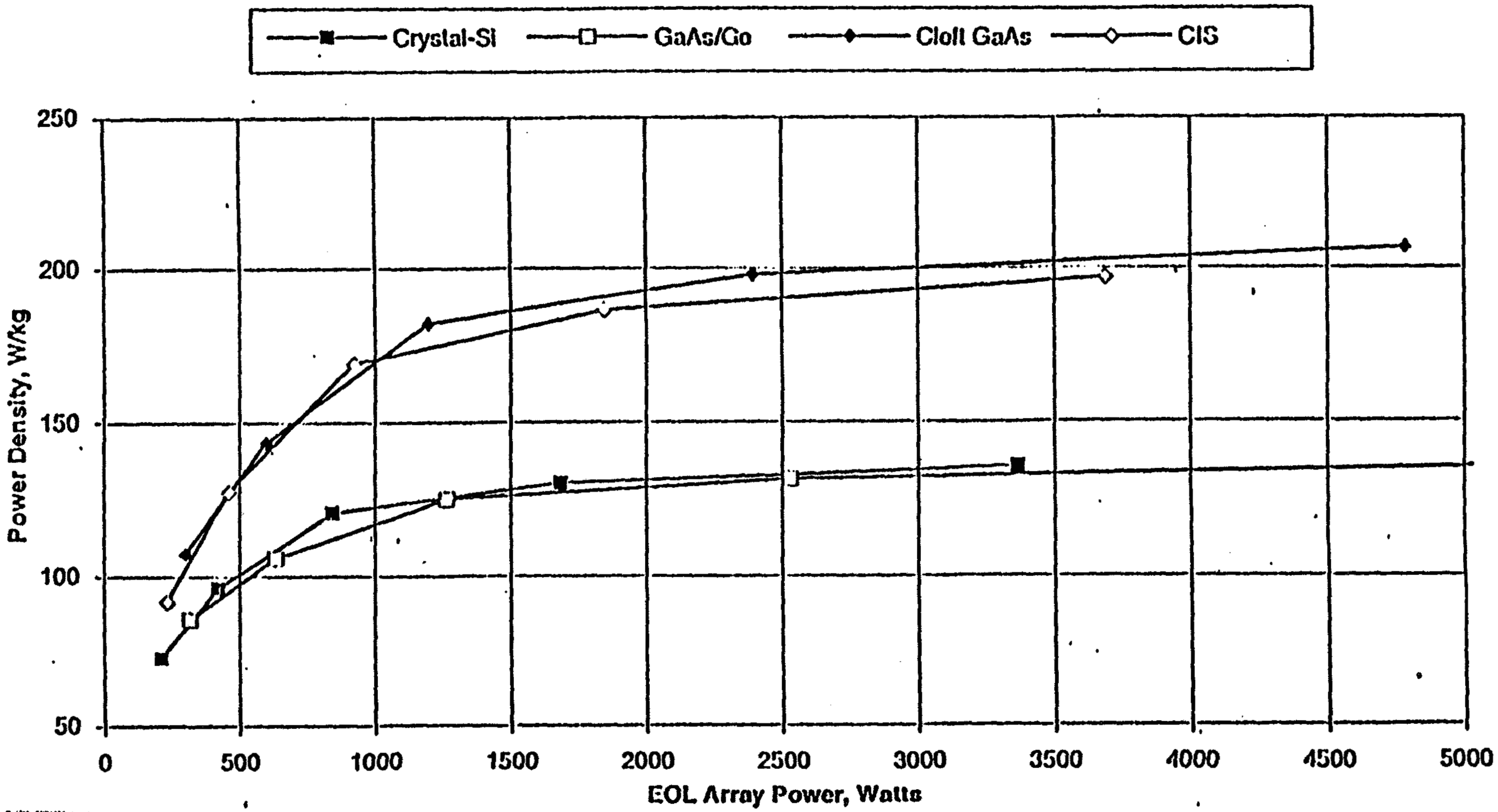
VU-92-GW-022e

C-4

EOL POWER DENSITY vs DEPLOYED ARRAY POWER

APSA-Type Flexible Blanket/Torus Deployed

3 Year LEO



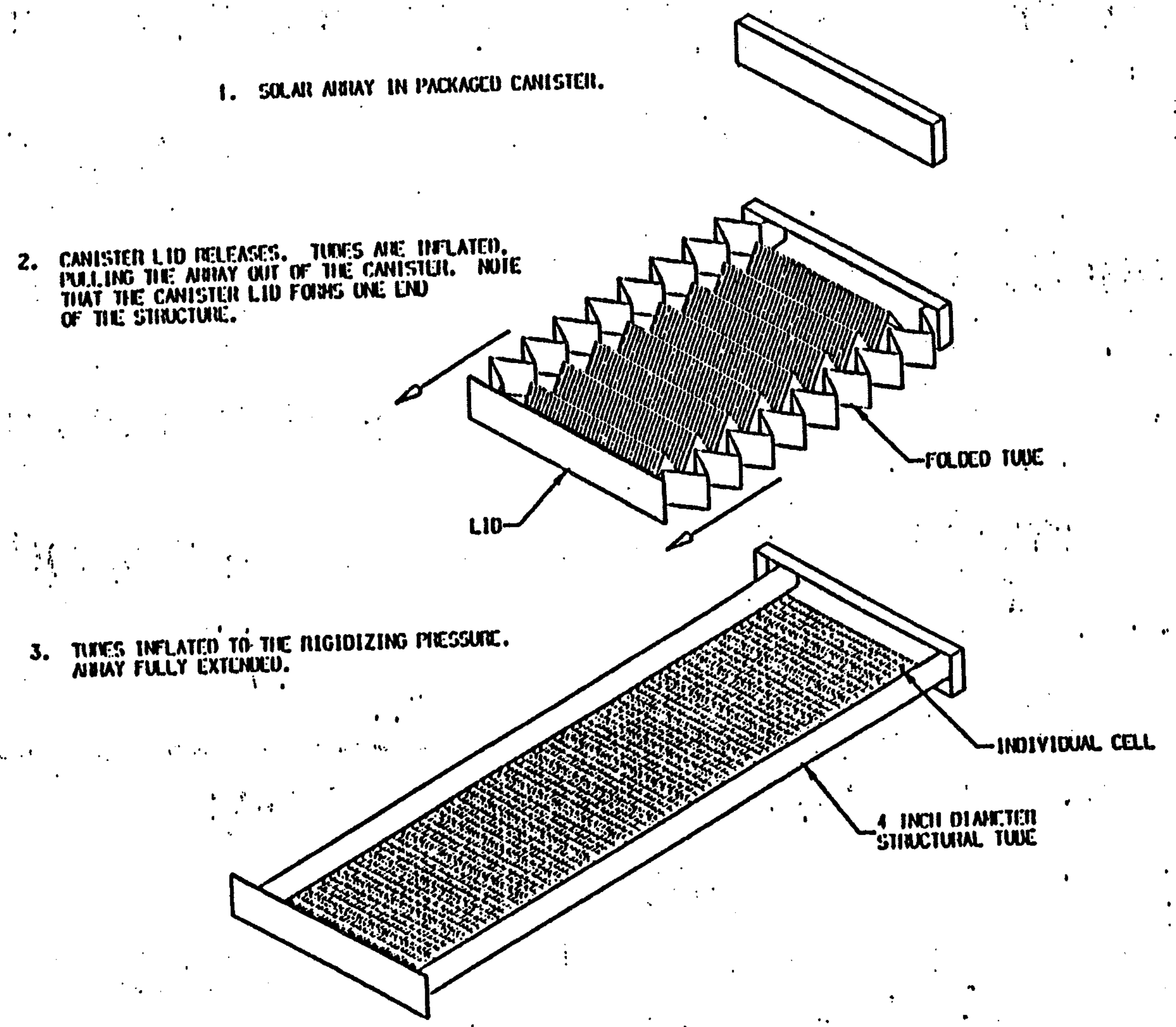
VU-92-GW-022c

DEVELOPING AN INFLATABLE SOLAR ARRAY

Major Sub-Systems

- Housing/Cover
- Tube Booms
- Inflation System
- Solar Blanket Assembly

ITSAT SOLAR ARRAY
PH2 Design



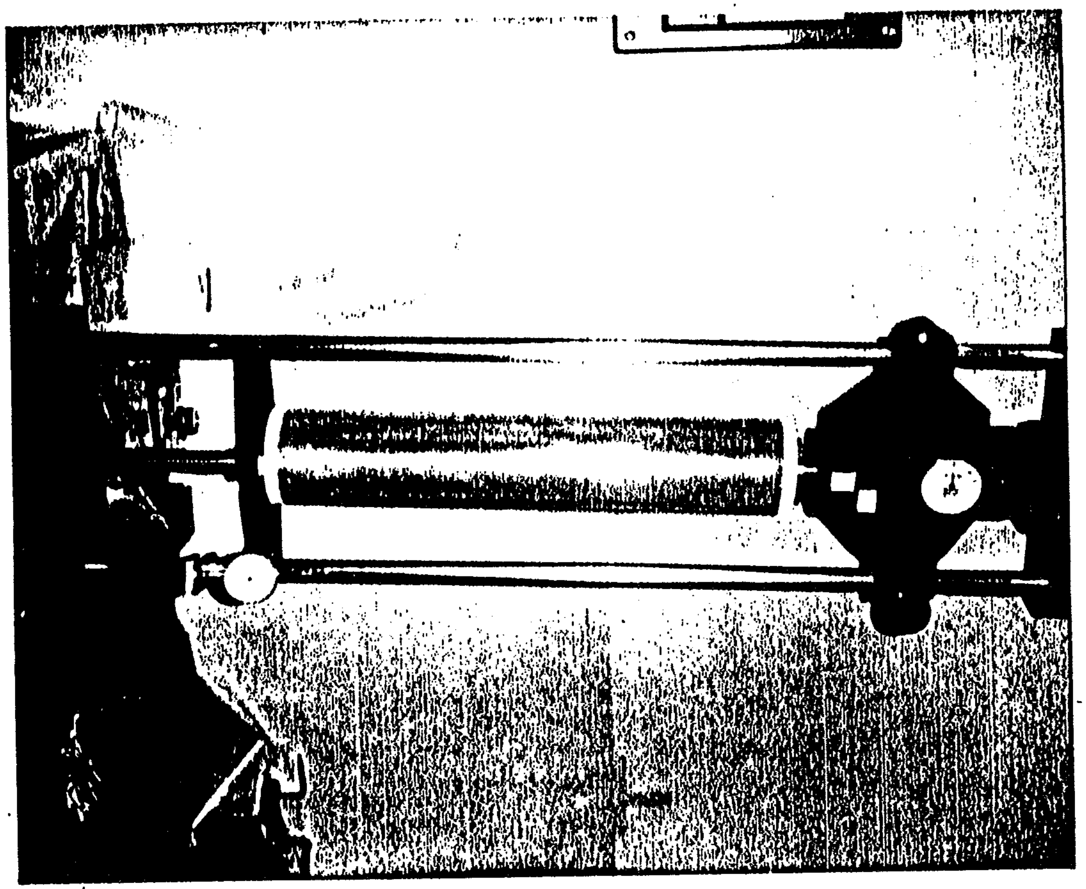
VU-92-GW-022g

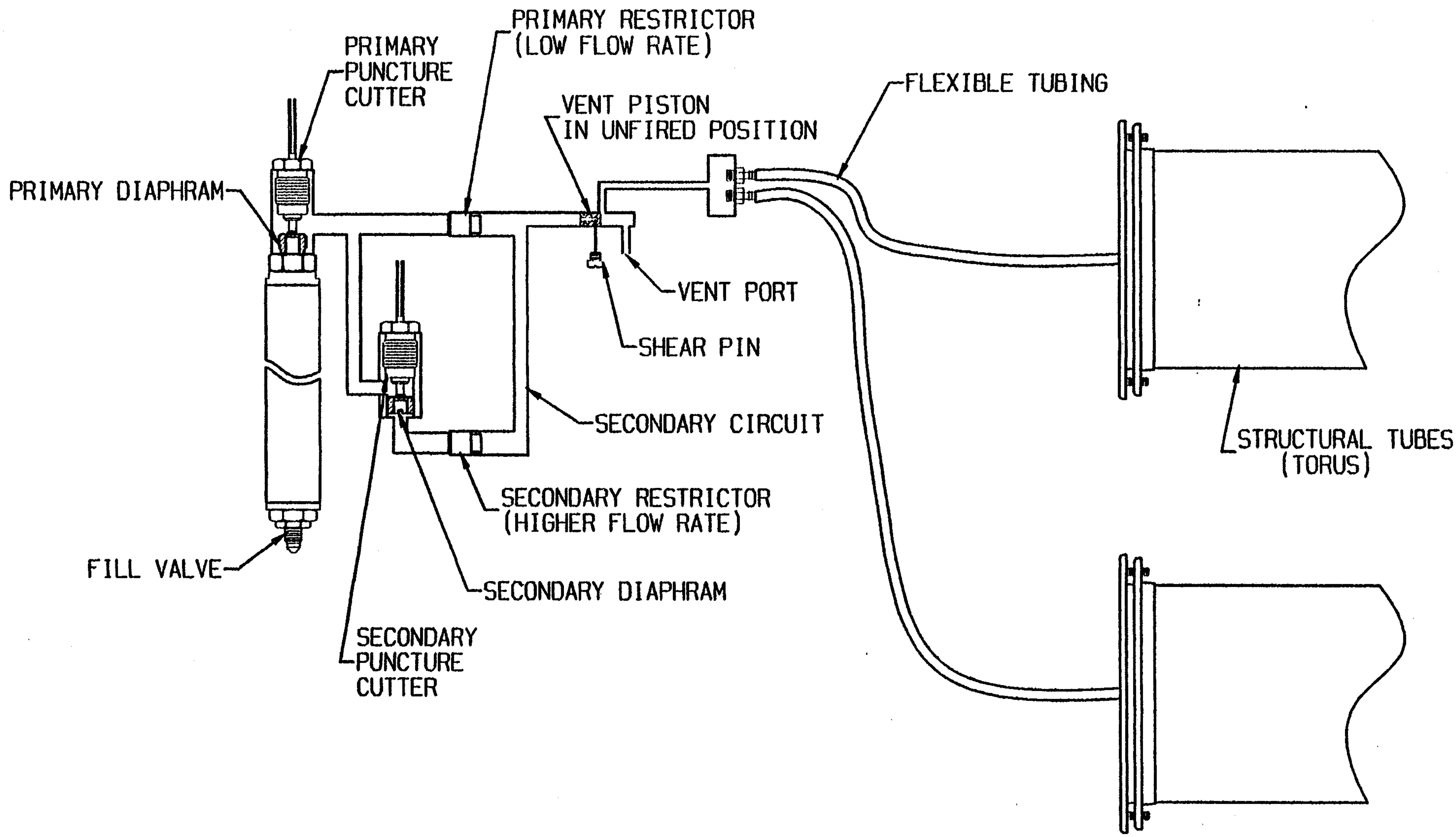
DEVELOPING AN INFLATABLE SOLAR ARRAY

Rigidizable TubeDesign

- 3-ply Kapton-Aluminum-Kapton Laminate
- 4.0 inch Diameter
- Compression Loading Requirements 12 lbs
- Rigidization Pressure 22 psi
- Current Design Margin is Approximately 2.5

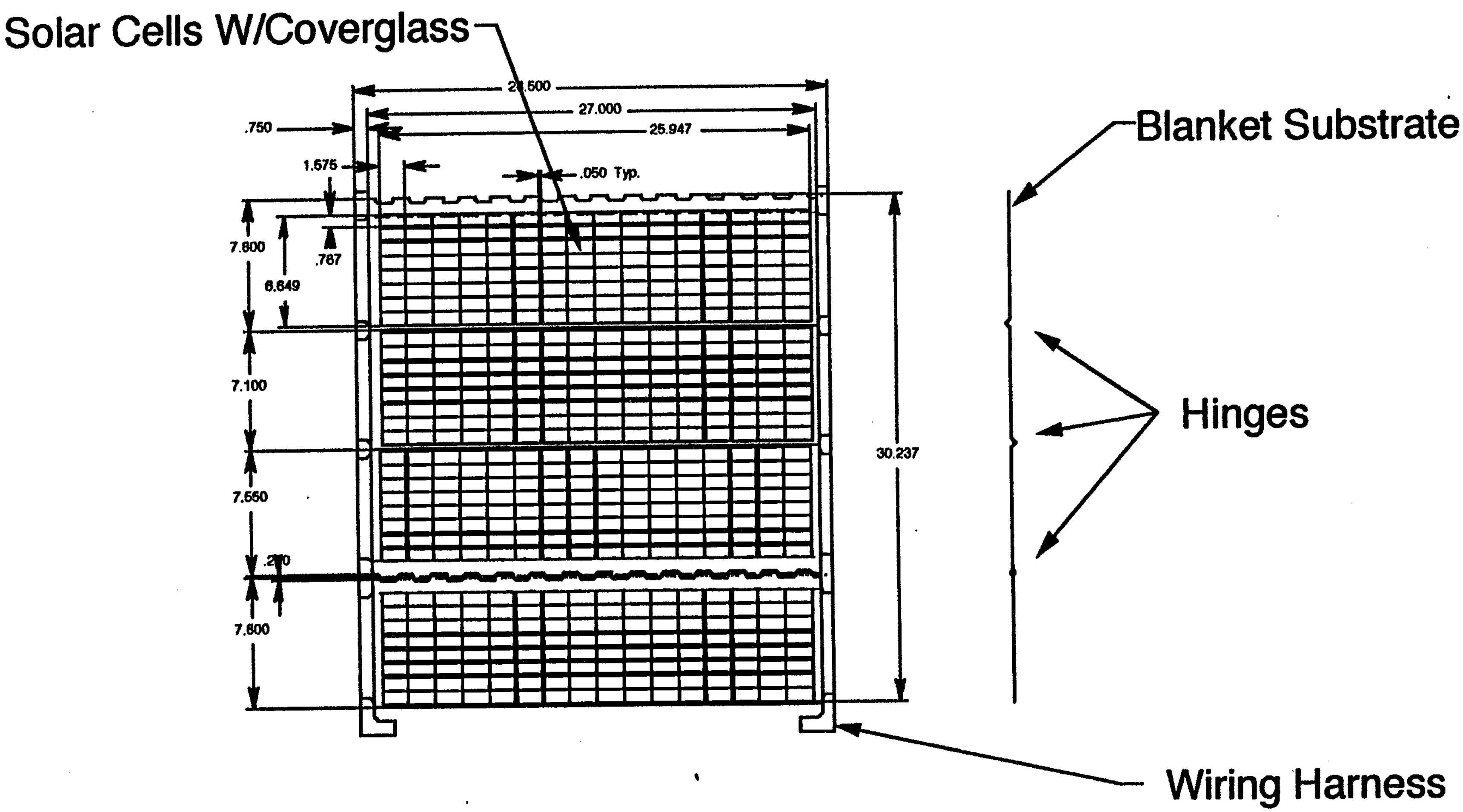
This data is from actual test results




PRIMARY PUNCTURE CUTTER
PRIMARY RESTRICTOR (LOW FLOW RATE)
VENT PISTON IN UNFIRED POSITION
FLEXIBLE TUBING
PRIMARY DIAPHRAM
VENT PORT
SHEAR PIN
SECONDARY CIRCUIT
STRUCTURAL TUBES (TORUS)
SECONDARY RESTRICTOR (HIGHER FLOW RATE)
FILL VALVE
SECONDARY DIAPHRAM
SECONDARY PUNCTURE CUTTER

ITSAT SOLAR ARRAY

Typical Blanket Segment



DEVELOPING AN INFLATABLE SOLAR ARRAY

Program Summary

- Current W/Kg estimates are well above State-of -the-art
- A variety of cell types can be used
- Modular design to adapt to a variety of satellites
- Recurring costs are anticipated to be low

N93-28715

SODIUM-SULFUR CELL TECHNOLOGY FLIGHT EXPERIMENT (SSCT)

516-44
159218
p-8

Carl R. Halbach, Principal Investigator
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Space Systems/LORAL
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Introduction

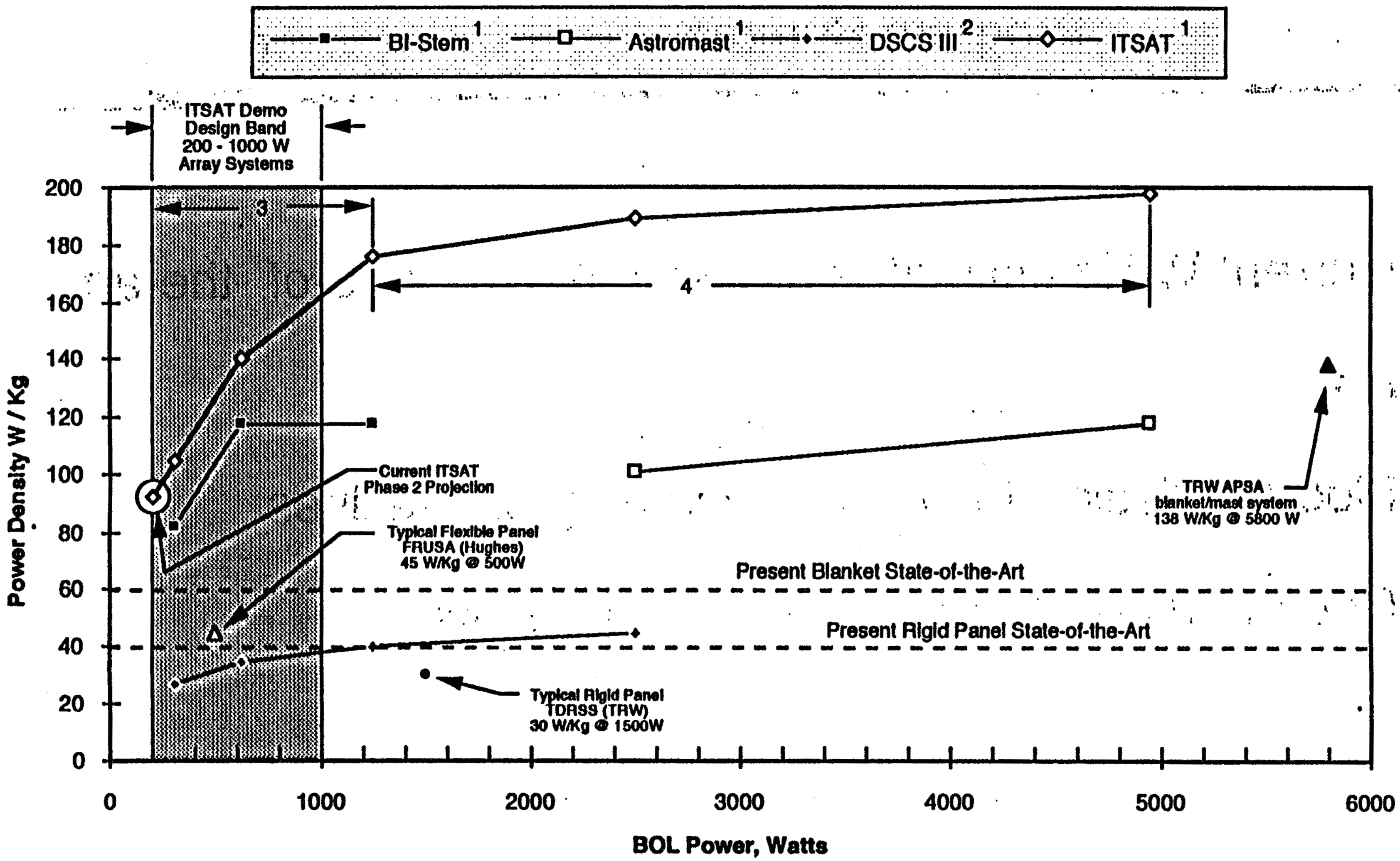
The sodium-sulfur battery is emerging as a prime high-temperature energy storage technology for space flight applications. Useable specific energy of two to three times that of $Ni-H_2$ batteries is the key advantage of Na-S. This represents a significant launch cost savings and increased payload mass capability. Additionally, the Na-S battery offers high power capability attractive for some future military and science satellite applications. While ground testing has shown the Na-S cells to be versatile (capable of operation in any orientation and capable of handling severe vibration and shock loads), the basic cell has yet to be qualified for operation in the microgravity environment of space. There is buoyancy separation of the very viscous liquid sulfur and more fluid sodium polysulfides in the cell on the ground. Spatial control of the molten reactants is critical to proper cell operation. The present design of the cathode is based on capillarity and differential wettability for reactant transport within the cell. Thus the cell is believed to be suitable for operation in microgravity environments.

A Na-S cell demonstration is planned for a 1995-96 NASA Space Shuttle flight which focuses on the microgravity effects on individual cells. The experiment is not optimized for battery performance as such. Rather, it maximizes the variety of operating conditions which the Na-S cell is capable of in a relatively short 5-day flight. The demonstration is designed to reveal the effects of microgravity by comparison with ground test control cells experiencing identical test conditions but with gravity. Specifically, limitations of transport dynamics and associated cell performance characteristics should be revealed.

Experiment Description

The Na-S Cell Technology Flight Experiment consists of three separate experiments designed to determine cell operating characteristics, detailed electrode kinetics and reactant distributions. The experiments, summarized in Table 1, are controlled by an autonomous Experiment Control Unit (ECU). Subordinate controllers include a Power Conditioning Unit (PCU), Thermal Control Electronics (TCE) and a Data Conditioning Unit (DCU). A total of 6 Na-S cells enclosed within 2 thermal enclosures will be mounted to a HitchHiker-M structure top pallet within the Shuttle cargo bay, as indicated in Figure 1. The PCU, having high power dissipative loads, will also be mounted on a top pallet for improved heat rejection. The ECU, TCE and DCU will be

Deployment Comparison: ITSAT vs Bi-Stem vs Astromast vs DSCS III (BOL/GEO/Crystal-Si)



1. L'Garde ITSAT with crystalline silicon cells on flexible blanket.
2. Rigid panel design.
3. ITSAT based on aluminum-kapton laminate.
4. ITSAT based on U-V cured resin structure.

TI 922

integrated into a side mounting plate. Space Shuttle energy is supplemented with Ag-Zn batteries side mounted in a Gas canister. The batteries allow for high power charging of the Na-S cells and support an orderly-emergency shut down.

TABLE 1. FLIGHT EXPERIMENT MATRIX

TEST	TEST No.	NUMBER OF CELLS IN SPACE	GROUND	TEMP °C	DISCHARGE-CHARGE**** RATES
CELL CHARACTERIZATION	IA	2	2	350	DISCHARGE: C/2, C , 3C/2, 2C CHARGE: C/4, C/2, 3C/4, C
	IB	2	2	350 & 300	DISCHARGE: C/2, C CHARGE: C/4, C/2
CURRENT/TEMP DISTRIBUTION	II	2*	2*	350 & 300	DISCHARGE: C/2, C +Interrupts CHARGE: C/4, C/2 +Interrupts
REACTANT DISTRIBUTION (THERMAL QUENCH)	III-1	4**	4**	350	2 @ FULL CHARGE After C/2 CHG 2 @ HALF CHARGED After C/2 CHG
	III-2	2***	2***	350	1 @ HALF CHARGED After C/2 DIS 1 @ FULL DISCHG After C/2 DIS
	TOTAL	6	6		

* SPECIALLY INSTRUMENTED CELLS
** REUSE TEST #I CELLS
*** REUSE TEST #II CELLS
**** SECONDARY CHARGE (@C/8) INCLUDED IN ALL CHARGE CYCLES

Test Planning. Because sulfur molten salts are extremely sluggish and the cell cycle times are of long duration, it is not possible to maintain low-g conditions for an adequate duration to simulate the experiment on earth.

The criteria used for selecting these space experiments was that they could not be properly simulated with on-ground tests. The feature common to all of the selected tests pertains to the unknown spatial distribution of cathode reactants (molten S and Na_2S_x, x = 3 to 5) with respect to the Na^+-conducting solid electrolyte (Beta"-alumina). Together with the large volume shrinkage (>30%) of the reactants during charge, formation of insulating sulfur or void volume adjacent to the electrolyte could result in premature termination of the charge cycle and loss of capacity on the subsequent discharge cycle. Although differential surface tension is utilized in the cathode design to optimize gravity-assisted operation, it is unclear whether phase consolidation and segregation of immiscible components will occur without gravity assistance, so that capillarity can remain functional in space.

The overall objective of the Na-S Cell Flight Experiment is to validate or refine the cell design codes for optimum performance and extended operation in space environments. The specific objective of Experiment I is to determine cell characteristics as a function of the operating temperature, charge rates and discharge rates for comparison with response on earth. These operating parameters will be varied from moderately slow cycles (C/2 Discharge, C/4 Charge) to very fast cycles (2C Dis, 1C Chg) at several temperatures to support conceptual engineering designs for diverse space applications.

The cell current time line for Experiment I, for example, is shown in Figure 2. Following one balanced conditioning cycle, the cells are operated for 3 cycles at 350°C at each test condition. Two of the charge/discharge rates are repeated at 300°C.

The objective of Experiment II is to determine variations in the distribution of reactants and reactions within the cathode volume as they develop in real time during operations in space. The objective of Experiment III is similar to Exp II - to document the spatial distribution of cathode reactants as a function of the cell's state-of-discharge and as a function of the dynamics of its previous electrical cycle as it is established without gravity. Experiment III involves a rapid thermal quench of the Exp I and II cells to immobilize the reactant distributions for analysis by DPA after re-entry.

Design Considerations

The two most demanding aspects of the Experiment design are thermal control and minimization of required STS energy. Because Na-S cells must be maintained at about 350°C for proper operation, and since each test requires separate conditions, two thermal enclosures are required. During most of the 5-day mission, heat generated within the Na-S cells is not adequate to offset thermal losses, hence substantial energy is required from the shuttle bus.

Cell Enclosure. The cell enclosure design is shown in Figure 3. Ceramic components provide mechanical support and electrical isolation for each cell, the interconnections, instrumentation leads, and the heater elements. Redundant heaters, positioned between cells for more uniform temperature, are sized for 150W for adequate heatup rates, but are derated to 60W during most of the mission to prevent excessive overtemperature. The inner container housing, a welded structure with hermetic electrical feedthroughs, provides absolute containment of any cell reactant leakage.

An outer module structure (Figure 4) with insulation is sealed by gaskets to form a third hermetic enclosure. Each experiment module is anchored to the HH-M top pallet with a NASA-LeRC designed support frame attachment that permits elongation of the enclosure as it periodically is heated and cooled, but which constrains transverse motion.

Thermal Design. To minimize heat loss from the enclosure, evacuated MLI (Multi-Layer Insulation) is utilized on the large area sidewall, and a high quality fibrous insulation (Min-K 1301) is used on the ends. Three or four large electrical conductors are required per enclosure to provide flexibility in control of test currents up to 80A. Power and instrumentation leads must be carefully sized to minimize heat loss across the thermal gradient. Steady thermal loss for the present design is projected at about 50W per enclosure, resulting in approximately 10 kWh of energy demand for thermal control.

Three aspects of thermal control are: (1) heaters for initial heatup and maintenance of test temperature, (2) limited cooling during occasional high-current cell tests (Exp I), and (3) massive cooling for reactant thermal quench (Exp III) as well as for any emergency situation to quickly secure the experiments for re-entry.

A Variable Conductance Insulation System (VCIS) is incorporated into the enclosure design. By injecting low pressure Helium gas into the MLI system, heat conductance dominates over the normally low radiation transfer rate. To re-establish good insulative properties, the helium is vented to space. When evacuated, the VCIS thermal loss is predicted to be less than 10 watts. In contrast, high effective cooling rates in excess of 400W are predicted for gas pressures of a few tens of Torr. A pressure regulated-helium gas supply is incorporated to adjust the cooling rate for the separate experiments.

Power Control Unit (PCU). It is an unfortunate paradox that to test high-power high-efficiency Na-S cells, a very high power/high energy source is required. Because of shuttle power limitations, we have selected large Ag-Zn primary batteries to provide the peak power and additional energy for recharge of the Na-S cells. The PCU circuit configuration is shown in Figure 5. Two independent PCU circuits are required to control the experiments since they are being performed simultaneously.

Linear regulators were selected for current control in order to minimize EMI interference with the extensive instrumentation required. Parallel MOSFETs are designed to control 40A charge and 80A discharge conditions. Details of the switching network for Experiment I are shown in Figure 6. Mechanical relays were selected for the switching network to minimize circuit losses because current must flow through at least four contacts in series to provide the desired interconnection of the test cells to the charger and the load. Recovery of energy from the discharge of the Na-S cells was not deemed practical because it occurs at low voltage and high amperage. In addition, the available energy is relatively small compared to the required primary energy.

The overall energy requirement for the 5-day Flight Experiment, shown in Table 2, is approximately 32 kWh which substantially exceeds the normal HH-M allocation to individual experiments. Energy required to power the control systems is about 12 kWh, and that for electrical heaters to overcome thermal losses from the two module housings is about 10 kWh. The subtotal of 22 kWh is within the HH-M energy budget, assuming the Na-S experiment is granted half the normal HH-M energy (in proportion to its area usage) plus a request for half the available supplemental HH-M energy. The maximum power level for these loads is well within allowable values.

TABLE 2. ESTIMATED ENERGY REQUIREMENTS

LOAD	AVG	PEAK	ENERGY	PURPOSE	SOURCE
Control					
ECU	30W	---	3.6kWh	Electronics	
TCE	15	---	1.6	Electronics	
PCU	30	---	2.0	Electronics	
DCU	10	---	1.1	Electronics	
RESERVE	30	---	3.3	Electronics	
Totals	115		11.6		STS-BUS
Heater Power					
TEST I	50W	150W	4.5kWh	Heatup & Maintain	
TEST II	50	150	4.9	Temperature	
TEST III	100	100	0.5		
Totals	140	340	9.9		STS-BUS
Cell Charging					
TEST I		320W	6.6kWh	Recharge	
TEST II		130	3.3	NA-S Cells	
TEST III		260	0.3		
Totals		440	10.2		AG-ZN BATT

One-third of the total energy is required to recharge the Na-S cells, and much of this occurs at high peak power levels. Such peak power demands if added to the other continuous loads would exceed HH-M maximum power levels and would constrain the timing of the individual experiments to avoid overlapping the power peaks. By incorporating the Ag-Zn primary battery source, the separate cell experiments can be performed independently, and the overall energy requirement falls within the budget for HH-M experiments.

Mission Planning

Payload integration and pre-launch testing are particularly simple because the Na-S cells are inactive at ambient temperature. Following orbit stabilization and cell heatup, a space conditioning cycle is included for all cells to disrupt the gravity-induced reactant distribution that was carried into space by the cells being frozen prior to launch.

Because of the short 5-day mission and the time to heat/cool the cells, careful planning of test sequences is necessary. Contingency plans will be developed for alternate tests and for maximizing information should a shuttle emergency or experiment malfunction develop and require premature shutdown by the autonomous controller. Cells will be secured with all reactants frozen prior to re-entry.

Acknowledgement

We gratefully acknowledge the program support of Olga Gonzalez-Sanabria and the technical suggestions of Harold Leibecki of NASA Lewis Research Center during the preparation for this critical space experiment.

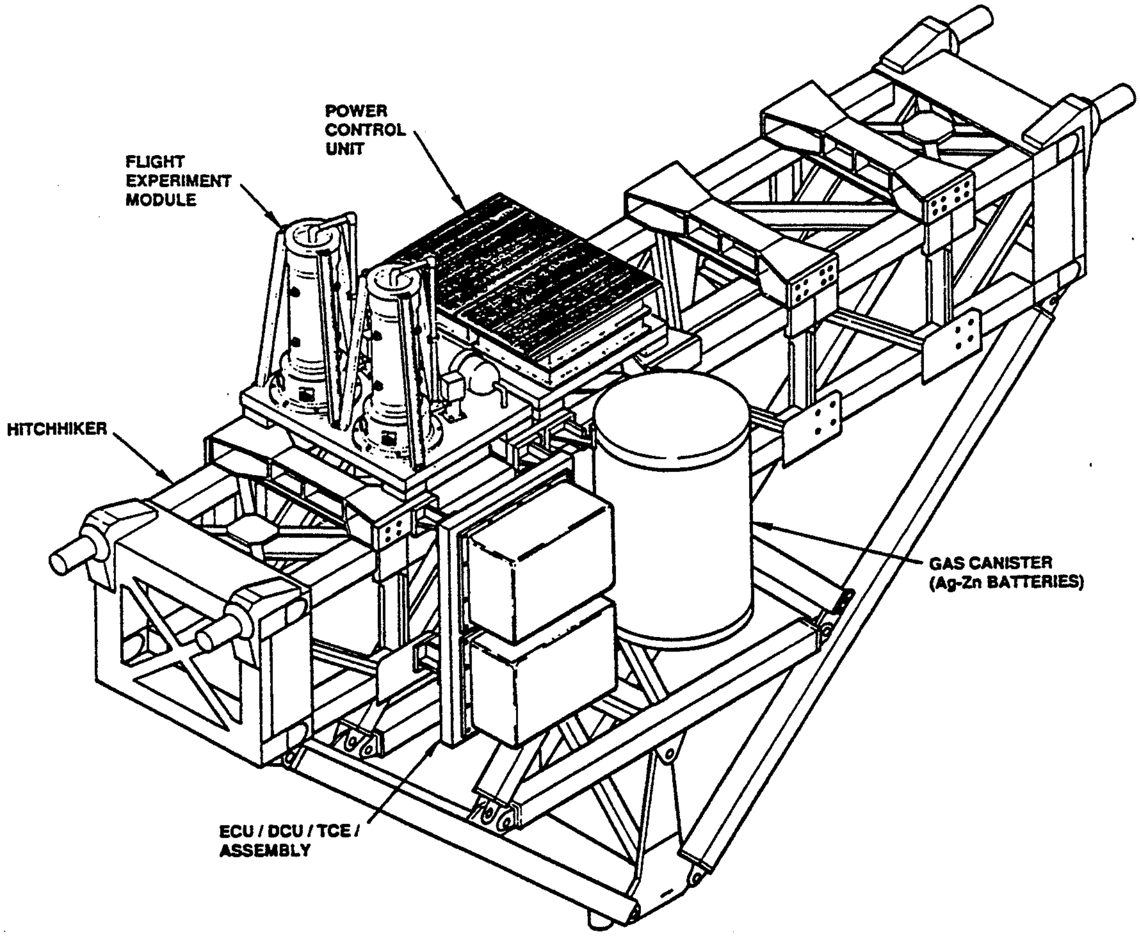


Figure 1. SSCT PAYLOAD CONFIGURATION

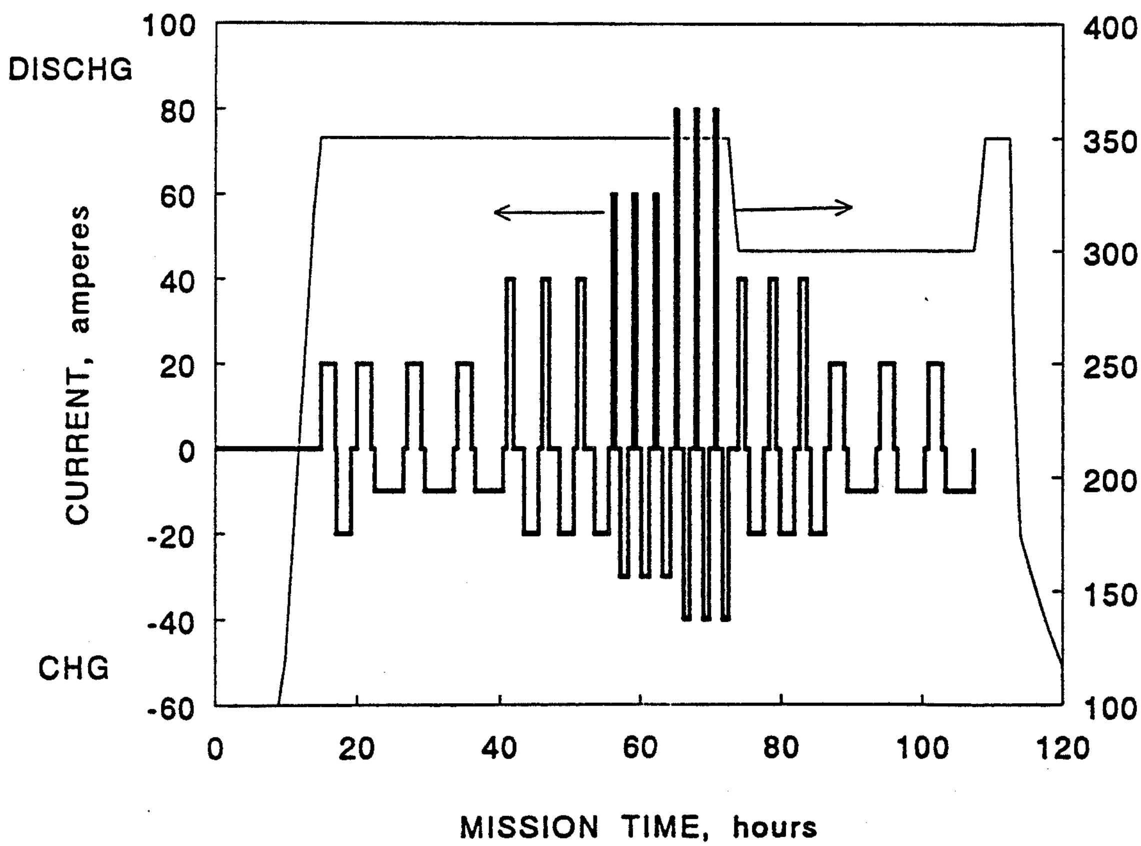


Figure 2. CELL CURRENT TIME LINE - TEST I

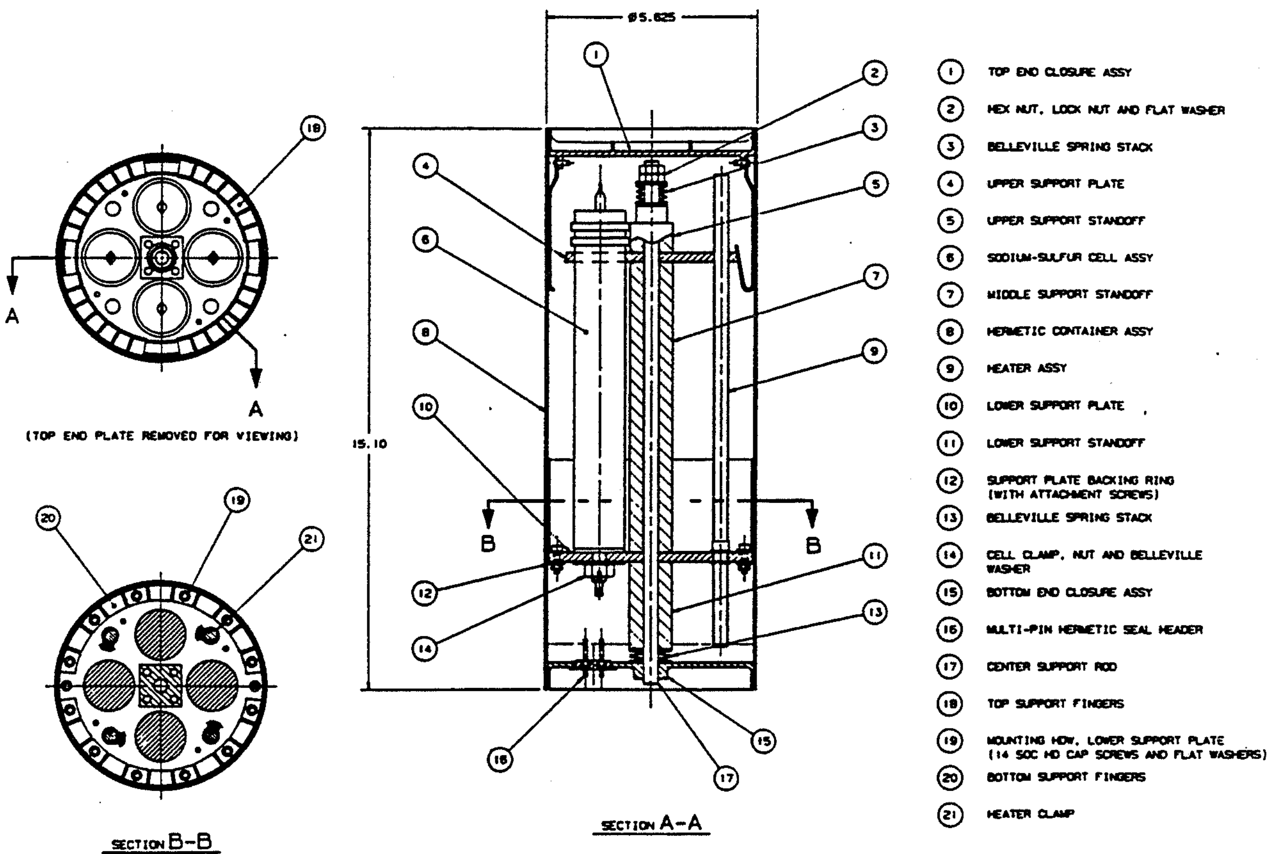


Figure 3. INNER CONTAINER ASSEMBLY

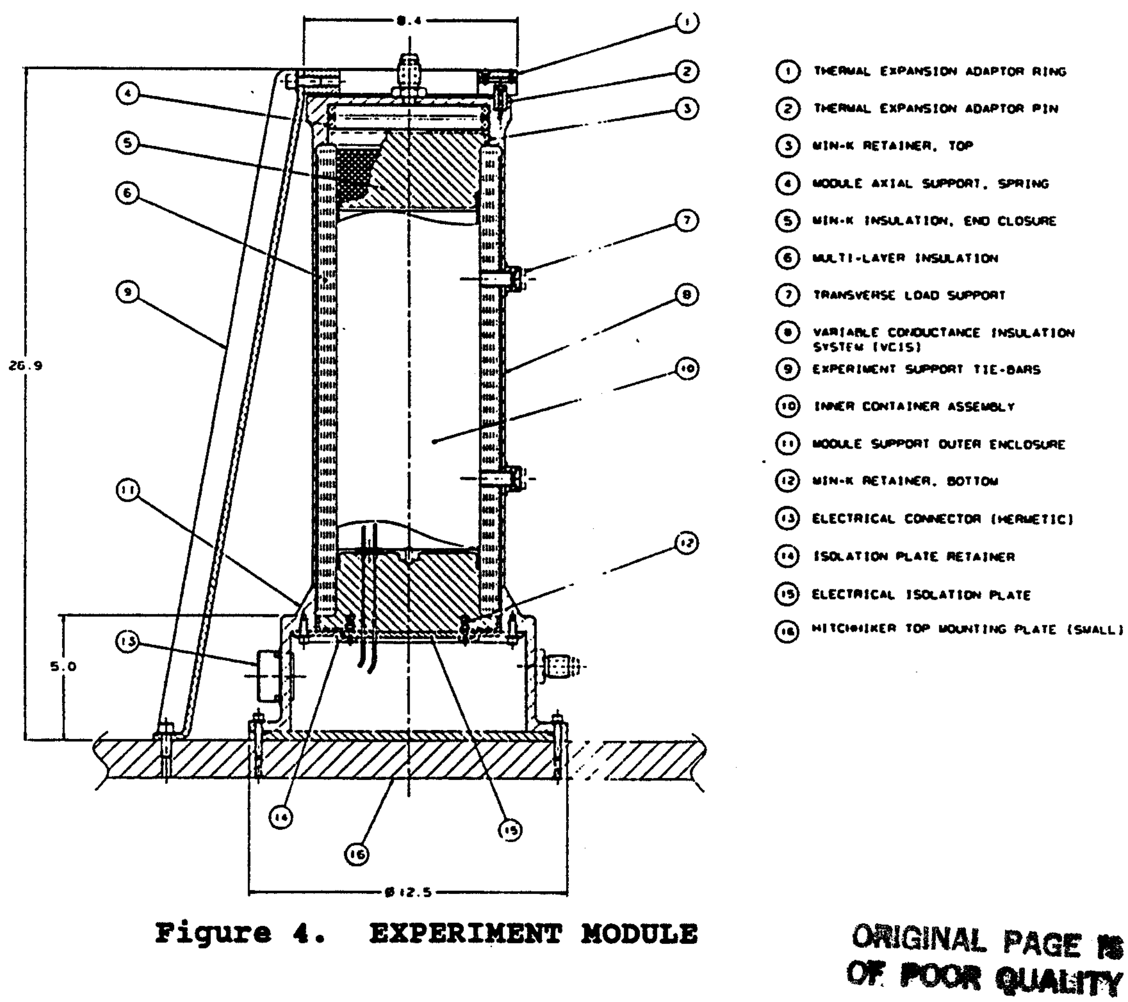


Figure 4. EXPERIMENT MODULE

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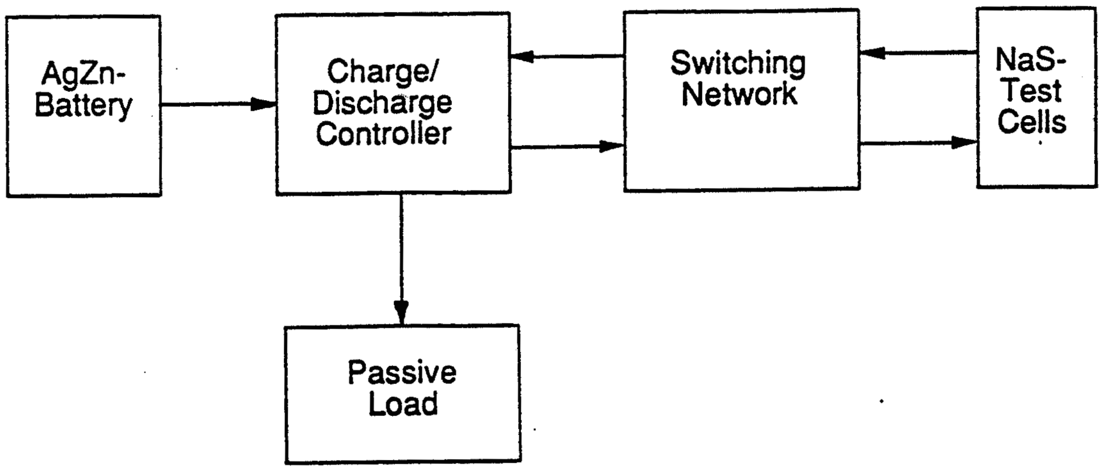


Figure 5. POWER CONTROL UNIT CONCEPT

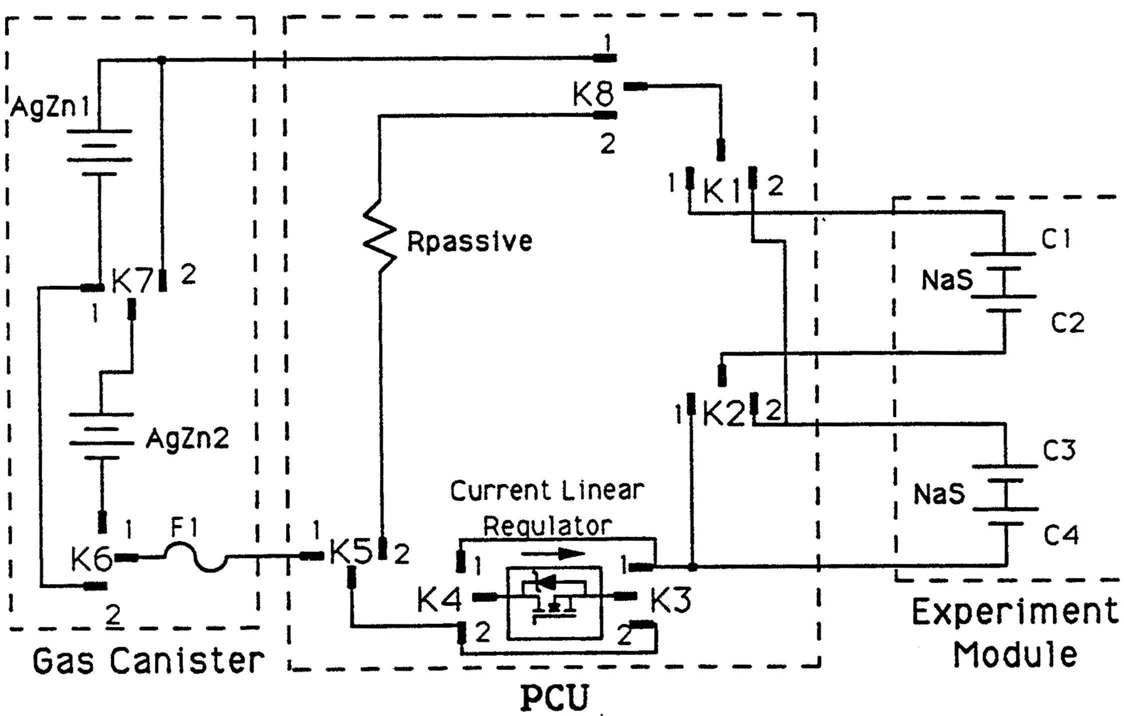


Figure 6. SWITCHING NETWORK CIRCUIT

AEROSPACE TECHNOLOGY DIRECTORATE

POWER TECHNOLOGY DIVISION

NASA
Lewis Research Center

NASA/DOD FLIGHT EXPERIMENTS
TECHNICAL INTERCHANGE MEETING
OCTOBER 6, 1992

SOLAR ARRAY MODULE PLASMA INTERACTION EXPERIMENT (SAMPIE)

Dr. Dale C. Ferguson, Principal Investigator
Space Environment Effects Branch
NASA Lewis Research Center

Summary

- Space data is badly needed
  - a major obstacle to orbiting HV power systems is Plasma Interactions
  - Ground testing cannot reliably predict on-orbit behavior; space data needed (PIX data is still being analyzed for insight into real space behavior)
  - future designs will rely heavily on computer models, very little space data exists for validation
  - SAMPIE will be the first space experiment to provide data on the performance of various solar cell technologies under identical environmental conditions

Summary (cont)

- Design is highly modular
  - Conceived as "testbed" for emerging solar cell technologies
  - Can be easily repackaged for non-shuttle deployment
- SAMPIE will have a wide ranging impact
  - Technology base will benefit directly from understanding of HV systems behavior in plasma environment
  - Space Station Freedom depending on data return from SAMPIE

SAMPIE was never designed to be SSF specific; BUT recent developments have rendered SAMPIE data critical. (WP-04 is generating CR for plasma contactor, specifically requires SAMPIE's data for contactor optimization)

SAMPIE QUICK SUMMARY

- OAST IN-STEP Shuttle Experiment
- Hitchhiker experiment, Shuttle Payload Bay
- Manifested on OAST-2 in late January, 1994
- Passed CoDR, NAR, Phase 0/1 Safety Review, PDR
- CDR at end of October, 1992
- LEO Plasma Interactions Experiment
- Heritage - SPHINX, PIX, PIX II, VOLT-A
- NASA Lewis experiment, built with in-house contractor (Sverdrup)

Experiment: Objectives

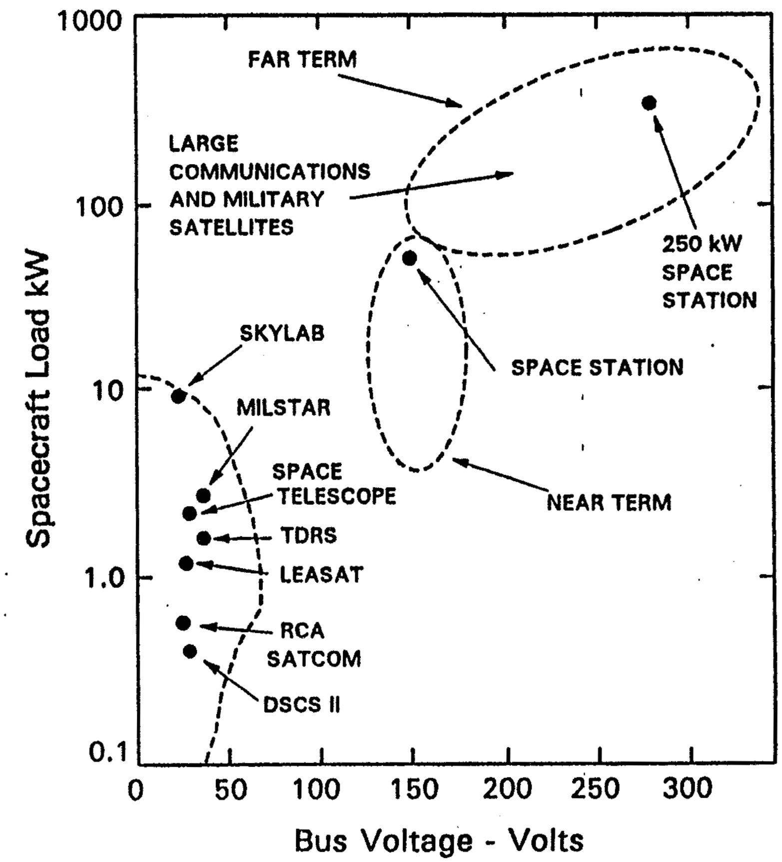
The objective of SAMPIE is to investigate, by means of a shuttle-based space flight experiment and relevant ground-based testing, the arcing and current collection behavior of materials and geometries likely to be exposed to the LEO plasma on High Voltage space power systems, in order to minimize adverse environmental interactions.

Space Power Systems

- Historically systems were low voltage, < 100 V
  - Earliest systems used batteries
  - 28 Volt technology inherited from aircraft industry
- Future systems desire much higher voltage
  - more efficient, I^2R losses smaller for given P = IV
  - less cable mass to orbit
- Proposed systems range from:
  - hundreds of volts for SP-100, to
  - thousands of volts for orbit transfer vehicle using solar electric propulsion

TRENDS IN SPACECRAFT POWER

From: Space Vehicle Design
M. D. Griffin
J. R. French



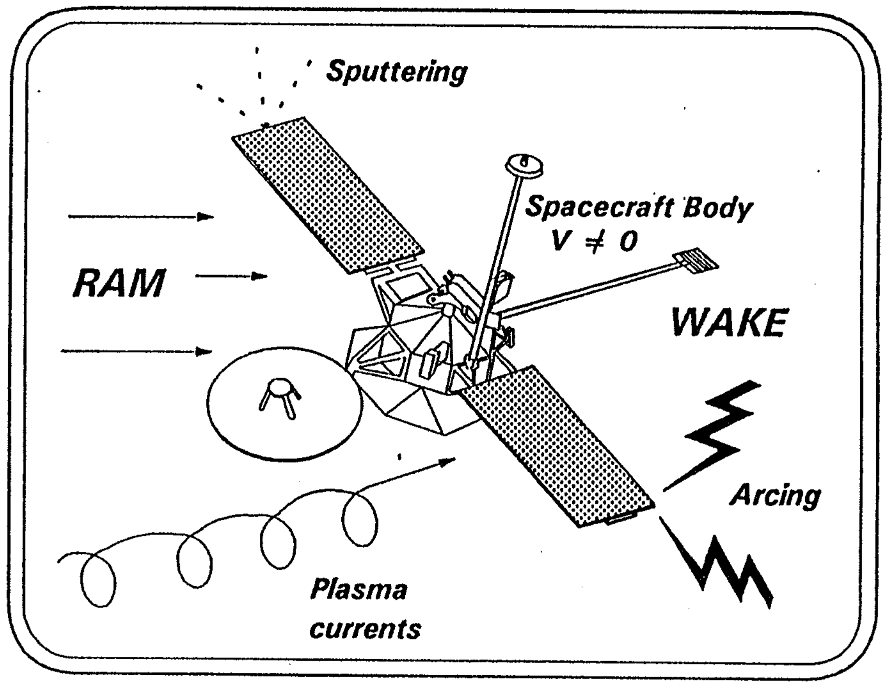
SACE1.DRW

The Problem

High voltage systems immersed in ionospheric plasma will inevitably interact

- **Negative potential ⟶ arcing**
  - results in broadband EMI, electrical transients
  - has been known to completely destroy solar arrays in laboratory
  - high energy inbound ions cause sputtering

- **Positive potential ⟶ parasitic current drain**
  - even a small pinhole can "snapover", result in large power losses
  - equilibrium current balance determines floating potential of spacecraft



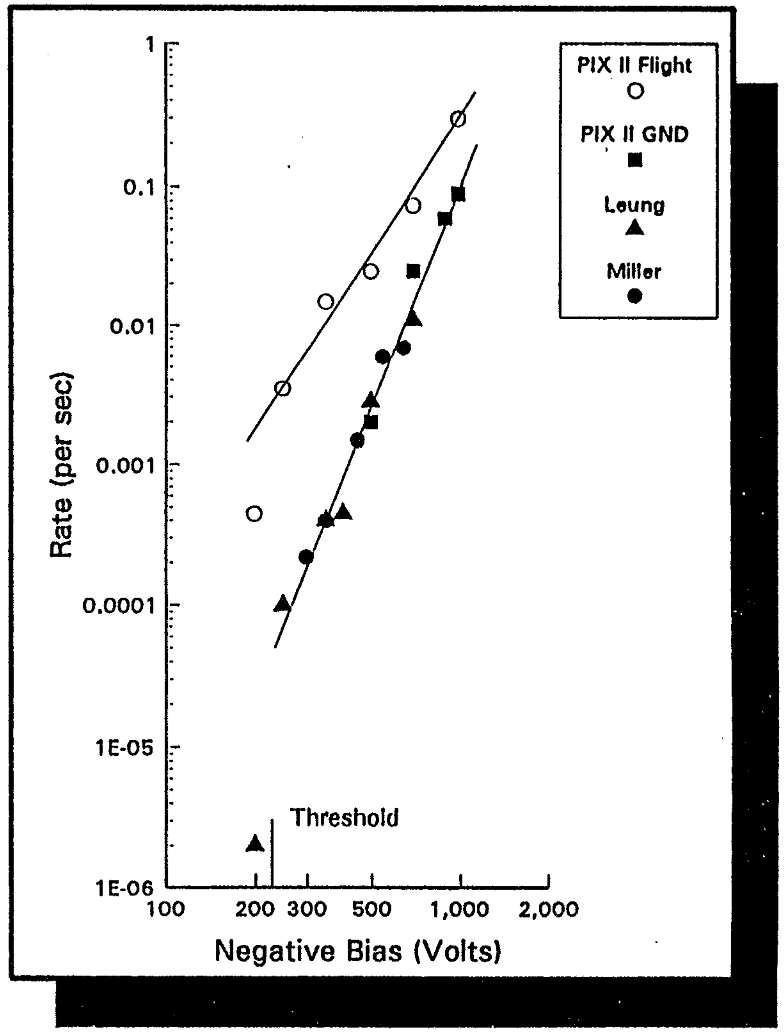
PROBLEM.DRW

Ground vs Flight

- Ground tests results are always different than flight
- Ground tests cannot reproduce environment

	LEO	Ground
Temp (eV)	.1 - .2	2 - 3
Dominant Ion	O^+	Ar^+
Flow	Mach 20 Ram/Wake	Thermal

- Chamber walls are impossible to compensate for
  - Secondary electron emission occurs from walls
  - SPEAR tests showed discrepancies even with huge Plum Brook chamber



Present Knowledge
Arcing

- Arcing is a negative potential phenomena
  $(M_e << M_{ion})$
- Arcs occur at conductor/insulator junctions, may propagate directly into plasma
- Threshold voltage seems to exist, but
  - different for different materials
  - not predictable from existing theory
  - depends on plasma conditions
- Produce EMI, transients in nearby circuits
  - measured for VOLT-A, may require waiver of shuttle requirements

Present Knowledge
Arcing
(continued)

- Arc rate depends (in a complicated, poorly understood way) on:
  - voltage
  - ion current
  - materials, principally conductor, some evidence for dependence on properties of nearby insulators
  - plasma conditions
  - geometry
- Arc rate seems independent of array area (number of cells)
- Arc rate/duration depends on:
  - array capacitance ⟶ RC time constant ⟶ duration
  - overall circuit ⟶ how much power available ⟶ strength

PKARC2.DRW

Knowledge Gaps; Arcing

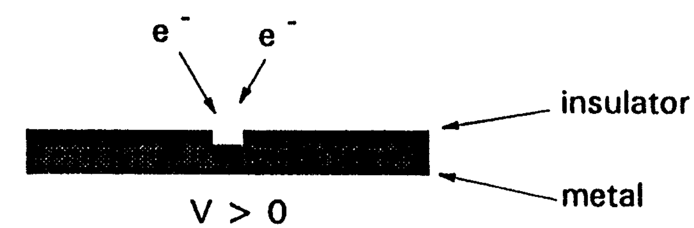
- Above mentioned dependencies of rate, threshold, strength, etc. are not quantitative
- Existing knowledge based on behavior of old technology silicon cells. Theoretical framework and emperical scaling laws are not developed sufficiently to predict behavior of new cells.
- Fundamental nature of breakdown and arcing is controversial and poorly understood
- Role of ion energy not clear, ram/wake effect may be critical

Present Knowledge; Current Collection

- Parasitic current collection is a positive potential phenomenon

$$M_e << M_{ion}$$

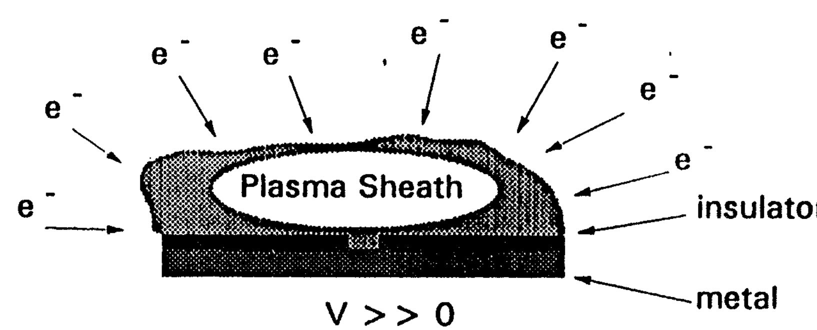
- The major concern is "SNAPOVER"



large surfaces collect large current,
small surfaces - small current

BUT

at HV, a very small exposed conductor, e.g. solar cell interconnect or pinhole in insulation can cause plasma sheath to form over large area, entire surface effectively becomes conducting



PECURI.DRW

Knowledge gaps; Current Collection

Physics of sheath formation is poorly understood, when does it form, how big will it be, how conductive, how does it depend on:

- Bias voltage
- material properties (e.g. vapor pressure, work function)
- geometry - appears to be critical, poorly understood
- electron energy, i.e. ram/wake effect

Objectives
(Summary)

The current selection of experiments for SAMPIE will:

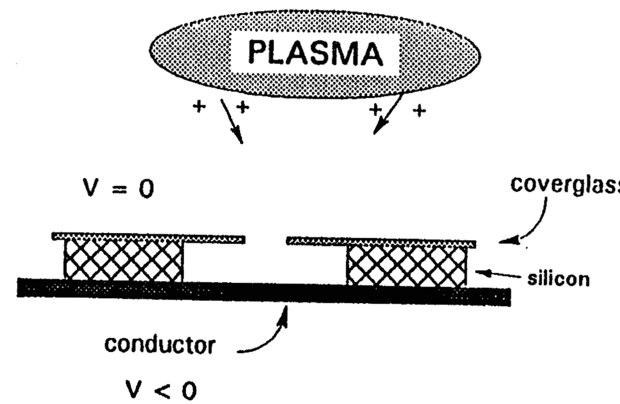
- Directly test plasma interaction effects of several solar cell technologies
  - APSA, Space Station, standard silicon
- Test arc suppression techniques with modified cell arrays
  - Two proposed techniques will get direct space test
- Study basic phenomena underlying Technology challenges
  - Pinhole experiment will study current collection; family of I-V curves will validate NASCAP/LEO, test existing theories
  - Two sets of metal samples will study arcing: family of arc rate vs bias curves will explore effects of material properties
  - Study dielectric breakdown of anodized aluminum
- Provide data for model validation
  - Most of the above will serve this purpose
  - Silicon experiment with "guardrings" for scaling effects
  - Modified SSF cell array with selected edge coating will study current collection

OBSUM.DRW

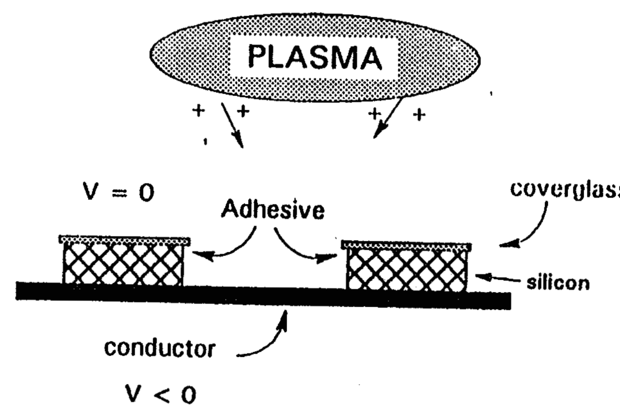
Objectives
(cont)

Two arc mitigation techniques will be explored:

- Extended coverglass:
  - * Coverglass extension shields triple points
  - * Charge buildup on coverglass will help repel inbound plasma ions
  - * NASCAP/LEO will calculate ion motion, sheath formation

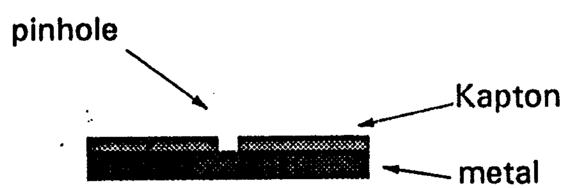


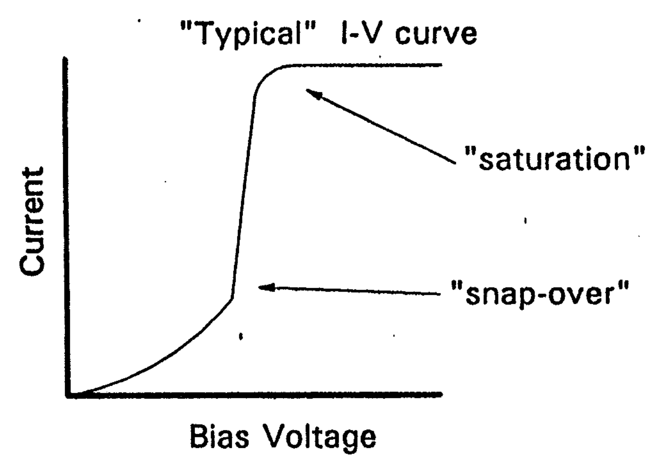
- Special Processing (cleaning)
  - * Researchers at PSI (W. Marinelli et. al.) have been studying arcing under contract from LeRC
  - * Results to date indicate a primary source of arcing is excess coverglass adhesive: proprietary cleaning processes are under development
  - * One cell coupon will be sent to PSI, processed, and returned for flight



Objectives
(cont)

- Design and fly a controlled experiment studying parasitic current collection
  - Geometry of solar cells, other flight hardware too complicated
  - Key parameter is pinhole size
  - Use 1 cm metal disks, covered with Kapton, each having a different pinhole





  - Result will be a family of I-V curves
  - Details, including snapover point, can be modeled using NASCAP/LEO, various analytic treatments
  - SAMPIE will fly 6 such pinholes; sizes to be chosen from ground tests, modeling

OBJ.DRW

Objectives (cont)

Study current collection vs solar cell edge treatments

- Background:
  - Without a plasma contactor, SSF would float -140 V wrt ionosphere as result of current collection.
  - SSF cells average approx 70% cell edge adhesive coverage, 4 mils coverslide overhang.
  - NASCAP/LEO predicts rapid improvement in floating potential as coverage exceeds 90% or overhang exceeds 6 mils.

- Experiment:
  - Need to validate NASCAP/LEO.
  - Use scaled down versions of SSF solar cells.
  - Systematically vary degree of edge coating and overhang. Nominal overhang is 4 mils, so use:

0 mils	3 mils
6 mils	9 mils

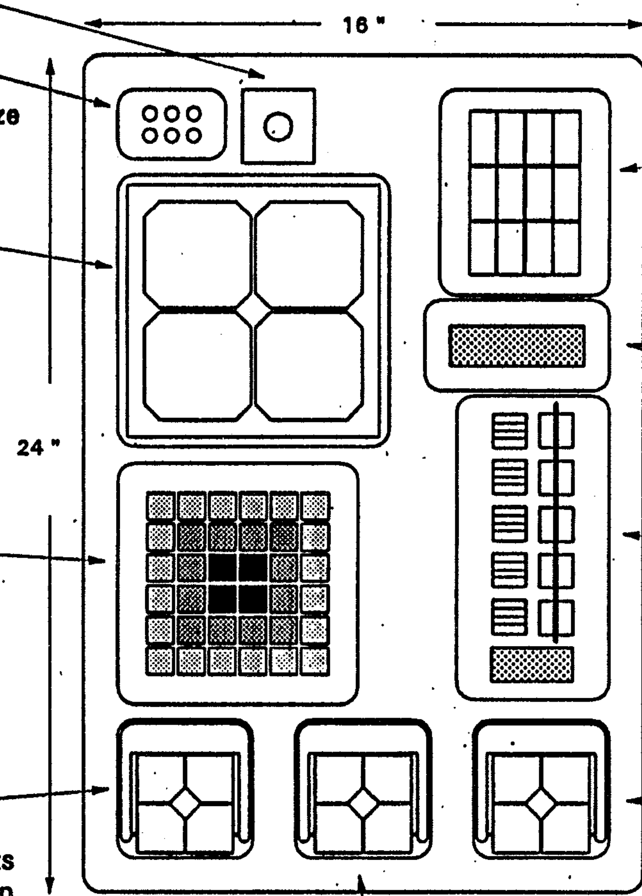
Objectives
(cont)

- Design and fly, on a space available basis, additional arcing experiments: One currently chosen
  - breakdown from anodized aluminum
    - * alloy and anodization chosen to be space station baseline

- Measure a basic set of Plasma parameters
  - Langmuir probe - plasma density, electron temperature
  - Pressure gauge - background pressure, fast enough to detect thruster firings
  - Sun sensor
  - V-body probe - monitor orbiter potential

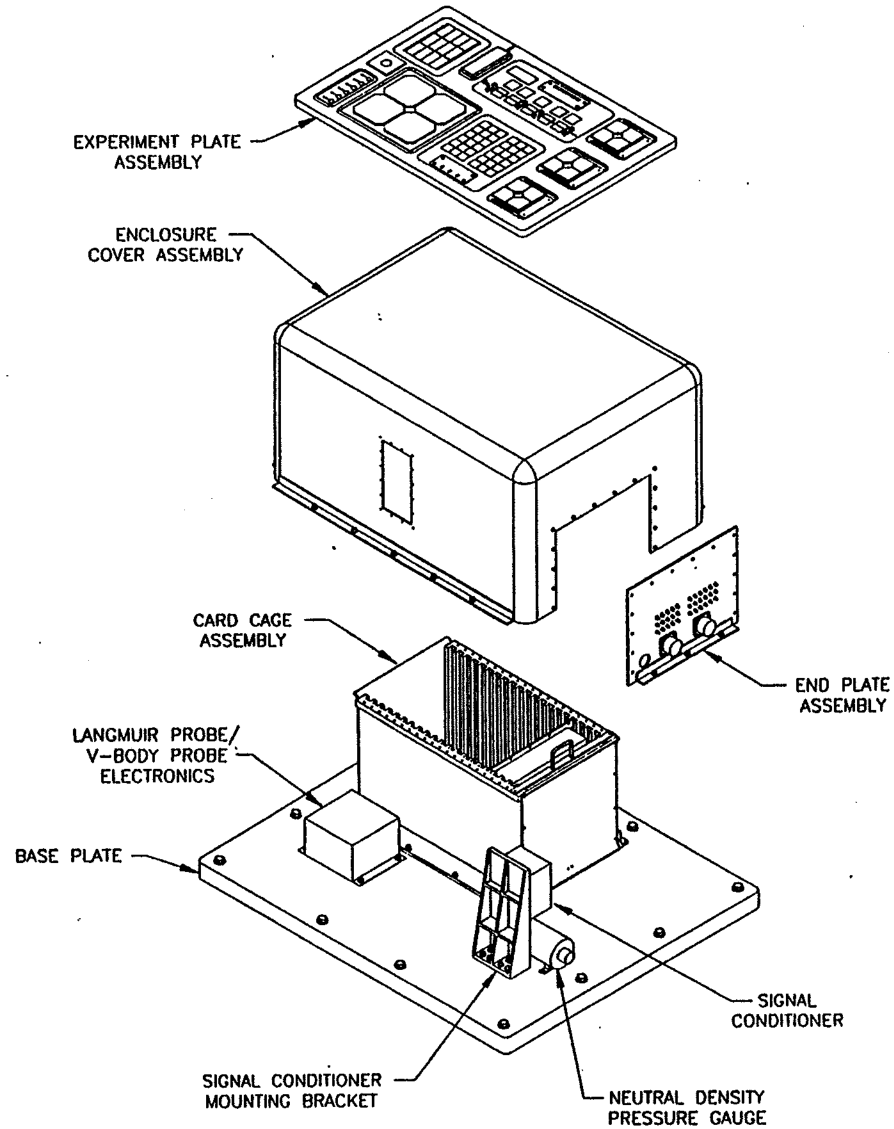
SAMPIE

Experiment Configuration

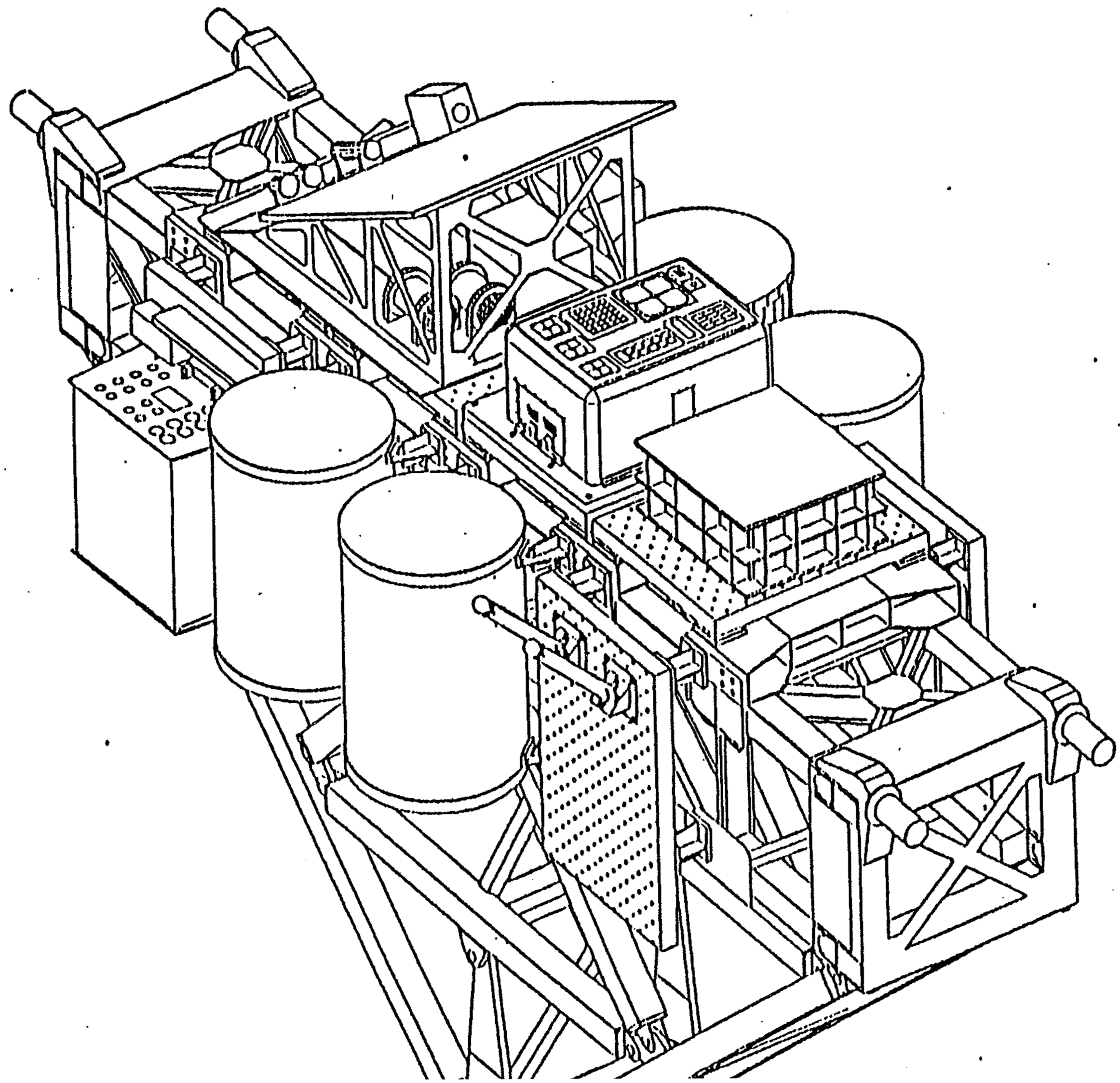


SAMPIE MAJOR COMPONENTS

- BASE PLATE (MPESS MOUNT)
- ENCLOSURE COVER ASSEMBLY
- EXPERIMENT PLATE ASSEMBLY
- CARD CAGE ASSEMBLY
- END PLATE ASSEMBLY
- LANGMUIR PROBE ELECTRONICS
- SIGNAL CONDITIONER
- NEUTRAL PRESSURE GAUGE
- LANGMUIR AND V-BODY PROBES



MAIN ELECTRICAL ASSEMBLY

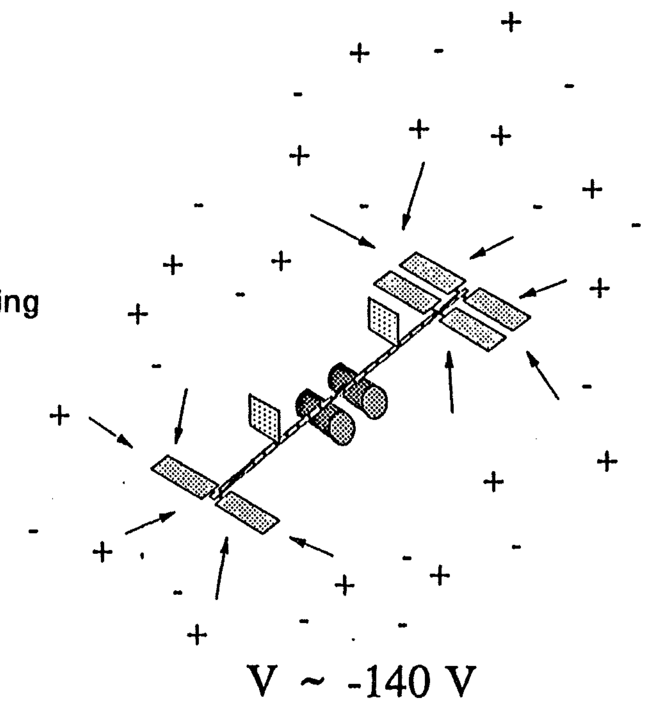


SAMPIE
and
The Plasma Contactor

SAMPIE was designed to study the interaction of HV systems with space plasma

SSF has inadvertantly become an immediate test of these effects

- Current collection by SSF will result in an equilibrium with the station "floating" about -140 volts WRT the ionosphere
- Predictions depend on details of:
  - cell collection characteristics - coverglass overhang, edge coating
  - ion collection by structure
- All predictions follow from modeling
  - NASCAP/LEO
  - Environmental Workbench (EWB)
- Models have only limited validation
  - sounding rockets
  - ground tests

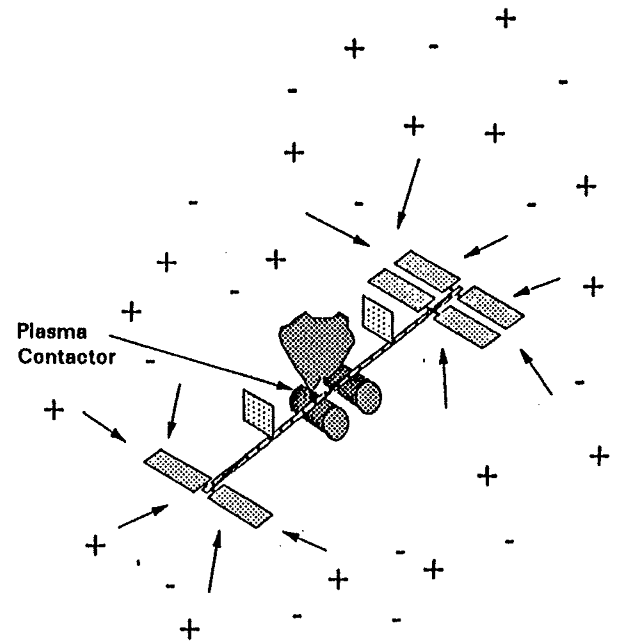


CONTACT.DRW

The Plasma Contactor
(CONT)

Over the past year, the problem, its implications, and possible design solutions have been studied extensively

- Plasma Contactor selected and baselined
  - Brute force solution, will unquestionably force SSF "near" plasma ground
- Issues:
  - How big should it be?
  - How much EMI will it cause?
  - Where on the structure should it be placed?
- Best guesses based on computer models
- SAMPIE will provide critical data allowing model validation and refinement of contactor design and placement



V ~ -20 V

CONTACT.DRW

THE PHOTOVOLTAIC ARRAY SPACE POWER PLUS DIAGNOSTICS (PASP PLUS) FLIGHT EXPERIMENT

by

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Cleveland, Ohio 44135

&

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Hanscom Air Force Base, MA 01731

Presented at Flight Experiments Technical Interchange Meeting
Monterey, CA
October 6, 1992

PASP PLUS FLIGHT EXPERIMENT

OUTLINE

- EXPERIMENT DESCRIPTION
- OBJECTIVES
- SOLAR ARRAY MODULES
- CONTROL & DIAGNOSTIC EQUIPMENT
- ILLUMINATED THERMAL-VAC TESTING
- SUMMARY

PASP PLUS FLIGHT EXPERIMENT

PASP PLUS: PHOTOVOLTAIC ARRAY SPACE POWER PLUS DIAGNOSTICS

- PRIMARY EXPERIMENT ON THE USAF APEX (ADVANCED PHOTOVOLTAIC AND ELECTRONICS EXPERIMENTS) MISSION
- PURPOSE IS TO TEST A VARIETY OF PHOTOVOLTAIC CELL AND ARRAY TECHNOLOGIES UNDER VARIOUS SPACE ENVIRONMENTAL EFFECTS

PASP PLUS EXPERIMENT MANAGED BY GEOPHYSICS DIRECTORATE, U.S. AIR FORCE PHILLIPS LAB

- NASA LEWIS HAS PRIMARY ROLE IN ASSISTING THE U.S. AIR FORCE IN THE INTEGRATION, TESTING & DATA INTERPRETATION OF THE INDIVIDUAL SOLAR ARRAY EXPERIMENTS
- POWER & THERMAL MGMT. DIVISION, PHILLIPS LAB HAS THE PRIMARY ROLE IN PV MODULE EXPERIMENT SELECTION AND EXPERIMENTOR INTERFACE (RESPONSIBILITY TRANSFERRED FROM WRIGHT LAB)

MISSION OVERVIEW

- MISSION: SSD/CLP MISSION P90-6, ADVANCED PHOTOVOLTAIC AND ELECTRONICS EXPERIMENT, APEX

- LAUNCH: PEGASUS, 2Q FY93

- ORBIT: Perigee: 190 nmi (350 km)
  Apogee: 1054 nmi (1950 km)
  Inclination: 70°
  Orientation: Sun Pointing ± 0.5°

- DURATION: One Year Minimum; Three Year Goal

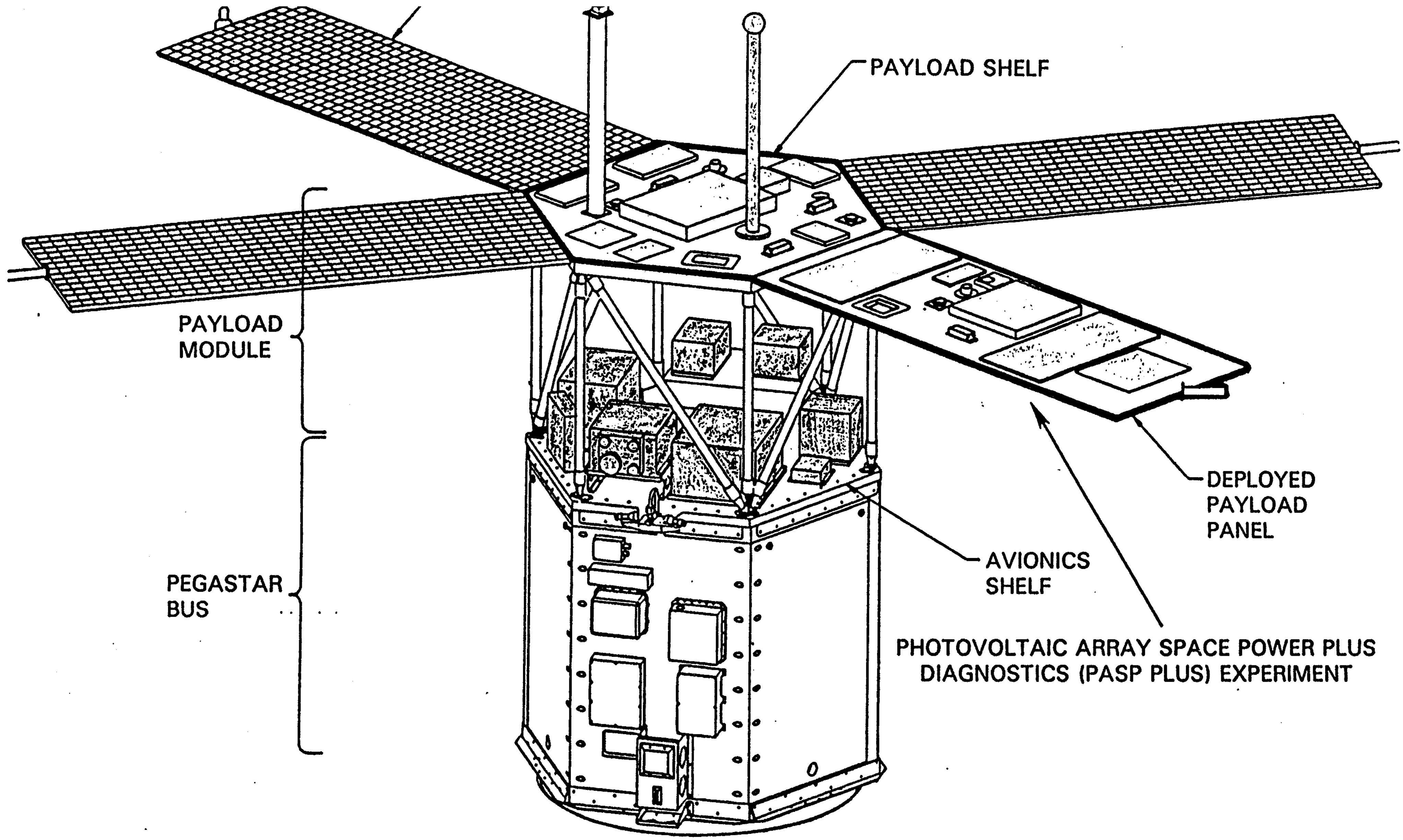
EXPERIMENT DESCRIPTION

- INTEGRATED SYSTEM OF ADVANCED SOLAR ARRAYS AND DIAGNOSTIC INSTRUMENTATION

  - ☐ Arrays: Many Different Materials and Configurations Represented in 12 Arrays Divided into 16 Separate Modules

  - ☐ Subsystems: Experiment Controller/High Voltage Generator
    Electron Emitter (Vehicle Potential Control)

  - ☐ Diagnostics: Transient Pulse Monitor (Arc Parameters)
    Langmuir Probe (Plasma Density & Temp.)
    Electrostatic Analyzer (Auroral Passage)
    Proton/Electron Dosimeter (Flux & Dose)
    QCMs and Calorimeters (Contamination)
    Sun Sensor (Concentrator Sun Pointing)



ADVANCED PHOTOVOLTAIC AND ELECTRONICS
EXPERIMENTS (APEX) SATELLITE

MSN#
(SSD/CLPM P90-6)

PASP Plus

OBJECTIVES

- DETERMINE HIGH-VOLTAGE OPERATION LIMITATIONS
  - Measure Plasma Leakage Current for Positive Biasing
  - Characterize Array Arcing for Negative Biasing

- QUANTIFY DEGRADATION CAUSED BY SPACE RADIATION
  - Performance Shown by I–V Curves
  - Dosimeter with Low-Energy Proton Capability
  - Contamination Monitors To Separate Effects

- PROVIDE FOR "SPACE QUALIFYING" NEW P/V TECHNOLOGIES

- ESTABLISH CAUSE-AND-EFFECT RELATIONSHIPS
  - Envir. Sensors To Indicate Space Conditions
  - Analyses, Modeling, Code Development

ON-ORBIT OPERATIONS

- EARLY OPERATIONS (FIRST EIGHT MONTHS)

  - ☐ Bias Selected Arrays to Specified Voltage Levels
    (up to –500 V and +500 V)
  - ☐ For Negative Biasing, Measure Arcing Parameters with TPM
  - ☐ For Positive Biasing, Measure Leakage Current with Electrometer
    (For some 100s of hours, electron emitter turned to control vehicle potential.)
  - ☐ Monitor Diagnostics To Characterize Plasma Environment
  - ☐ Monitor I–V Characteristics of All Arrays

- LONG-TERM OPERATIONS (UP TO THREE YEARS)

  - ☐ Monitor I–V Characteristics of All Arrays
  - ☐ Monitor Diagnostics To Determine Cumulative Radiation Dosage and Contamination Build-Up

PASP Plus

EXPECTED RESULTS

- HIGH-VOLTAGE OPERATION (SIMULATED BY BIASING)
  - ☐ Arc Pulse Parameters for Negatively Biased Arrays
    (pulse rate; amplitude, derivative, integral of largest pulse)
  - ☐ Leakage Current Parameters for Positively Biased Arrays
    (electron current below/above "snapover", with emitter off and on)
  - ☐ Over Flight Ranges of the Space-Environment Parameters
    (plasma density, velocity-vector orientation, auroral passage)

- RADIATION-INDUCED ARRAY POWER DEGRADATION
  - ☐ Continuing I – V Curves for All Array Segments
  - ☐ Continuous Radiation Dose and Flux Measurements
    (electrons and protons separately; emphasis on 5-10 MeV protons)
  - ☐ Over Three-Year Lifetime of Mission
    (possibly including a major solar proton event)

PASP PLUS FLIGHT EXPERIMENT

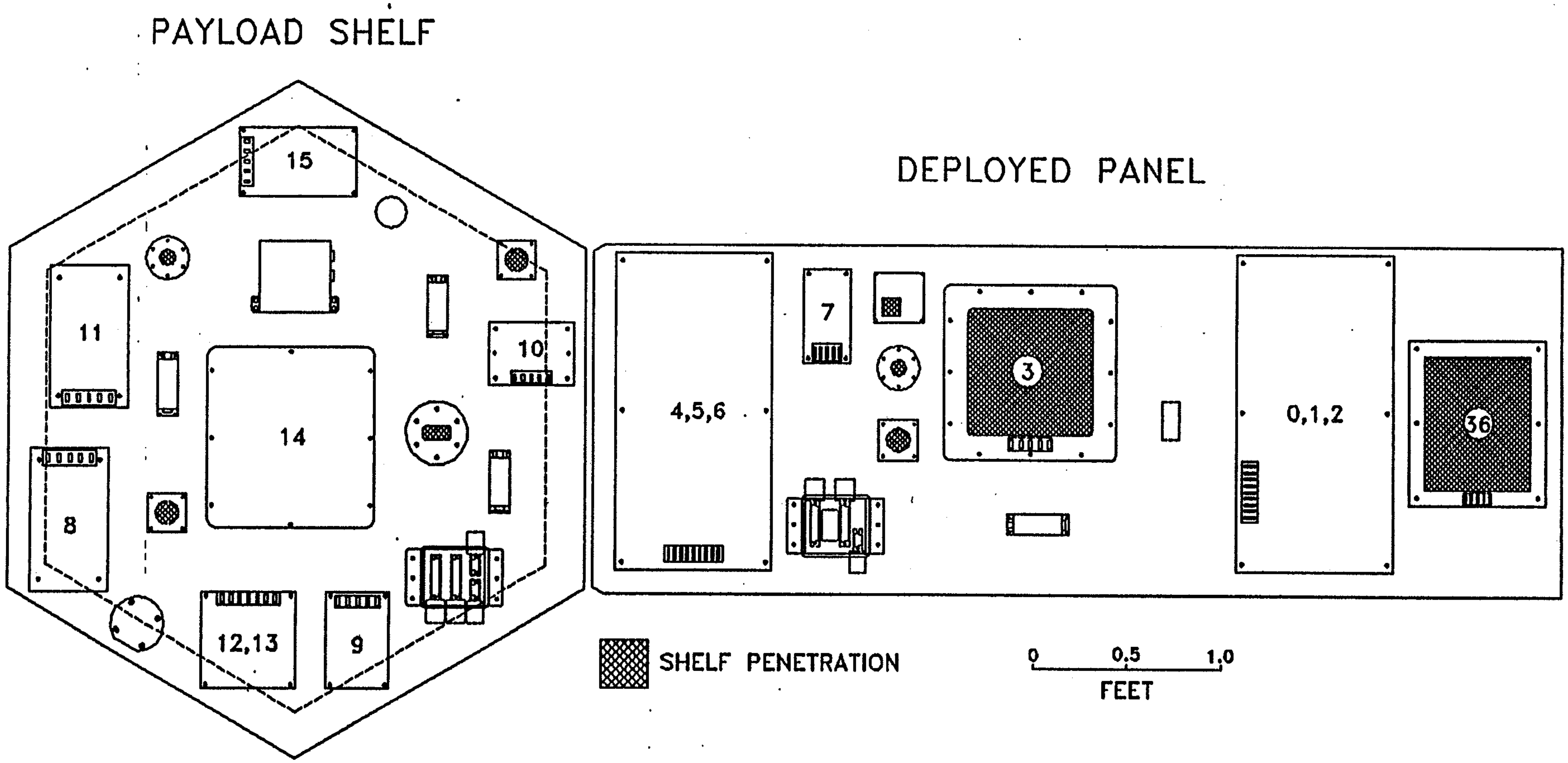
PASP PLUS CONSISTS OF 12 DIFFERENT EXPERIMENTAL MODULES (OBTAINED FROM VARIOUS DOD, NASA & INDUSTRY SOURCES) ALONG WITH A VARIETY OF ENVIRONMENTAL AND DIAGNOSTIC SENSORS

SOLAR CELL MATERIALS

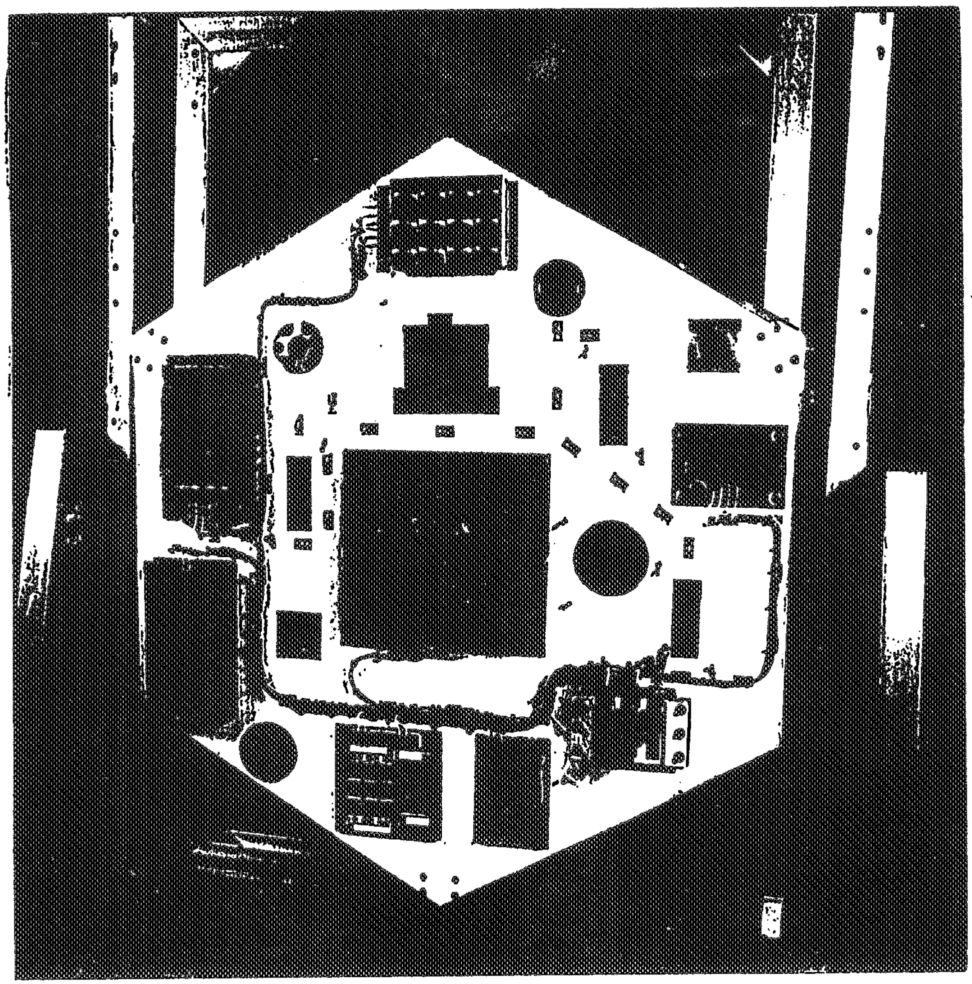
SILICON (Si)
GALLIUM ARSENIDE (GaAs)
INDIUM PHOSPHIDE
AMORPHOUS SILICON
GaAs/CIS TANDEM CELLS
GaAs/GaSb TANDEM CELLS
AlGaAs/GaAs MONOLITHIC CELLS

ARRAY DESIGNS/CONFIGURATIONS

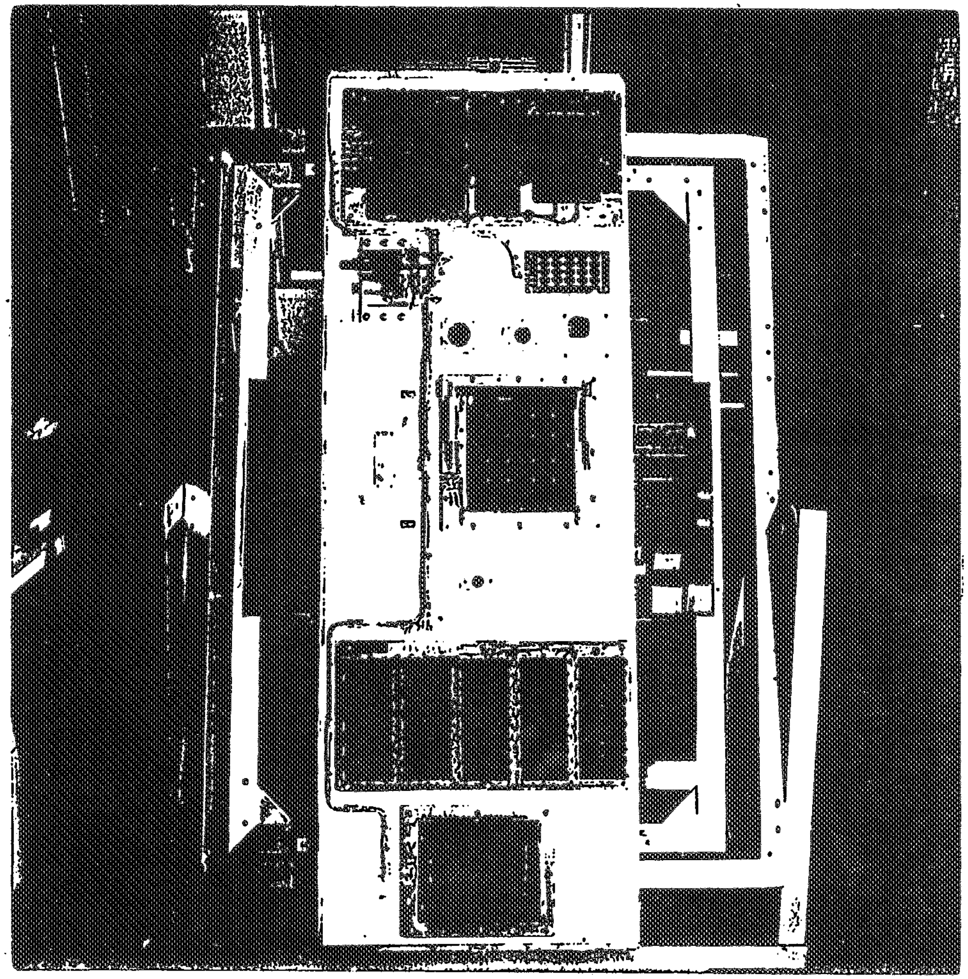
STANDARD Si & GaAs CELL CONFIG.
GaAs WRAP-THROUGH CONTACTS
SPACE STATION FREEDOM DESIGN
ADVANCED PHOTOVOLTAIC SOLAR ARRAY (APSA) DESIGN
MINI-CASSEGRAINIAN CONCENTRATOR
MINI-DOME FRESNEL LENS CONC.


PAYLOAD SHELF
15
11
10
14
8
12,13
9
DEPLOYED PANEL
4,5,6
7
3
0,1,2
36
SHELF PENETRATION
0
0.5
1.0
FEET

PASP PLUS FLIGHT HARDWARE



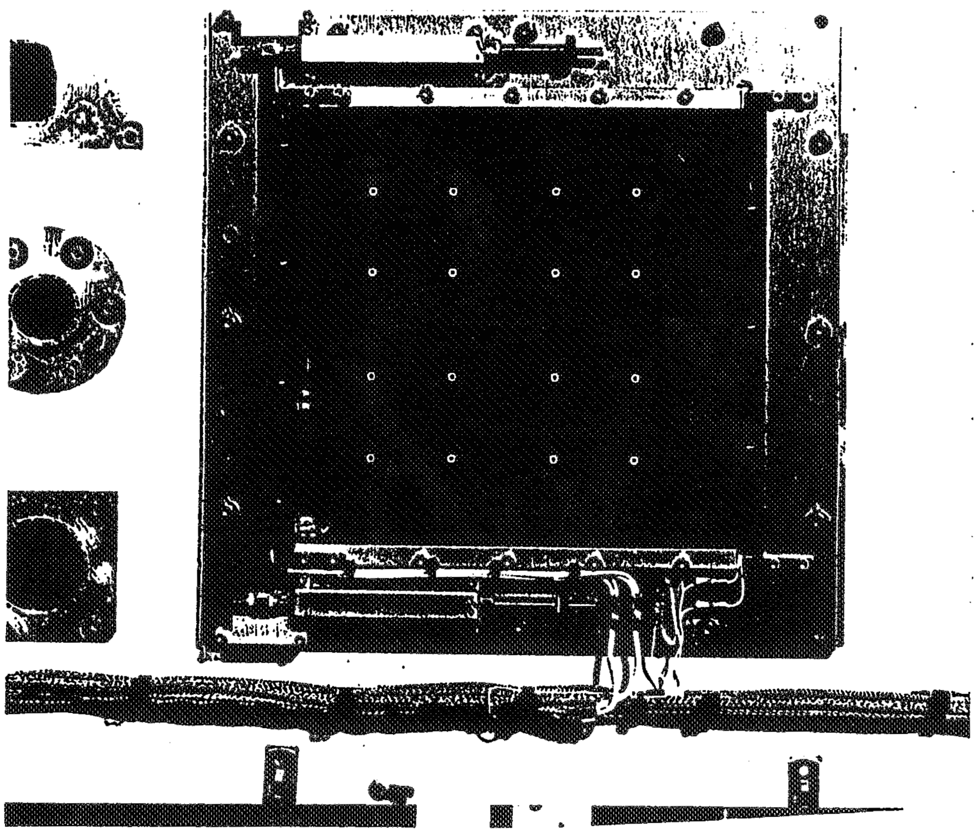
Payload Shelf



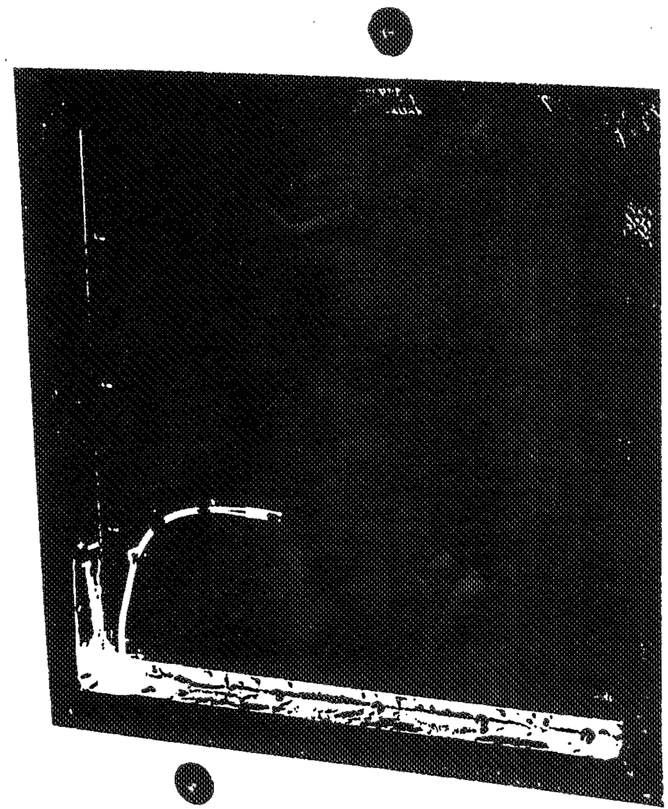
Deployed Panel

PASP PLUS FLIGHT EXPERIMENT

(Experiments relating to NASA-sponsored technology)



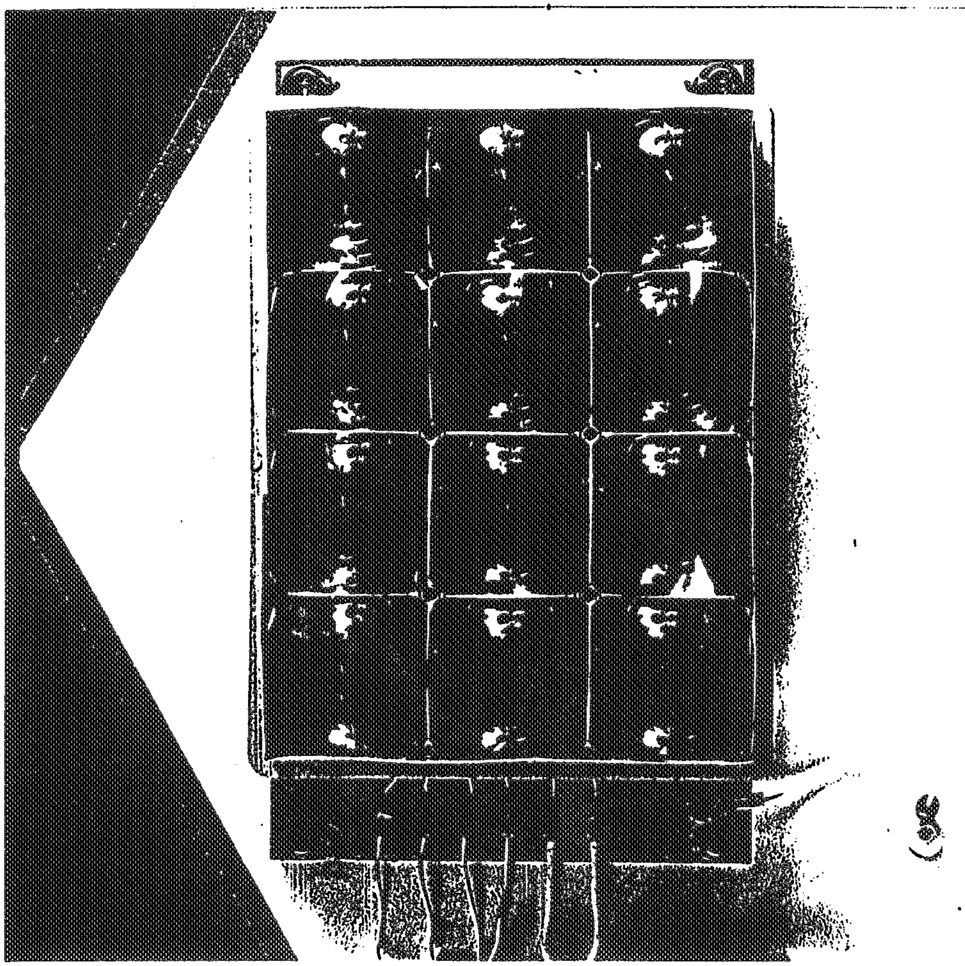
Front



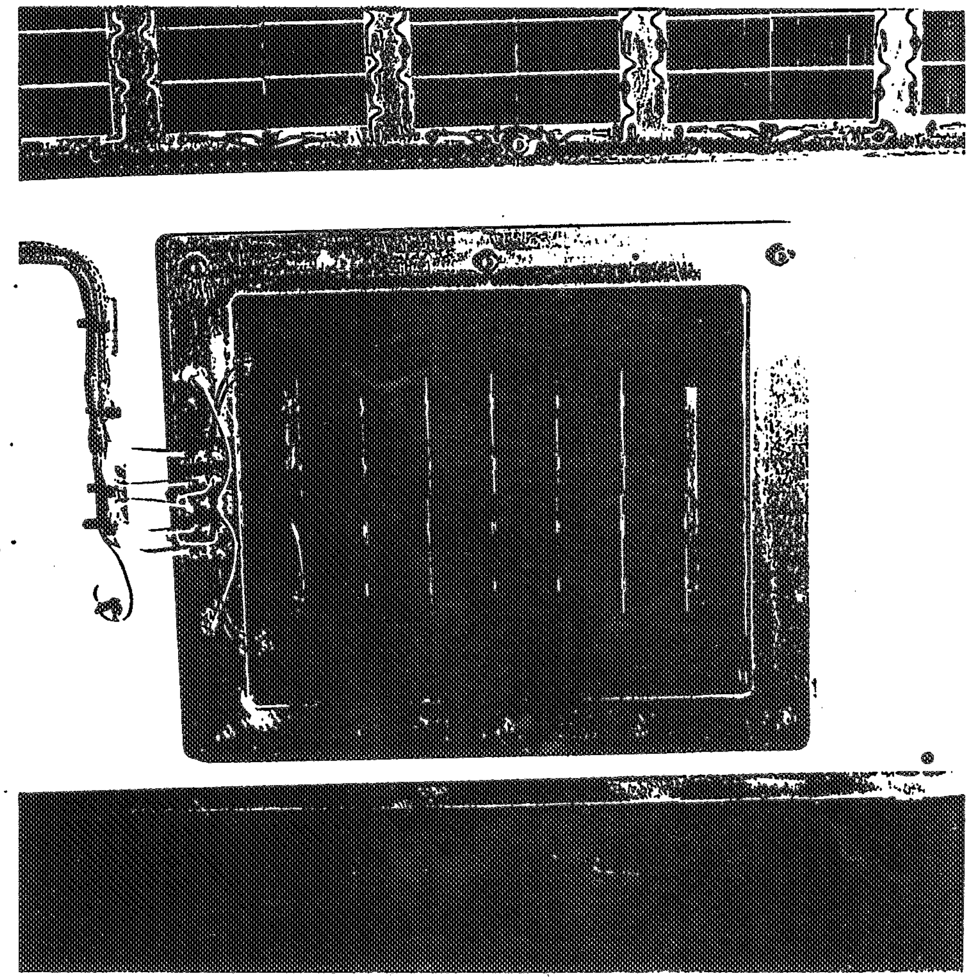
Back

PASP PLUS FLIGHT EXPERIMENT

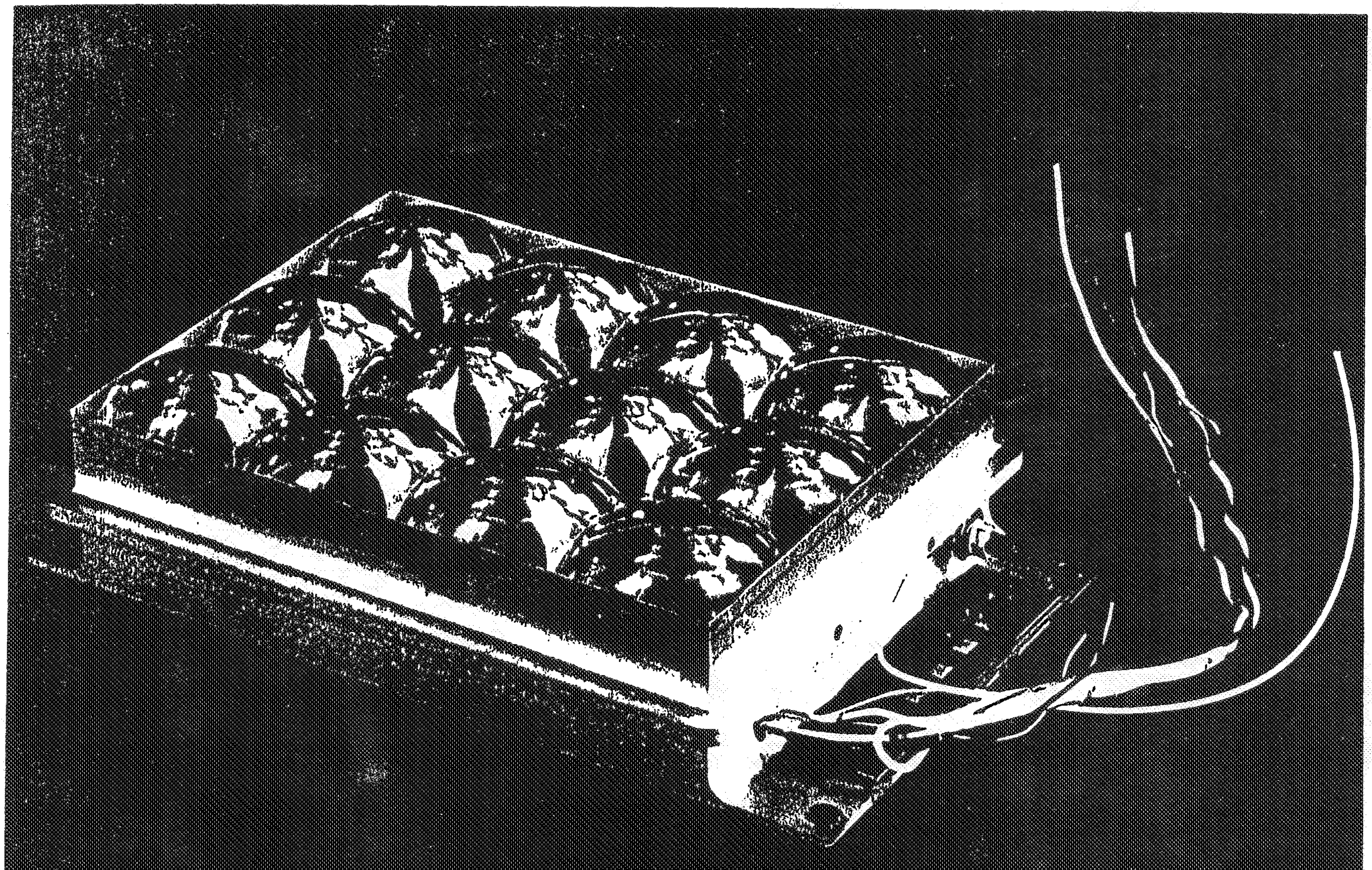
(Experiments relating to NASA-sponsored technology)

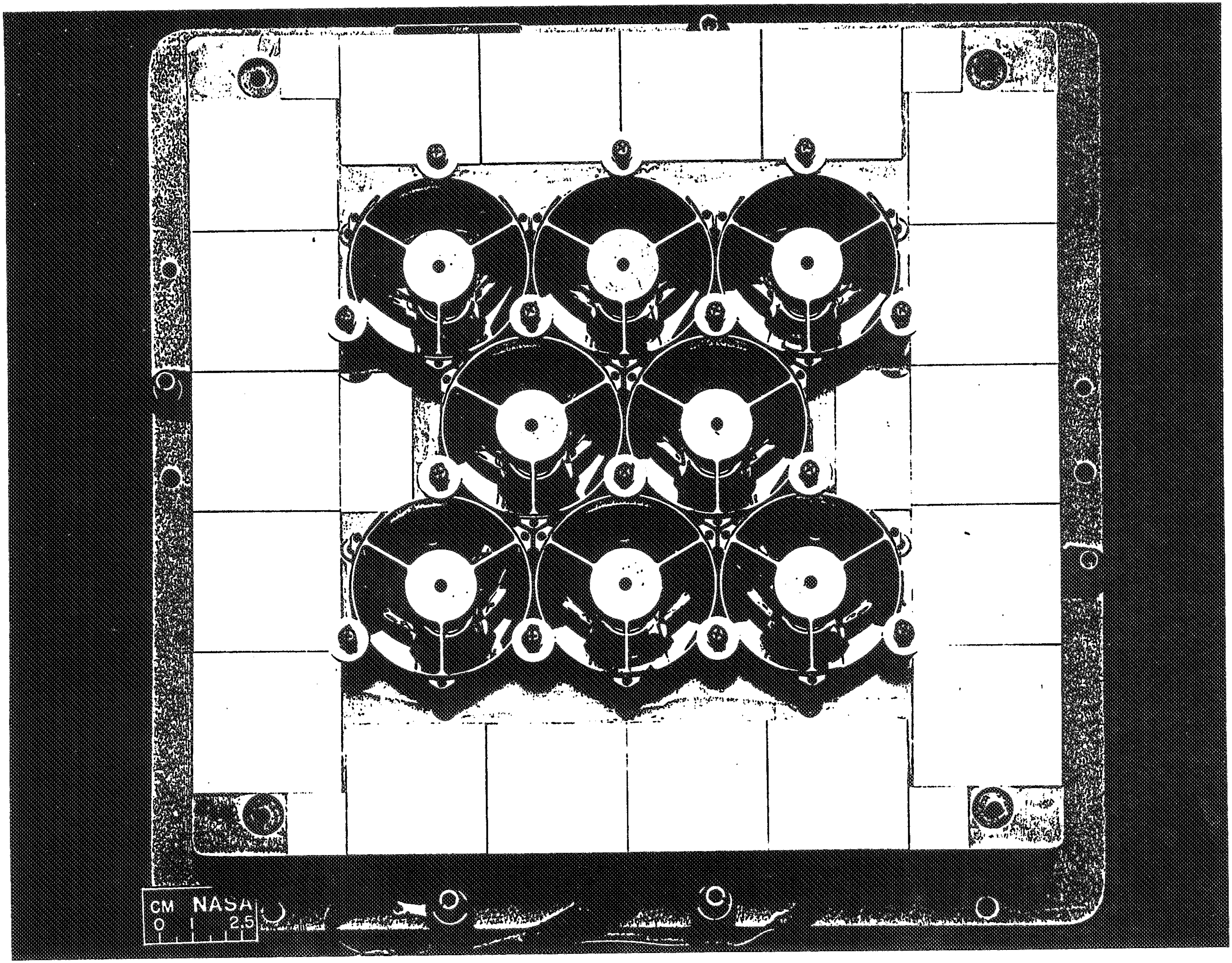


Mini-Dome Fresnel Lens
Photovoltaic Concentrator



Advanced Photovoltaic
Solar Array (APSA)




CM NASA
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EXPERIMENT-CONTROL INSTRUMENTATION

- MULTI-STEP HIGH-VOLTAGE GENERATOR
  - Four All-Positive or All-Negative Steps
  - Each Step about 26 sec Long
  - Bias-Value Range: 50 to 500 Volts
  - Minimum Step-Value Separation: 10 Volts

- ELECTRON EMITTER
  - Generate and Accelerate Outgoing Electrons
  - Help Balance Excess Incoming Electrons
  - Reduce Negative Vehicle-Frame Potential

PASP PLUS DIAGNOSTIC SENSORS

- **LANGMUIR PROBE**
  Measure Plasma Density and Temperature
- **ELECTROSTATIC ANALYZER**
  Detect Passage through Auroral Region
- **ELECTRON/PROTON RADIATION DOSIMETER**
  Outputs: Dose (Energy Deposited) and Flux
  Energy Ranges: Four for Electrons, Eight for Protons
- **CONTAMINATION MONITORS**
  QCMs and Calorimeters
- **SUN INCIDENCE-ANGLE SENSOR**
  Assure Concentrator Arrays Are Aligned

INTERACTIONS MEASURING INSTRUMENTATION

- **TRANSIENT PULSE MONITOR (Negative Biasing)**
  - Measure Electrical Characteristics of Arc Pulses
    Amplitude, derivative, integral, number per time interval
  - E-Field Sensors on Upper Deck and Deployed Panel
  - Current-Loop Sensor on High-Voltage Line
- **LEAKAGE-CURRENT ELECTROMETER (Positive Biasing)**
  - Electron Current, 1 μA to 20 mA
- **CURRENT – VOLTAGE MEASUREMENTS**
  - All 16 Array Modules (Biased or Not)
  - 64 Resistance Values from $R = \infty$ to $R = 0$
    corres. to open-circuit voltage V_{OC} and short-circuit current I_{SC}

PASP PLUS FLIGHT EXPERIMENT
ILLUMINATED THERMAL-VAC TESTING

TESTING CONDUCTED AT BOEING FACILITIES - JUNE 1992

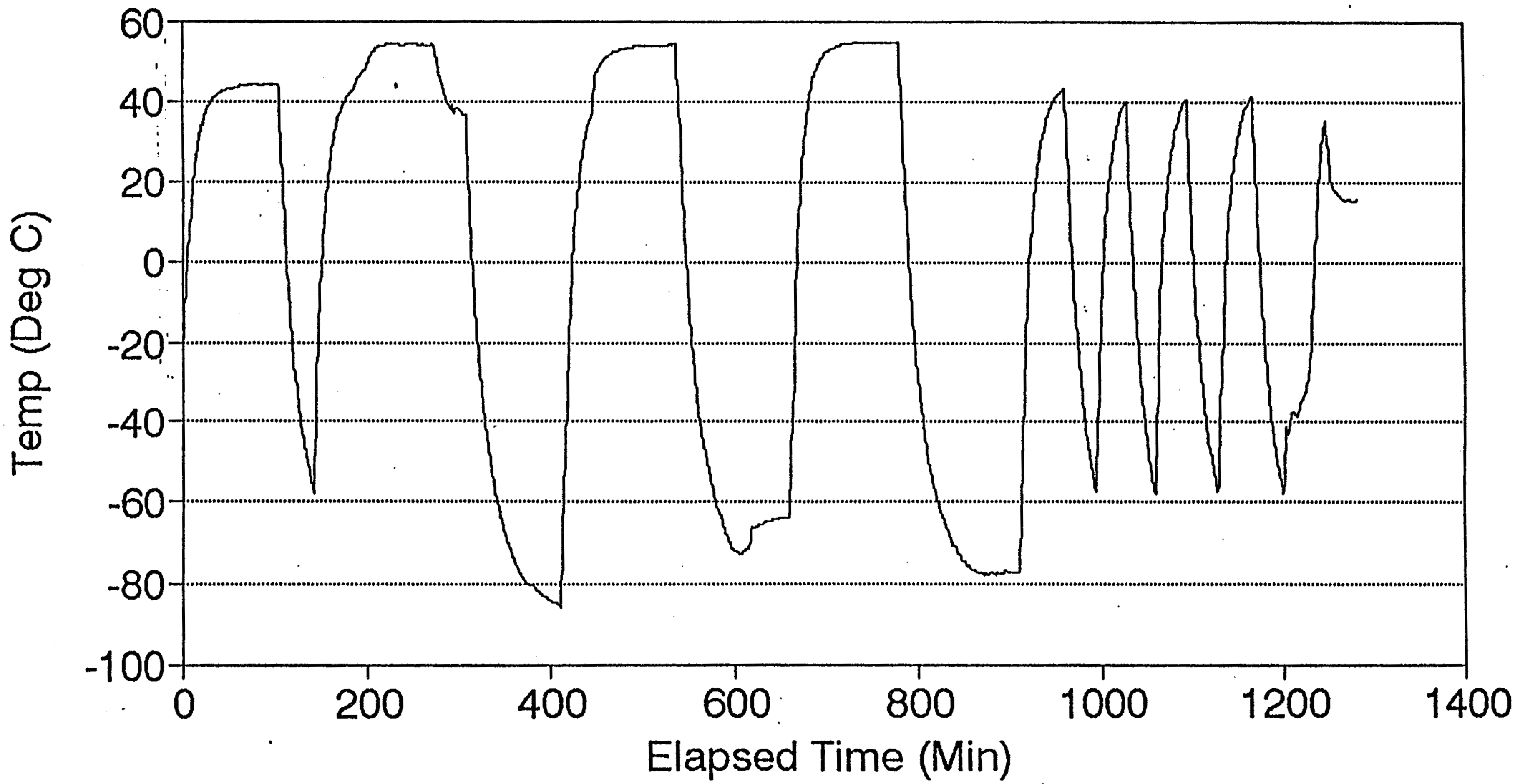
- DEPLOYED PANEL, PAYLOAD SHELF & ELECTRONIC CONTROLLER
- THREE INDIVIDUAL RUNS NEEDED TO COMPLETE TESTING

WHY WAS TESTING DONE?

- SIMULATE TEMPERATURE RANGE OF PANEL & ARRAYS UNDER FLIGHT CONDITIONS
- OBTAIN MODULE I-V CURVES AT VARIOUS TEMPERATURES
- NEED FOR END-TO-END TEST

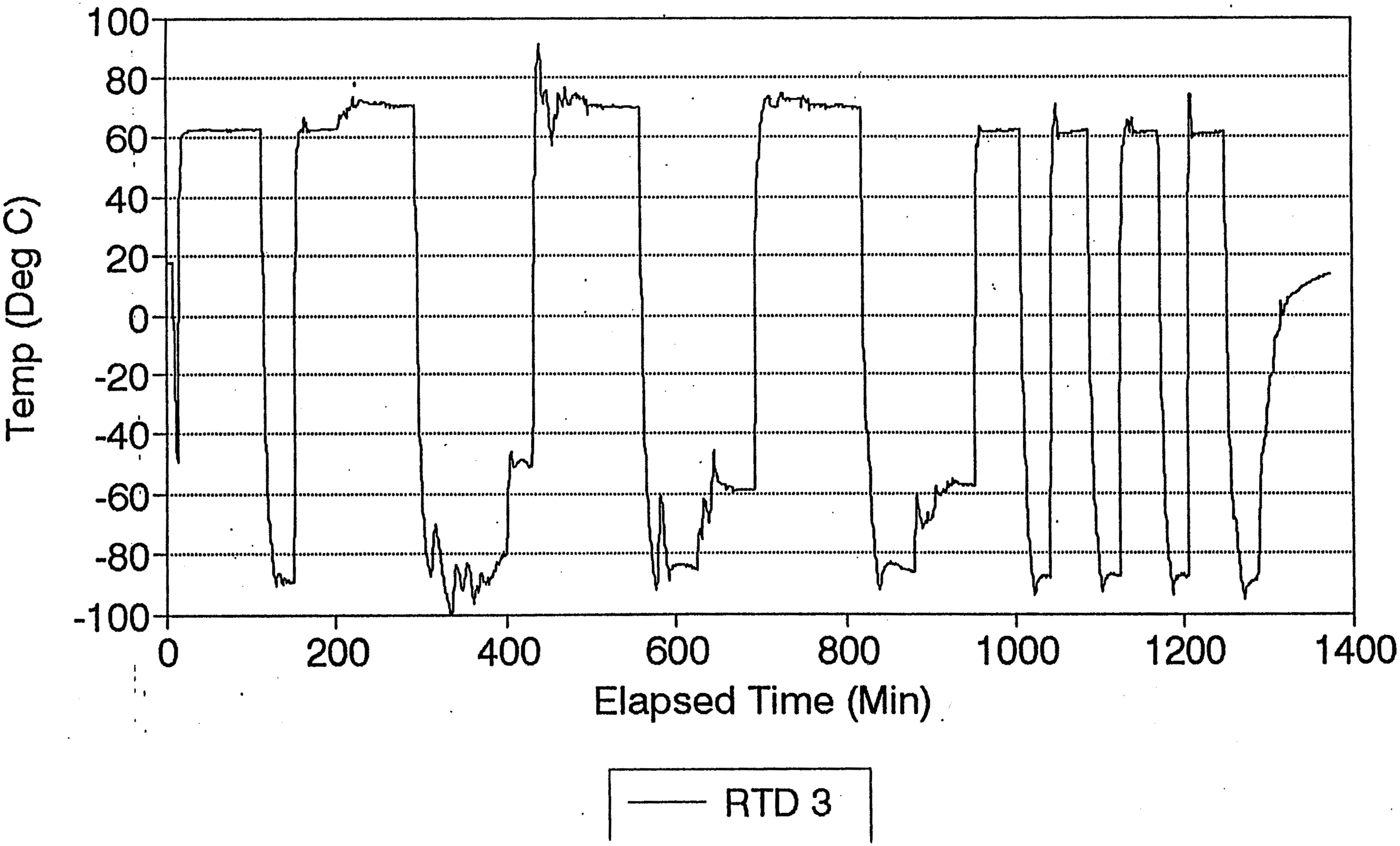
PASP PLUS FLIGHT EXPERIMENT
ILLUMINATED THERMAL-VAC TESTING

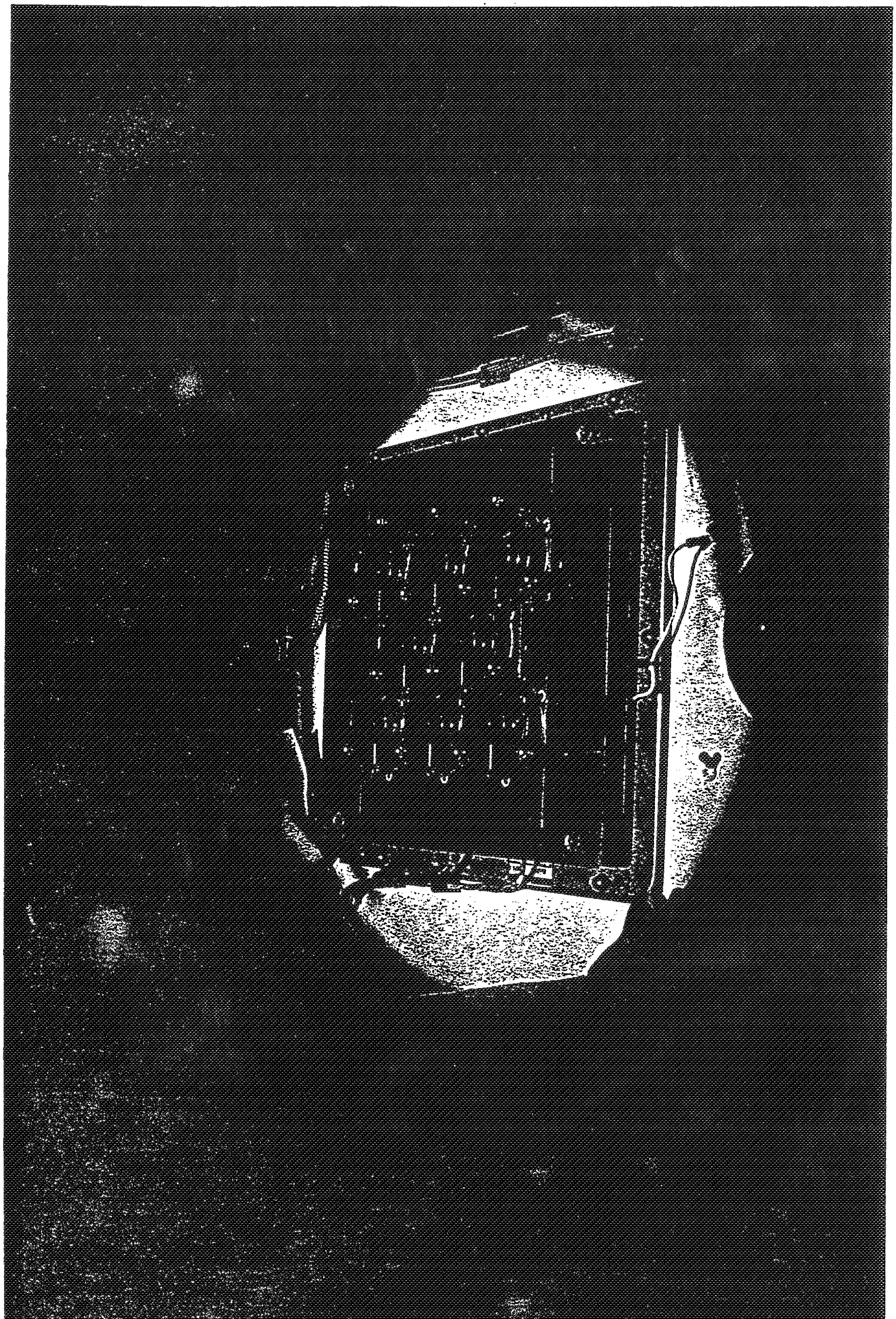
PV MODULE TEMPERATURE PROFILE (Array #0,1,2)



PASP PLUS FLIGHT EXPERIMENT
ILLUMINATED THERMAL-VAC TESTING

PV MODULE TEMPERATURE PROFILE (Array #3)





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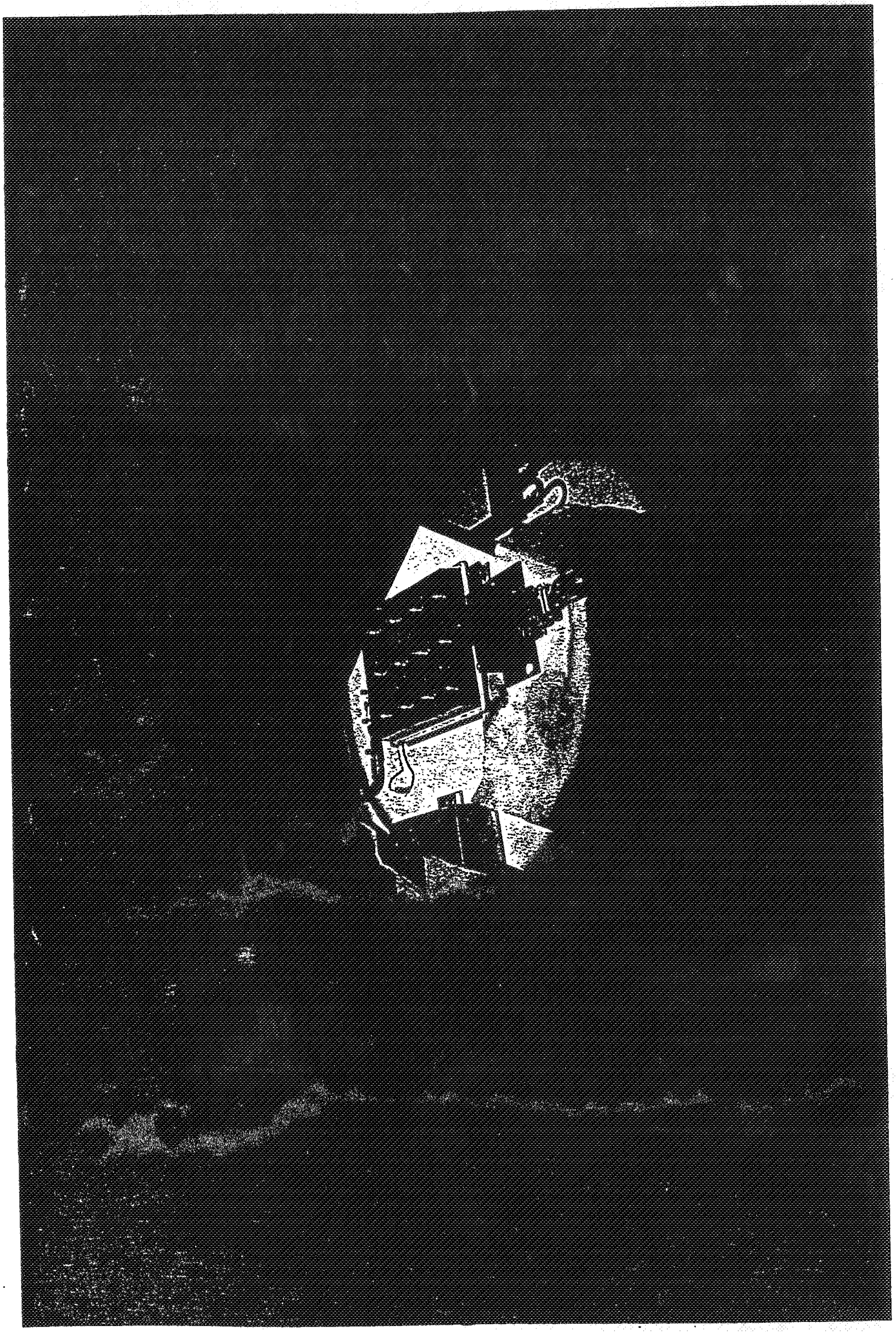

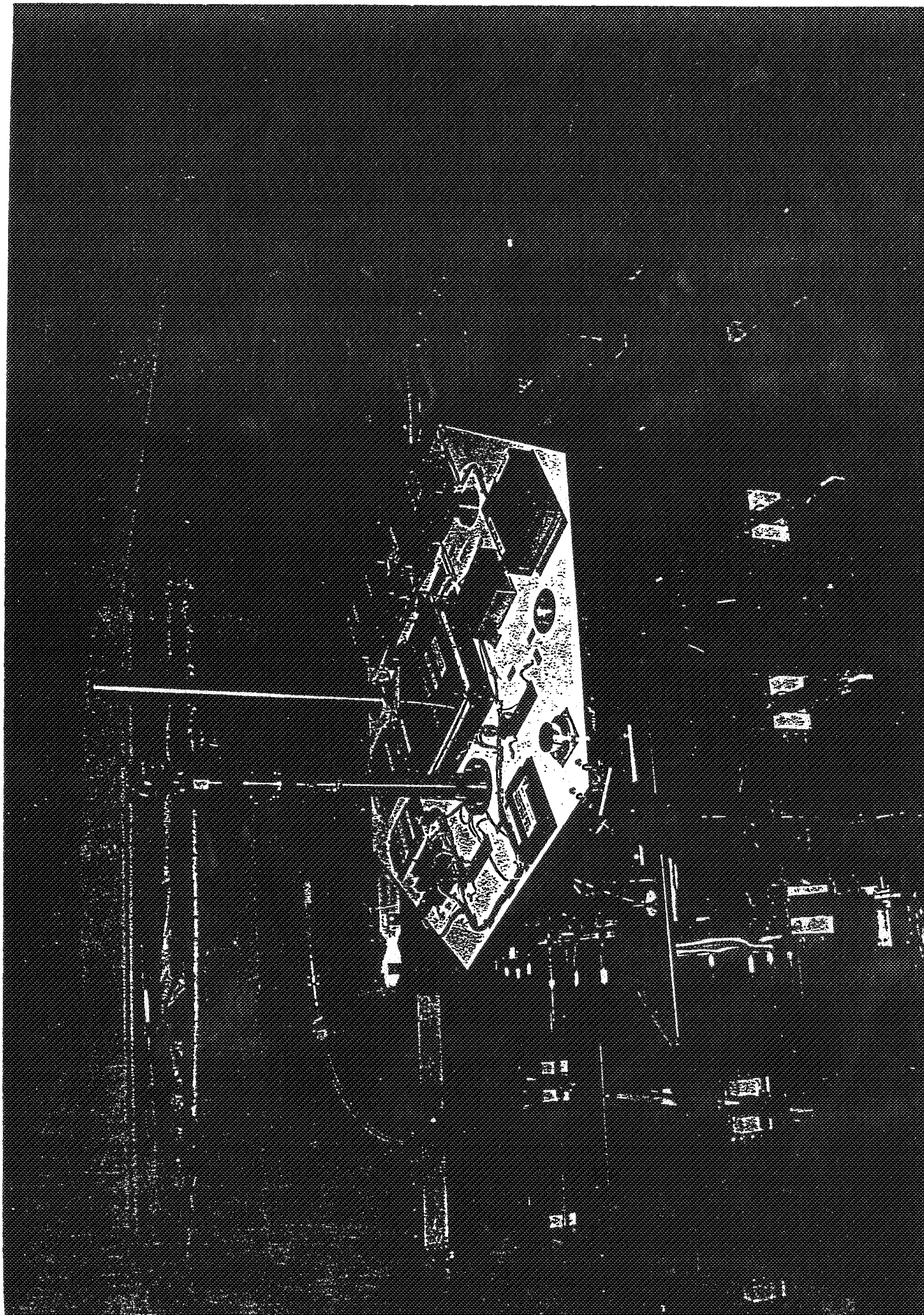

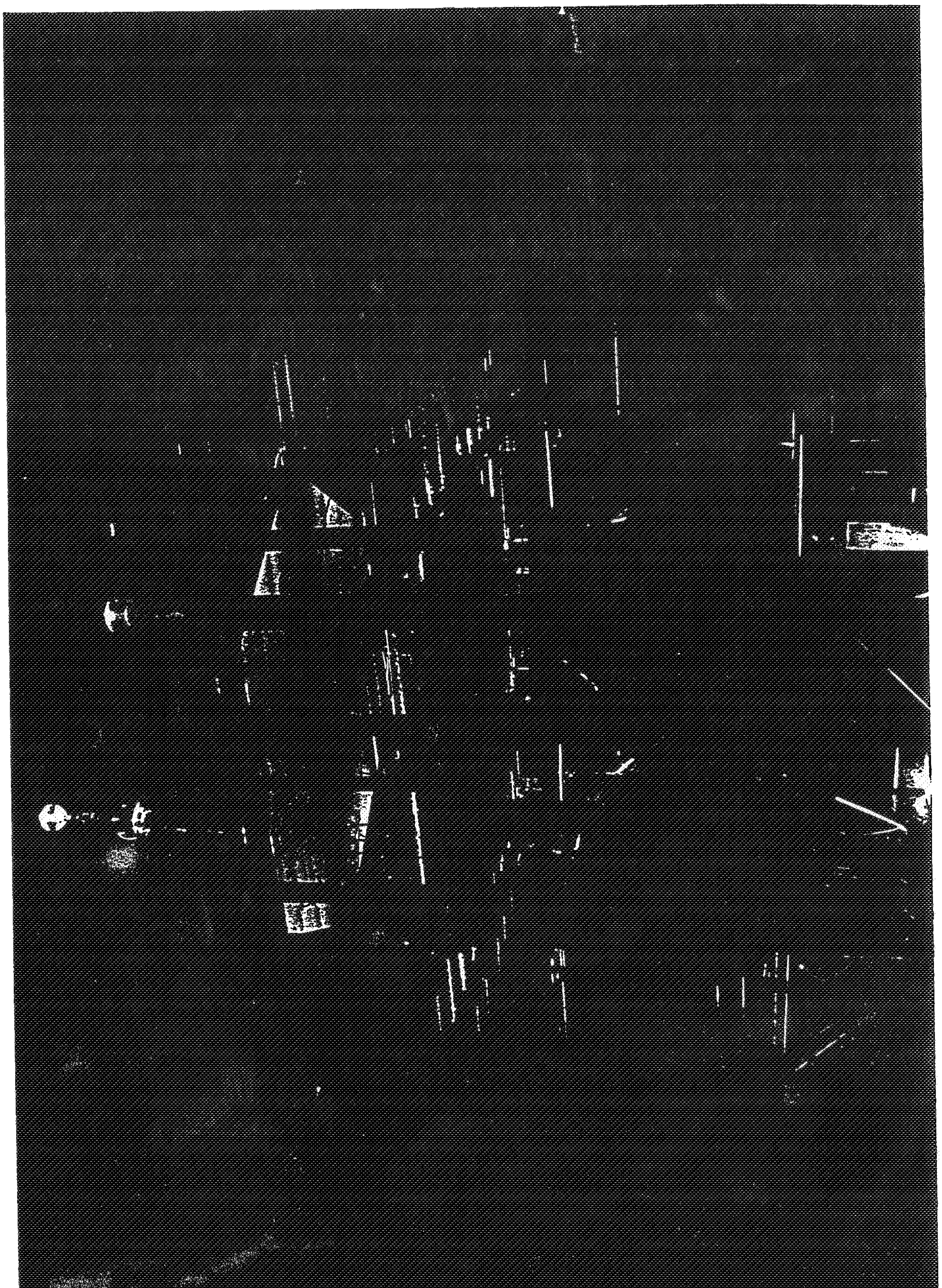

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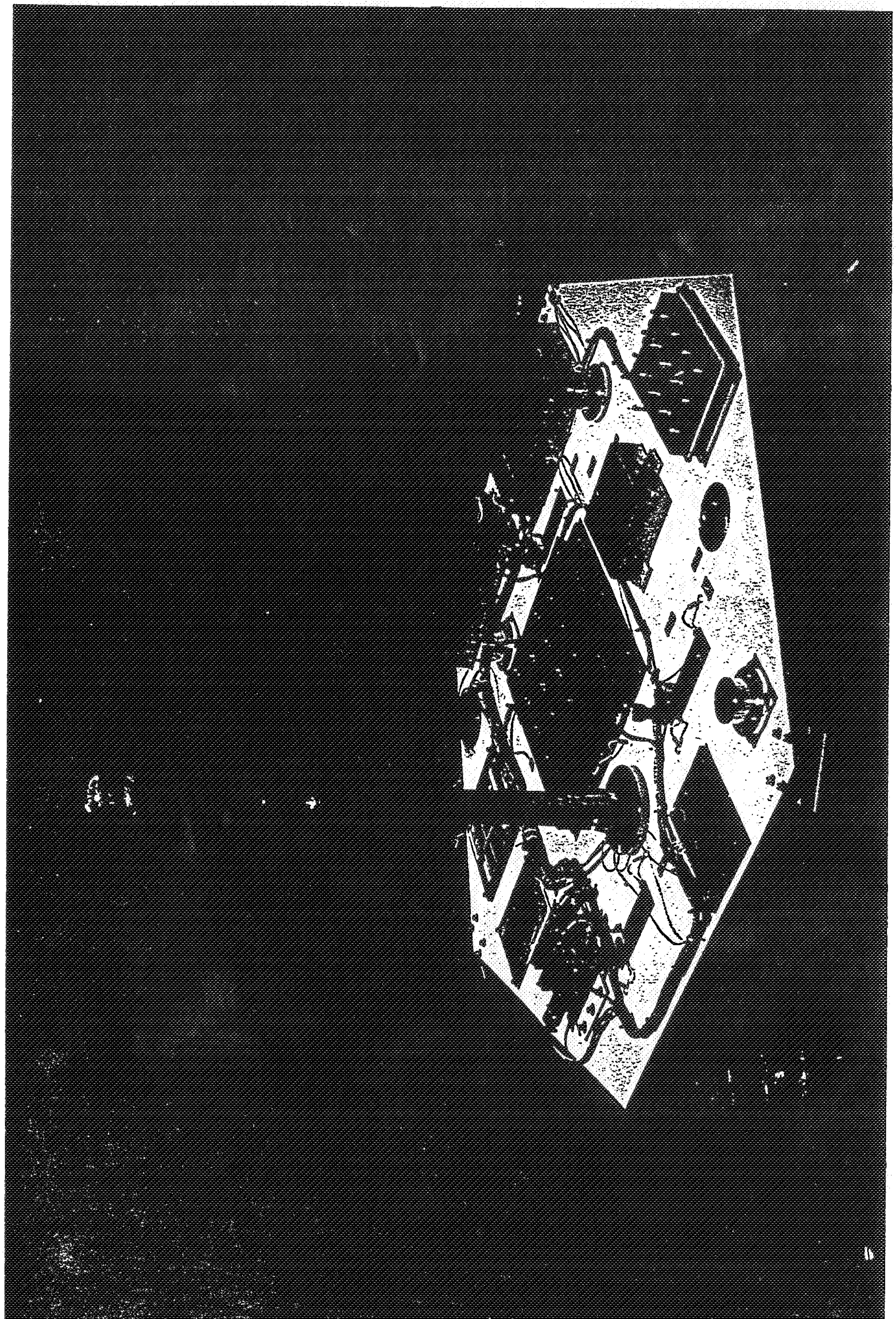

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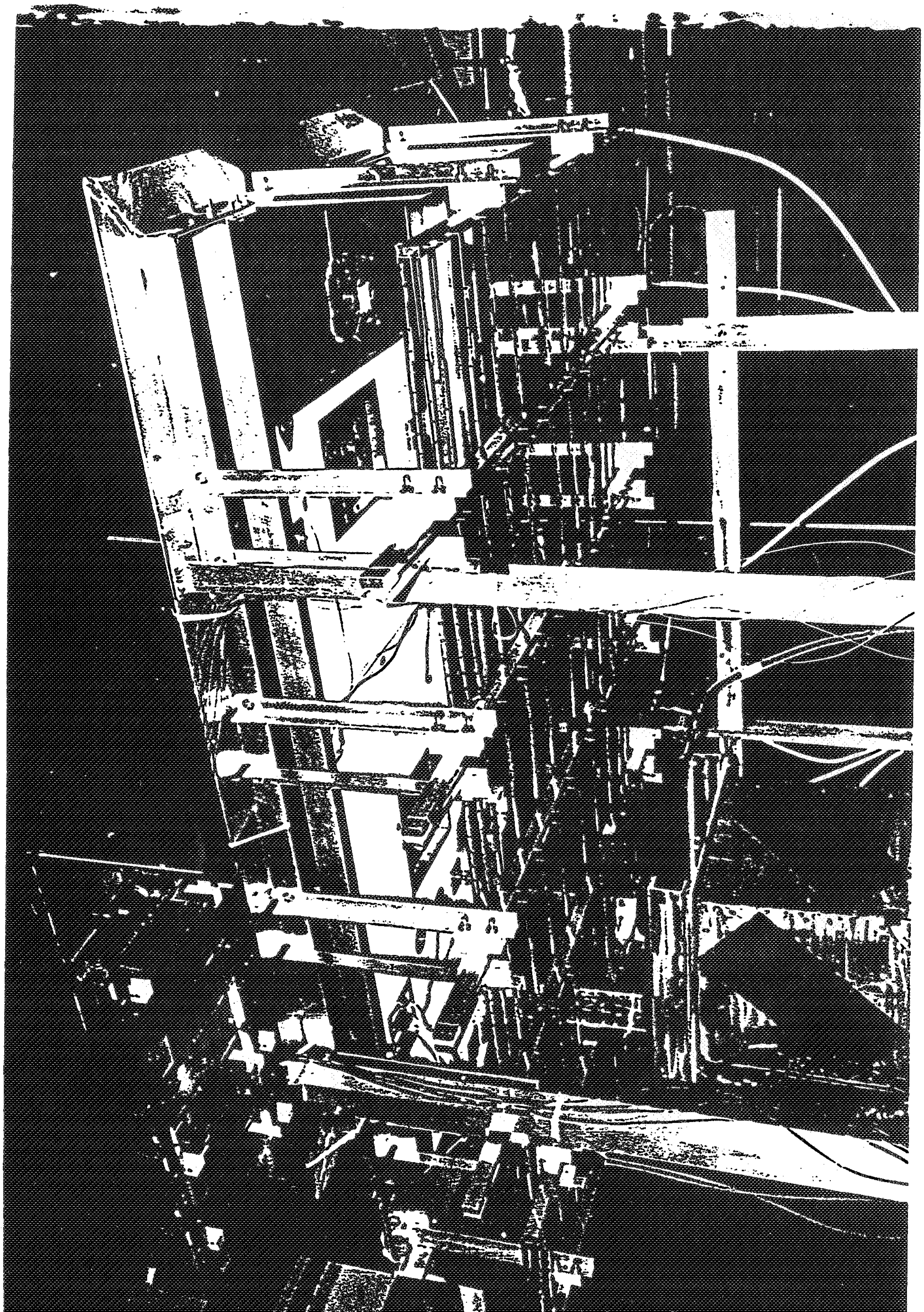
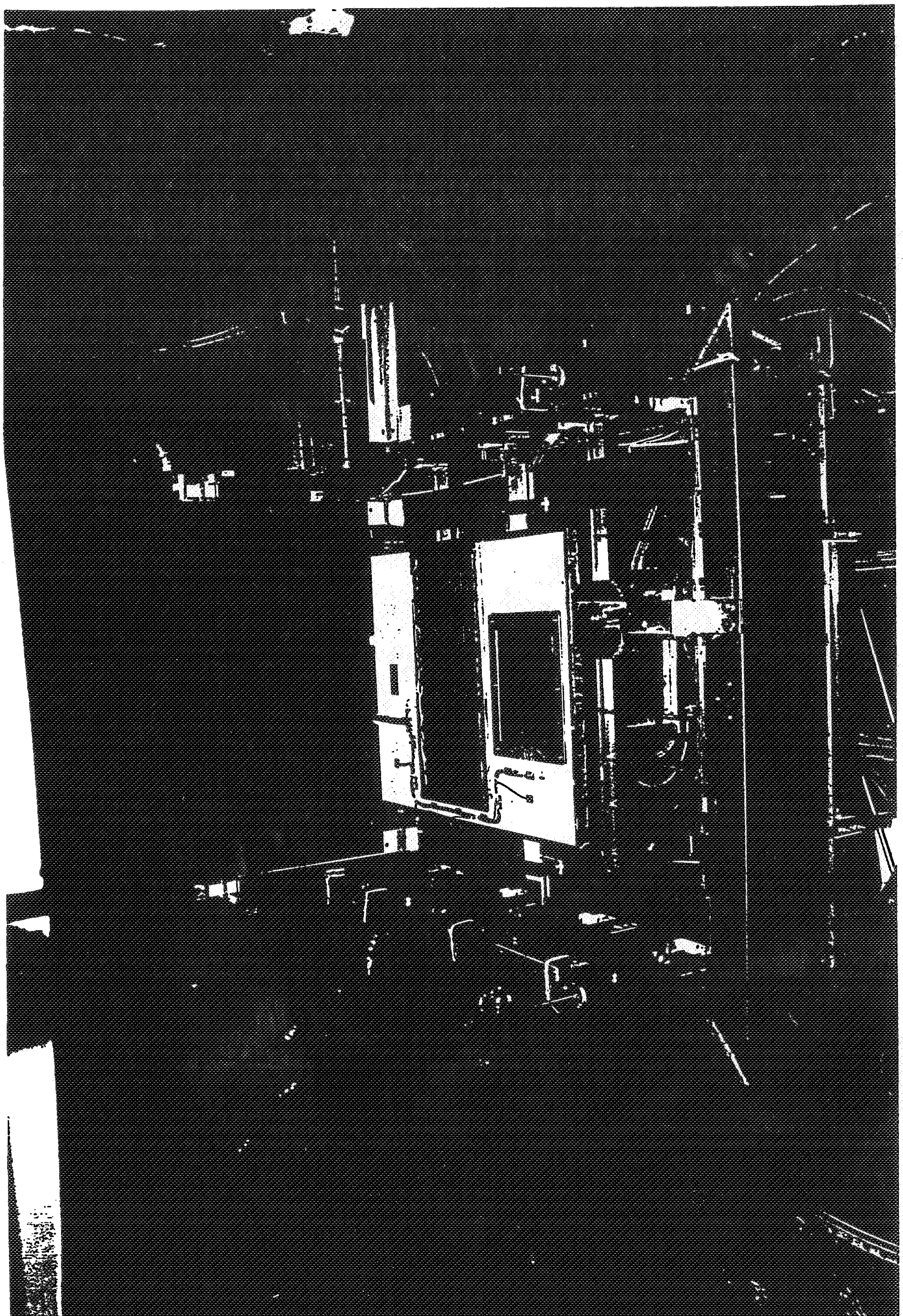
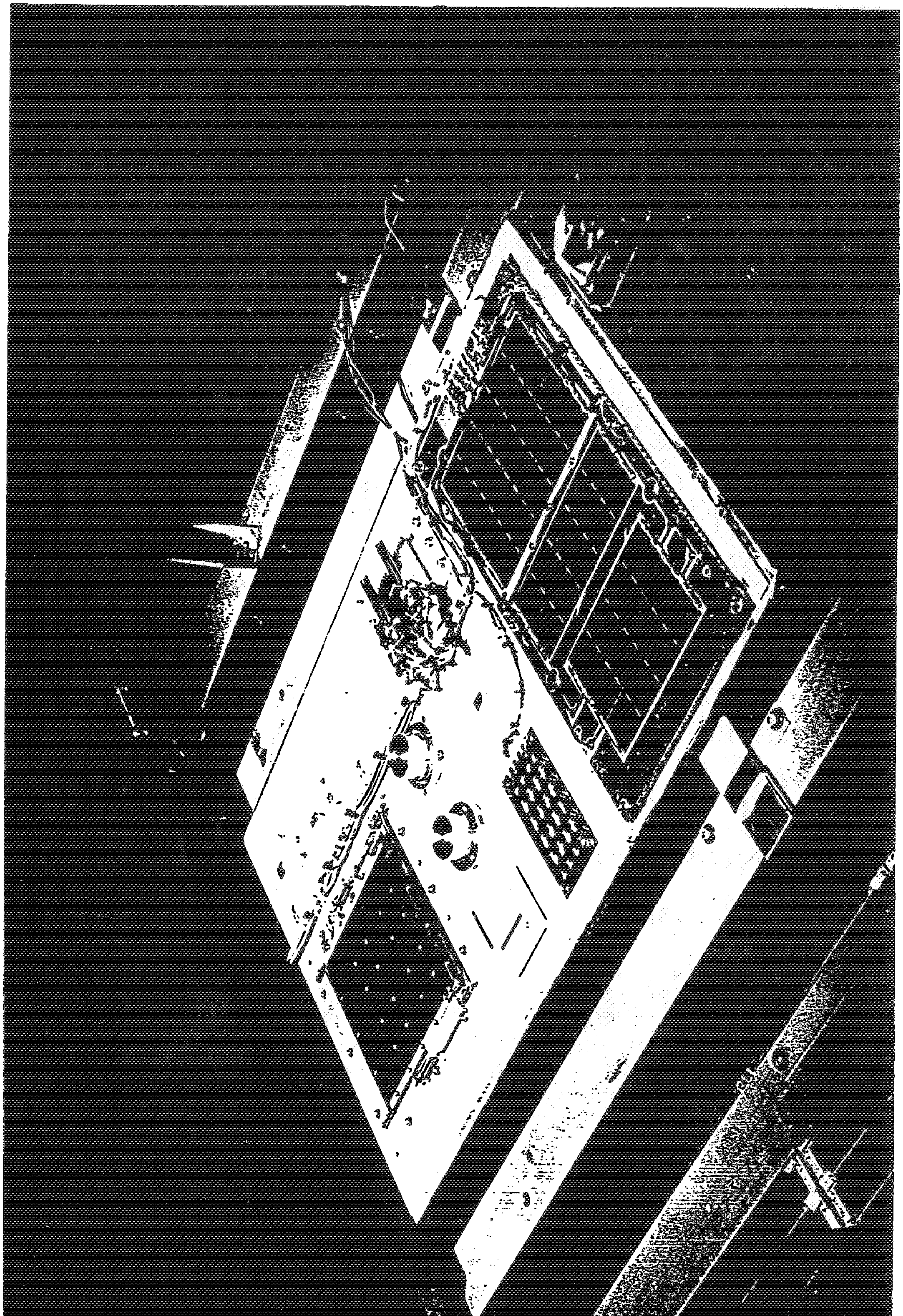
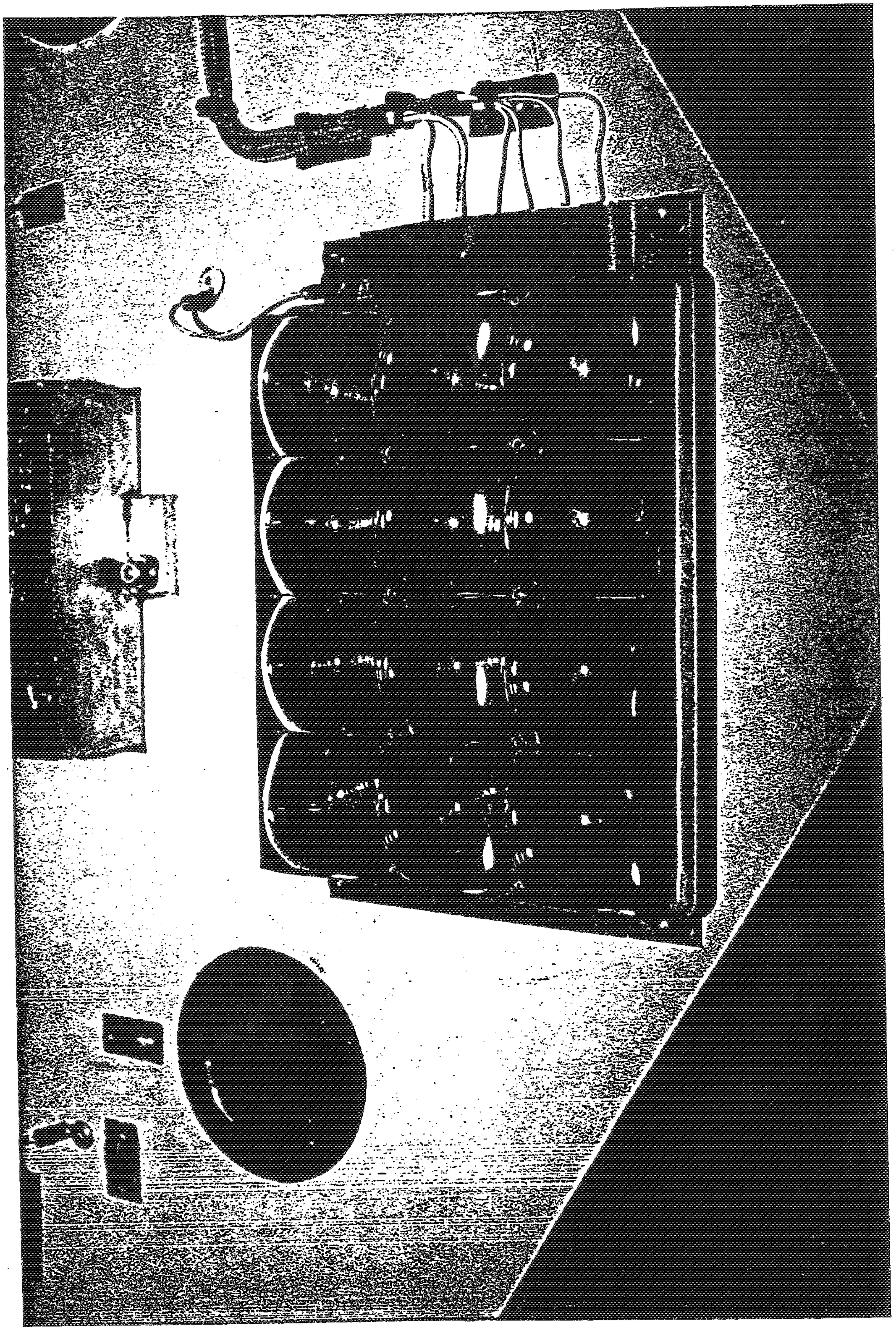


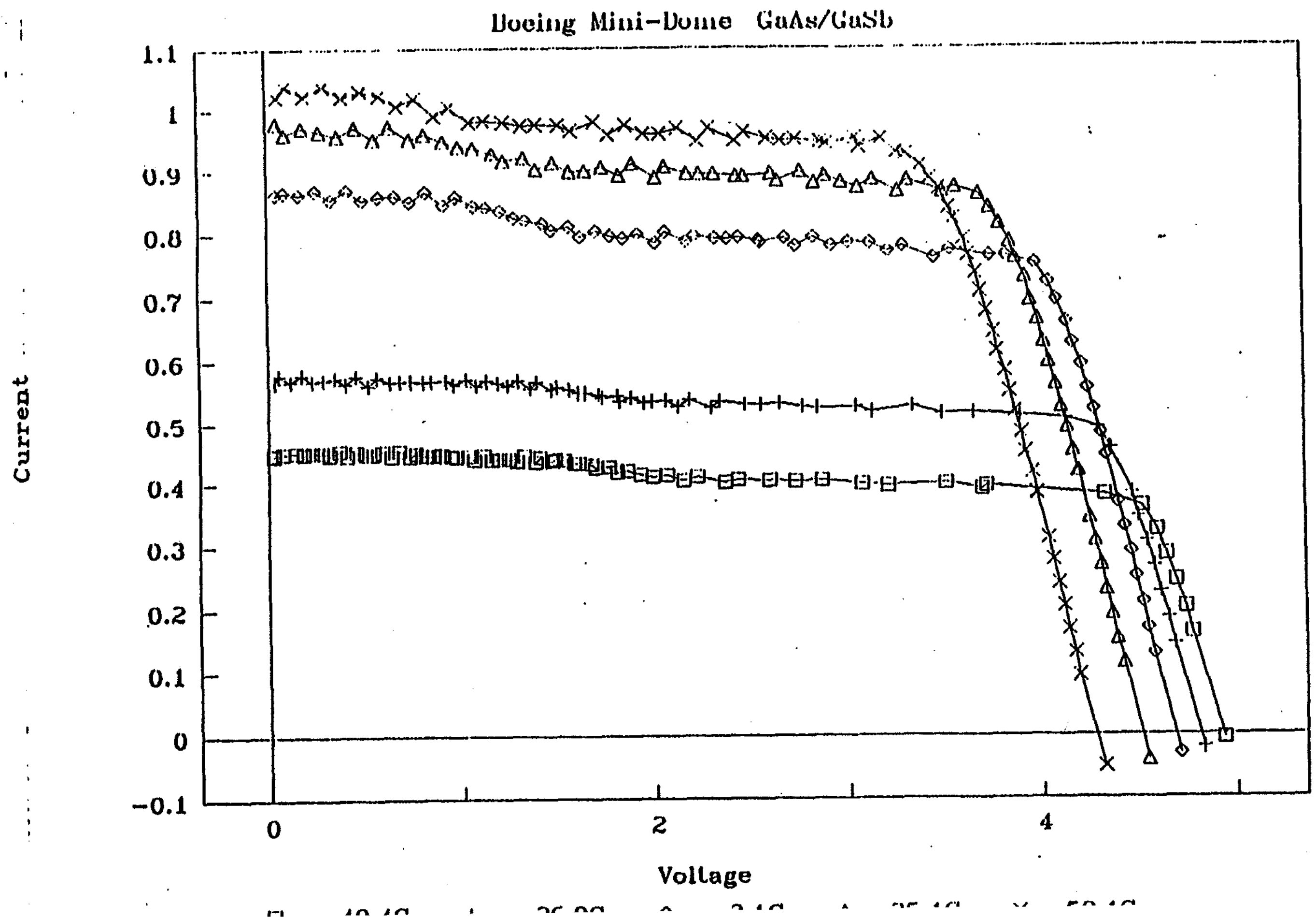






PASP PLUS FLIGHT EXPERIMENT
ILLUMINATED THERMAL-VAC TESTING

I-V MEASUREMENTS (Array #15)



PASP PLUS FLIGHT EXPERIMENT SUMMARY

PROGRAM STATUS

- PHOTOVOLTAIC MODULES INTEGRATED TO FLIGHT PLATES
- ILLUMINATED THERMAL-VAC TESTING COMPLETED
- PASP PLUS EXPERIMENT DELIVERED TO ORBITAL SCIENCES CORP. FOR INTEGRATION TO APEX SATELLITE
- PEGASUS LAUNCH - MAY 27, 1993

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Sodium Sulfur Technology Program NaSTEC

Lt Bob Highley, PL/VTPC

W. Andrew Somerville, Aerospace Corp.

The Phillips Laboratory Conventional Space Power Branch is responsible research and development of solar power generation, power management and distribution, and energy storage systems for the Air Force. Some of the technologies currently being investigated are thin film and multi-band gap solar cells, polymer based cells, and sodium sulfur technology cells. The NaSTEC program focuses on developing currently available sodium sulfur cells for use in space applications and investigating the operational parameters of the cells.

NaSTEC
Sodium Sulfur Technology Program
Program Goals

The goals of the NaSTEC program pertain to Na|S technology from US and UK vendors and will:

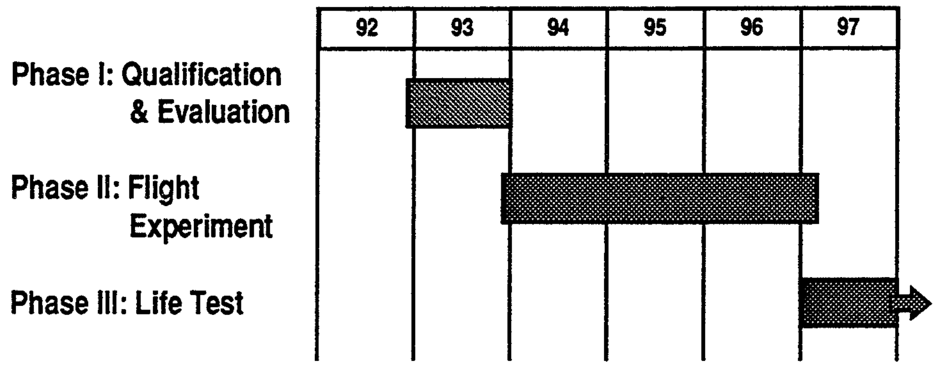
1. **Determine the operational parameters and verify safety limits of Na|S technology battery cells.**
2. **Test long term zero-g operation.**
3. **Create a life test database**

These efforts will ultimately make technology transfer possible for Na|S, thus enhancing current missions and enabling future missions.

The NaSTEC program will test cell samples from US and UK vendors to establish parameters of state of the art cells. The US cells offer high energy density as compared to standard NiH_2 cells, with a higher risk in operation. Cells from the UK offer a lower energy density than US cells, but with a higher reliability. The program will evaluate the pros and cons of both systems with regard to the goals outlined above and this information will be used to design advanced power systems for current and future missions.

NaSTEC
Sodium Sulfur Technology Program
Approach

The program is divided into three phases:



NaSTEC is comprised of three phases, each phase investigates a different aspect of cell research. Phase I concentrates on normal operational parameters and the upper and lower limits of both cell systems. Phase II will determine the effects zero-g, both immediate and long-term effects. Phase III will place a large population of NaS cells on extended life cycle test to build a database of cell operational data. Each of these phases is a self-sufficient program, but is designed to complement the other programs. This allows for flexibility in program planning and program management.

NaSTEC
Sodium Sulfur Technology Program
Phase I : Testing

The purpose of Phase I is to:

- **Determine normal operational parameters of Na|S cells**
- **Establish safety parameters for Na|S cells**
- **Develop an understanding of failure modes and internal operation of Na|S cells**

Research is to be performed by government agencies

- **Phillips Laboratory, Conventional Space Power**
- **Sandia National Laboratory, Storage Batteries Div.**
- **Naval Surface Warfare Center, Crane Div.**

This phase will produce an unbiased evaluation of both US and UK technologies

Phase I will test NaS cells both under normal operating conditions, and under extreme operating conditions. The results of the tests will give power system engineers more information about NaS cells and how to incorporate them into their power systems. The tests will give more insight into the failure mechanisms of NaS cells and operating procedures that will optimize cell performance. The tests are to be performed at two independent test facilities, Sandia National Laboratory , and Naval Surface Warfare Center, Crane Division. These government facilities ensure a unbiased evaluation of the two systems and expertise in the area. This phase should last one to one and a half years to produce the final results.

NaSTEC
Sodium Sulfur Technology Program
Phase I : Testing

Life Testing:	**Create a database of cell cycle data (C)**
Freeze/Thaw:	**Determine optimum freeze/thaw profile (S)**
Mech. Fract& Operation:	**Simulate shipping damage & subsequent operation (S)**
Overcharge:	**Charge cells past cutoff voltage and observe cell behavior (C)**
Overdischarge:	**Discharge cells past cutoff voltage and observe cell behavior (C)**
High Rate C/D:	**Determine the rate capability of the cells (C)**
Short Circuit:	**Observe cell behavior during a short circuit condition (S)**
Overtemperature:	**Determine the temperature limits of the cells (S)**
Vibrational Testing:	**Evaluate Na|S cells' hot launch capability (S)**

(C)=Crane (S)=Sandia

The tests have been designed to examine all aspects of cell operation with a minimum of duplication while maintaining statistical reliability of the data. Each facility will perform tests that best suit their particular capabilities. Each of the tests simulate some of the situations that the NaS cells might see in space, both normal and accidental. The goal of each test is outlined above and individual test plans will be formulated by the test facilities.

NaSTEC
Sodium Sulfur Technology Program
Phase II : Flight Experiment

The Sodium Sulfur Technology (NaSTEC) Flight Experiment will verify zero-g operation and evaluate NaS Battery performance.

NaSTEC will fly both US and UK NaS cell technology.

NaSTEC will operate a NaS cell module in zero-g with a simulated GEO cycling regime.

- **Verify electrode material transport.**
- **Validate wicking design.**
- **Evaluate zero-g impact on operation.**
- **Evaluate thermal characteristics.**

Phase II is the heart of the NaSTEC program. The flight experiment is designed to verify the zero-g operation of NaS cells and observe any effect on battery performance. This experiment will fly both US and UK cells for complementary testing in Phases I and III. Some of the concerns associated with zero-g operations of NaS cells are with the electrode material transport in the reactive zones of the cells, zero-g degradation of cell performance over a long period of time, and thermal control of the cells without major impact on the rest of the space craft. This will be accomplished by cycling the cells in a simulated GEO cycling regime and analyzing the data as it is collected. The mission has a one year requirement with a three year goal.

NaSTEC
Sodium Sulfur Technology Program
Phase II : Approach

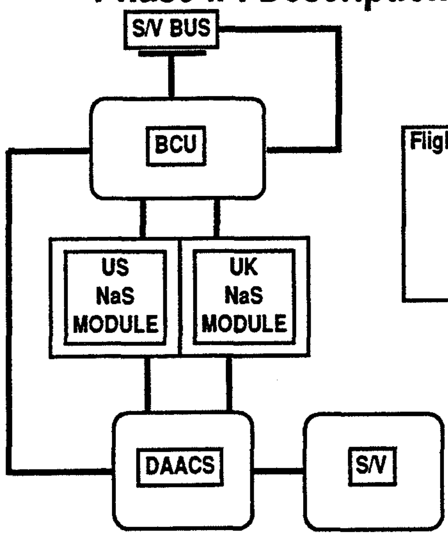
The Flight Experiment will be designed with several factors in mind:

- **Weight and volume to be kept at a minimum**
- **Power and data requirements reduced to moderate levels using charge scheduling and low data rates**
- **Keep costs down by using Class B and/or Proto-Qual hardware and reducing redundancy to a minimum acceptable level**

This approach will produce preliminary data concerning the operation of Na|S in zero-g, thus opening the door for further research and use as prime power on many missions

Experiment design is driven by economy in weight, volume, cost, and power requirements. Construction of the experiment will be contracted out, with the design developed through close interaction between the contractor and the experimenter. Weight and volume will be economized through careful planning of materials used and the structures required. Current capabilities of data collection and recording are more than sufficient for handling the requirements of the experiment, but to cost low on these components, data rates will be reduced to a minimum acceptable level. Power consumption of the experiment can be excessive, if not controlled. The requirement of the experiment package will be reduced through the use of a unique charge control system. The US and UK cells will be divided into two electrically separate cell packs. During cycling, one pack will be discharging and the other charging, using the excess power of the first pack. There will be some power drawn from the satellite bus to account for the losses in wiring, heater power, controller power, and any loss of capacity in either cell pack. This arrangement will help with thermal management and hardware requirements. As NaS cells are discharging, they give off heat. This heat is absorbed by the charging cells to maintain operating temperature with minimum heater power. Hardware is reduced by eliminating the need for external thermal management, and load radiators. All this will help reduce the cost of the experiment. The components to be used in the experiment will be Class B parts and possibly some proto -qual type parts. This may affect the weight, volume, and power budget, but savings in cost will compensate.

NaSTEC
Sodium Sulfur Technology Program
Phase II : Description



Flight Test Unit Requirements	
Orbit:	None
Weight:	27 kg
Volume:	0.1 cubic meters
Power:	80 watts(avg)

The exact requirements of the experiment have not been formalized, but the values listed above are close estimations. The small size and volume will allow for easy incorporation of the experiment into a wide variety of satellites. All efforts will be made to make the interfaces as generic as possible, as to be compatible with the system bus. The simulated cycling regime will make the experiment compatible with any actual orbit the host satellite may be in, though the only hard requirement is that the satellite must not be a spinner or a tumbler. This would induce centripetal force that would affect cell performance, and we would not be able to evaluate the cell's zero-g capability.

NaSTEC
Sodium Sulfur Technology Program
Phase III Life Test

Phase III will commence after operation of NaS cell technology has been demonstrated in zero-g and will develop into a large scale life cycle test

Data from the test will be compiled into a database which will then be used by system engineers to design satellite power systems

Large scale life test will also include experimental accelerated cycle profiles

Test profiles will include:

- GEO Cycling
- Accelerated Cycling
- Pulse Testing
- LEO Cycling
- User Defined Profiles

After the flight experiment is operational on orbit for several months, and initial cell operation is proven, a large scale cycling program will be started. This test will generate data that will be used by power system engineers to design power systems for satellites. The tests will look at all types of operating profiles including accelerated cycling, LEO cycling, GEO cycling, pulse testing, and user defined profiles. The test will last over five years with the possibility of becoming a permanent test. The data generated will be analyzed with respect to the data obtained from the flight experiment with regard to zero-g effects on cell performance. The data then will be compiled into a database for easy manipulation and access. This data will be accessible at first to governmental agencies and to others on special agreements between the requesting organization, the testing organization, and the cell vendors.

NaSTEC
Sodium Sulfur Technology Program
Conclusions

- This program is vital to the transfer of NaS technology to the end user and Phases I and II must be completed by FY97 and Phase III started
- Knowledge gained through this program will be used to design better power systems for satellites
- Each phase can stand alone if necessary, but not recommended
- Total cost of the program is estimated to be $10-$15 million for all phases

In order for NaS technology to become acceptable and available to the end user by FY 97, the goals of the NaSTEC program must be completed. NaSTEC will perform this task by completing Phases I and II, and by starting Phase III. NaS technology is at a point at which tech transfer can be made possible and power systems will be made more efficient through the use of NaS cells.

TRW

IN-STEP Two-Phase Flow (TPF) Thermal Control Experiment

Flight Experiments Technical Interchange Meeting

6 October 1992

Jeff Nienberg, Program Manager

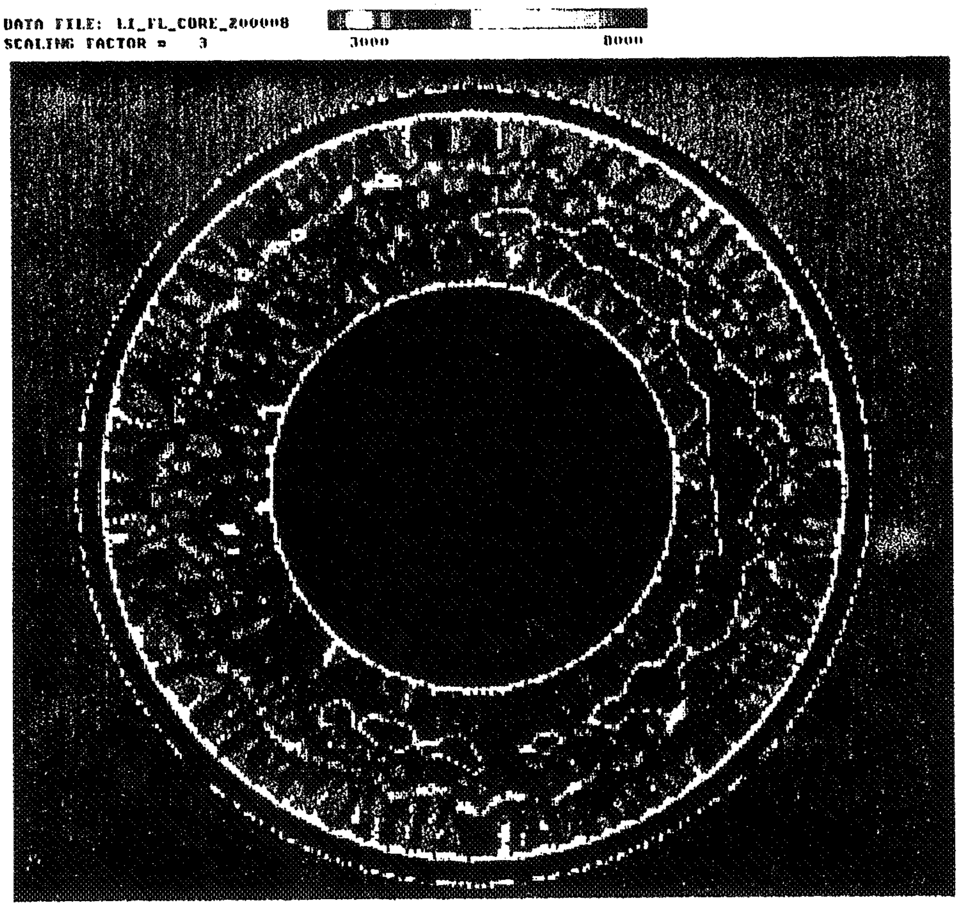
THERMAL ENERGY STORAGE (TES) FLIGHT PROJECT

EXPERIMENTAL CONDITIONS

EXPERIMENT	GEOMETRY	TES	WETTING CONDITIONS
1	ANNULAR	LiF	WETTING
2	ANNULAR	LiF/CaF_2	WETTING
3	WEDGE	LiF	WETTING
4	WEDGE	LiF	NON-WETTING

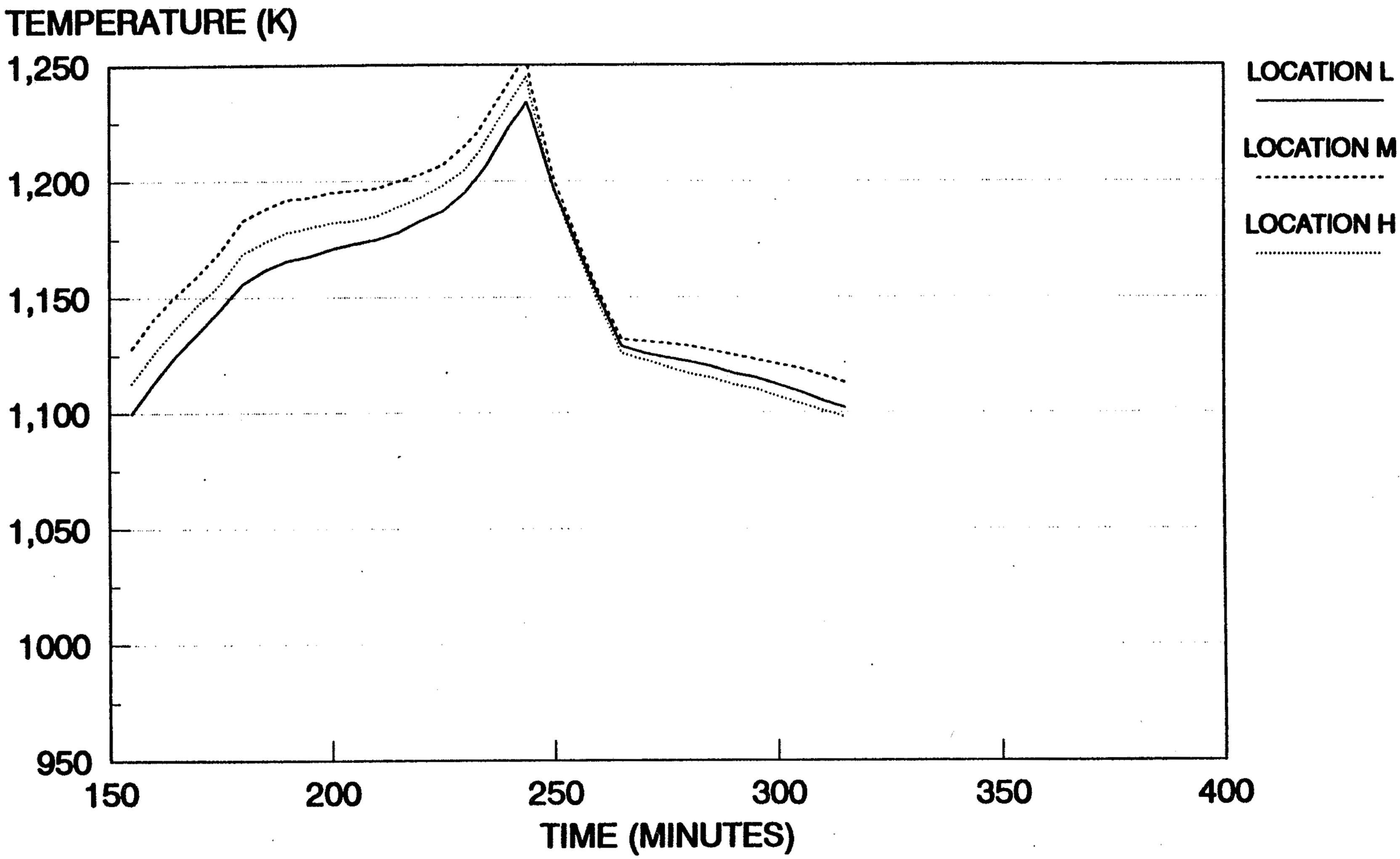
THESE EXPERIMENTS ENCOMPASS THE SIGNIFICANT VARIABLES AFFECTING VOID BEHAVIOR AND LOCATION

DATA FILE: LI_FL_CORE_200008
SCALING FACTOR = 3



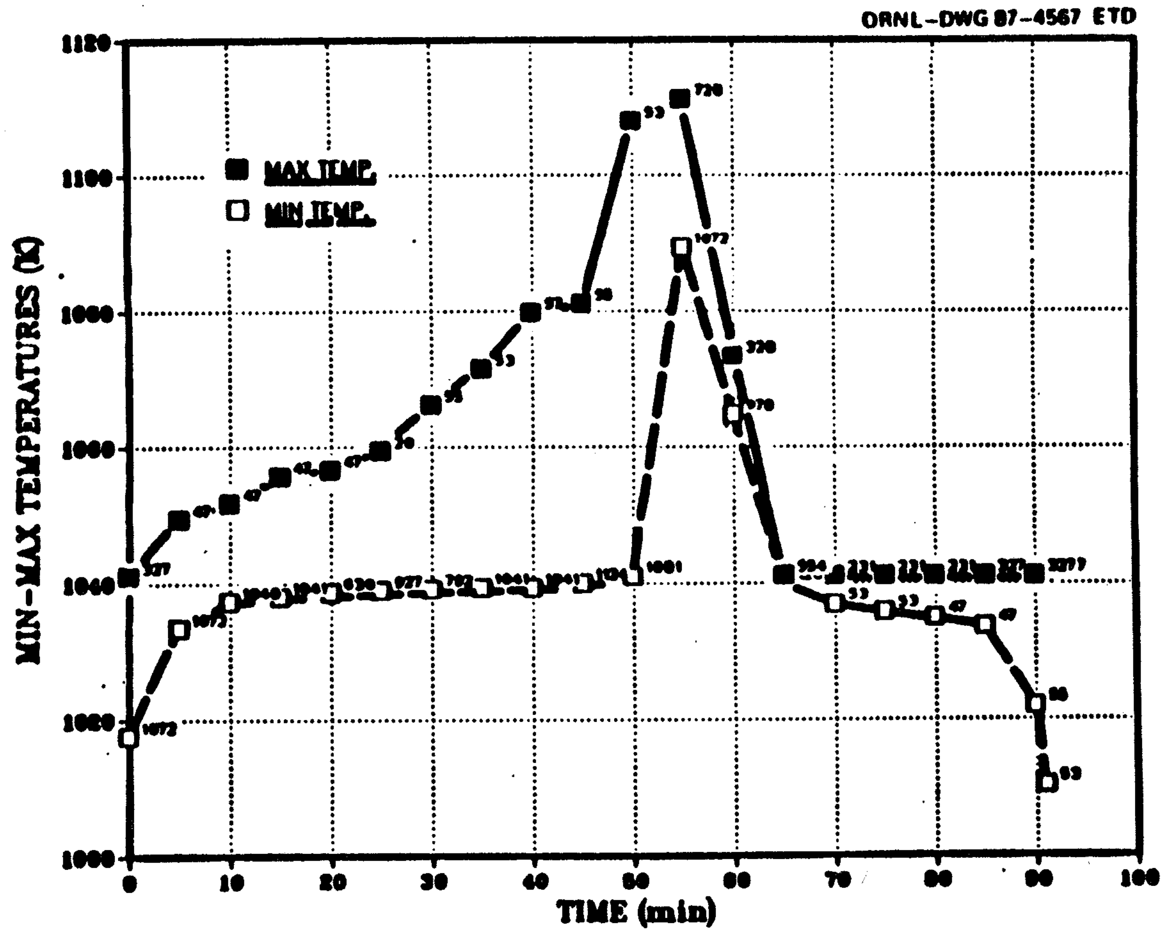
CANISTER TEMPERATURE, 1-g

CANISTER 0-degrees



D. NAMKOONG

TEMPERATURE HISTORY, 1-g



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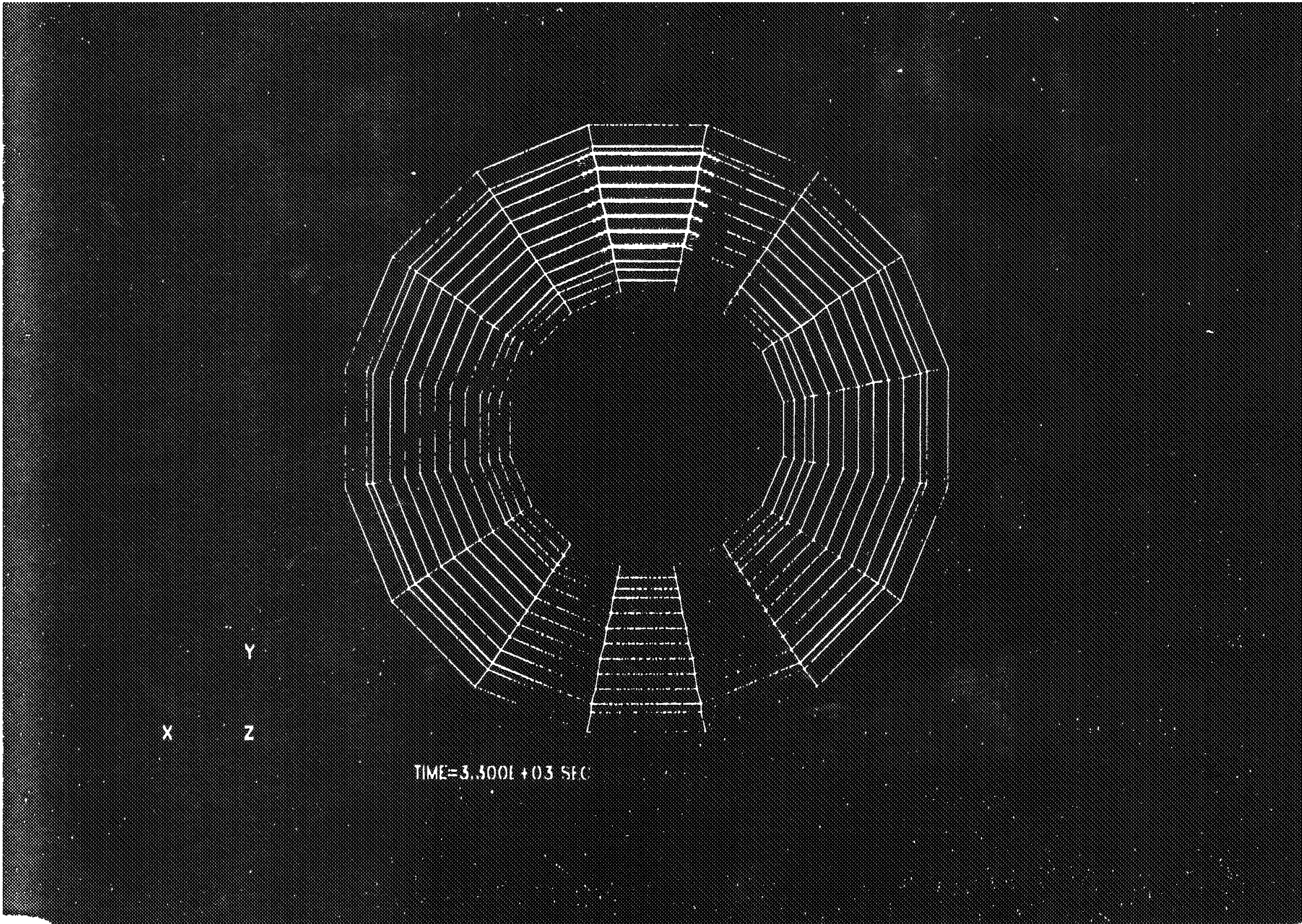
Y
X
Z
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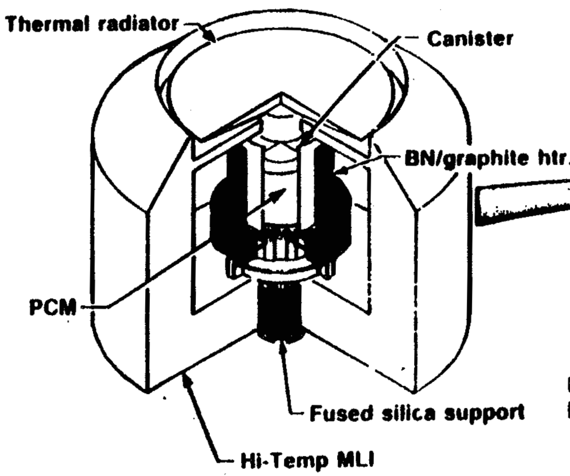
ACCOMPLISHMENTS

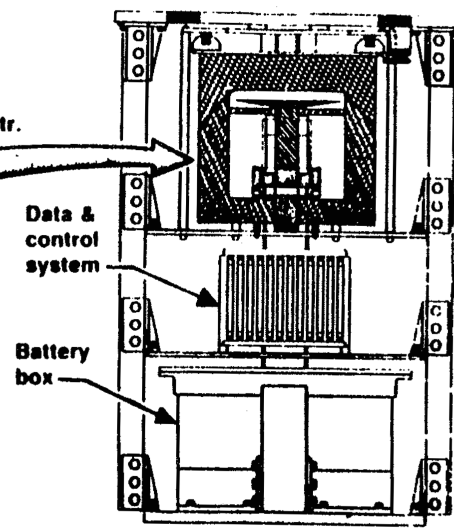
IDENTIFIED AND RESOLVED CRITICAL TECHNOLOGY ISSUES FOR TEST EXPERIMENTS 1 & 2

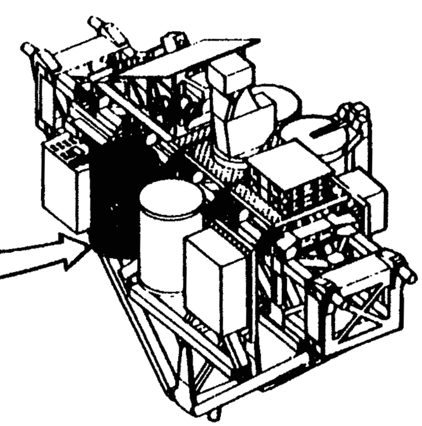
THERMAL ENERGY STORAGE TECHNOLOGY EXPERIMENT

TES-1 experiment section



Complex Autonomous Payload (CAP)



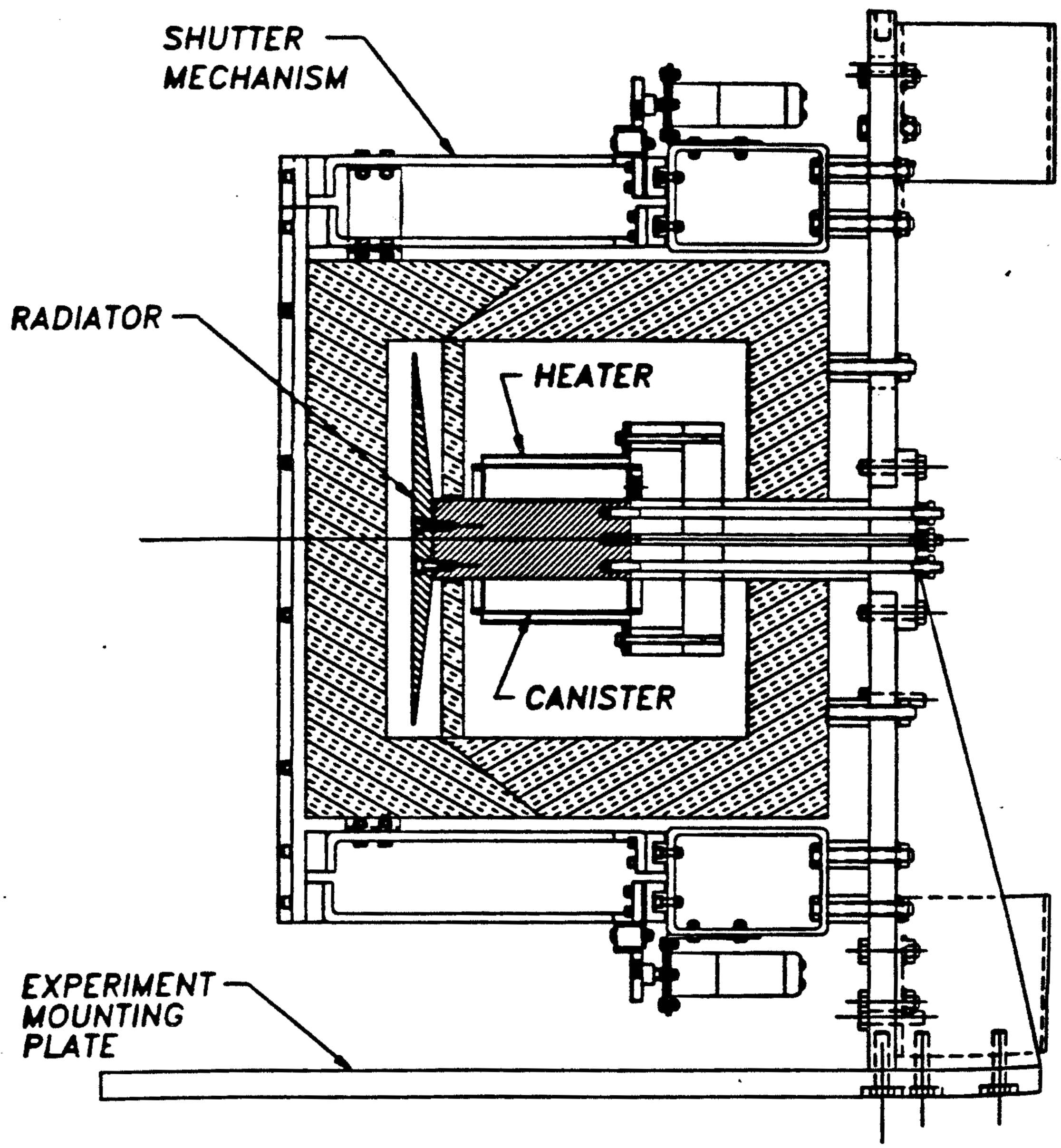


Hitchhiker-M (OAET-1)

DEVELOP IN-SPACE EXPERIMENTS TO CHARACTERIZE VOID SHAPE AND LOCATION IN LiF-BASED PHASE CHANGE MATERIALS IN DIFFERENT ENERGY STORAGE CONFIGURATIONS REPRESENTATIVE OF ADVANCED SOLAR DYNAMICS SYSTEMS

PM/PS: A.J. Szaniszlo

PI: D. Namkoong
SSC: Sverdrup



THERMAL ENERGY STORAGE (TES) EXPERIMENT

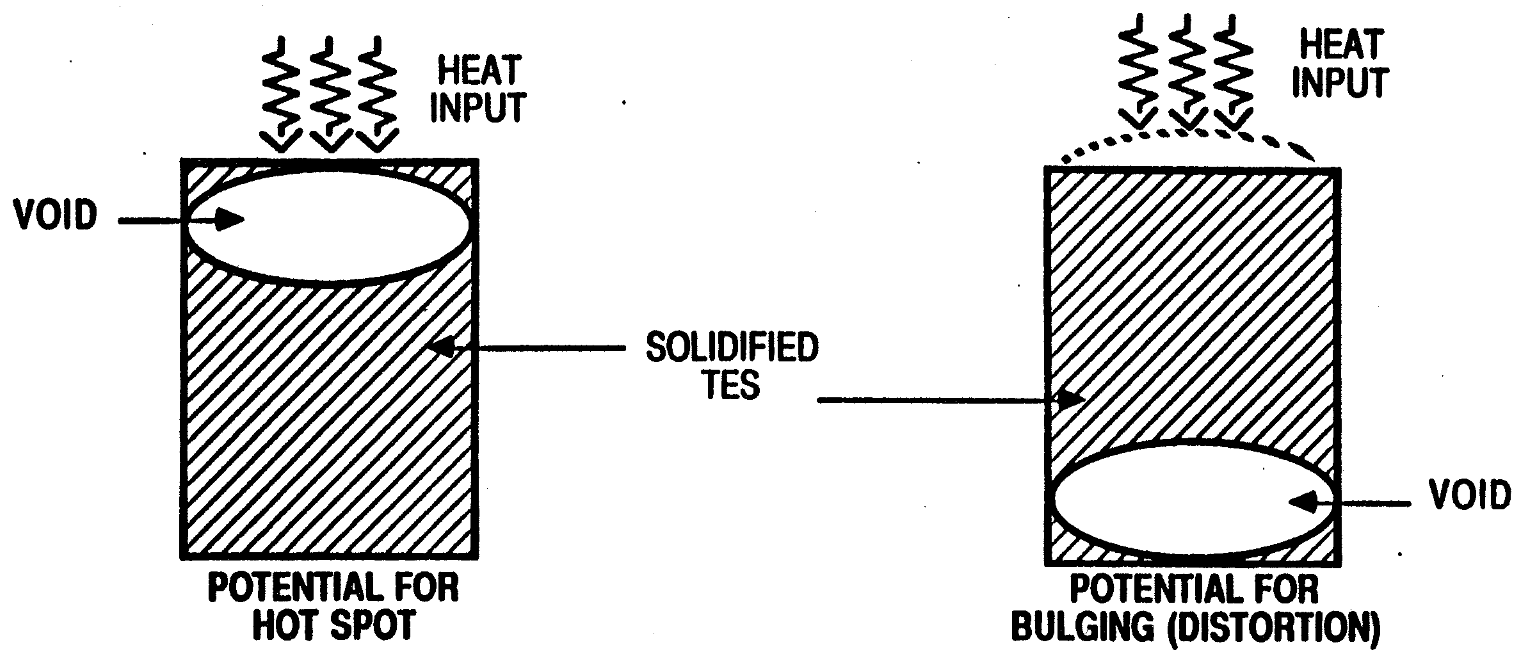
THERMAL ENERGY STORAGE (TES) FLIGHT PROJECT

NORVEX PROGRAM FEATURES

TIME STEPS	IMPLICIT
GEOMETRY	R-Θ-Z
GRAVITY	• 0-G • 1-G, ARBITRARY DIRECTION • CAN PROGRAM VARIABLE G(T)
CANISTER HEAT FLUX	INTEGRATED
VOID LOCATION	MOVING BOUNDARY NAVIER-STOKES
MELT FRONT	"ENTHALPY" METHOD, 3D
RADIANT H.T.	TWO GROUPS • TRANSPARENT • STRONG ABS
VOID H.T.	VAP/COND
STRESS CODE	ADINA
LIFETIME ESTIMATE	ASME CODE

THERMAL ENERGY STORAGE (TES)

VOLUME CHANGE WITH PHASE CHANGE -
IMPACT ON VOID BEHAVIOR IN MICROGRAVITY



THERMAL ENERGY STORAGE (TES) FLIGHT PROJECT

WHY THE PROJECT?

ADVANCED SOLAR DYNAMIC SYSTEMS REQUIRE BETTER TECHNICAL ASSESSMENT OF THERMAL ENERGY STORAGE (TES) SALTS UNDERGOING FREEZING AND THAWING

- SOLAR DYNAMIC SYSTEMS DESIGNED FOR SUN-SHADE ORBITAL MISSIONS INCLUDE TWO PHASE STORAGE OF ENERGY AS INTEGRAL PART OF RECEIVER
- LACKING DATA OF TES SALTS UNDERGOING FREEZE/THAW IN MICROGRAVITY, SOLAR DYNAMIC RECEIVERS HAVE BEEN DESIGNED CONSERVATIVELY -- HEAVIEST COMPONENT OF SYSTEM
- ADVANCED SOLAR DYNAMIC SYSTEMS ARE BASED ON LIGHTER WEIGHT, BETTER PERFORMANCE COMPONENTS

CONCLUSION:	NEED FOR ANALYTIC - EXPERIMENTAL BASIS TO DEVELOP CAPABILITY FOR ADVANCED SOLAR RECEIVER/TES DESIGNS

Two-Phase Flow Experiment

TRW

Background

Program sponsored by NASA GSFC, Thermal Engineering Branch

Part of the NASA/OAST In-Space Technology Experiments (IN-STEP) Program

TRW completed experiment definition (phase A) in August 1989

Engineering development phase (phase B) initiated July 1992

- Preliminary design of flight experiment
- Breadboard test and characterization of thermal control system (TCS)
- Non-Advocate Review

Flight development phase (Phase C/D)

- Final Design
- Experiment Fabrication and Assembly
- Environmental Testing
- Flight Operations and Post-Flight Analysis

Two-Phase Flow Experiment

TRW

Background (cont.)

Experiment configured for NASA Hitchhiker Shuttle Payload System

Two-phase thermal control system consists of

- Capillary pumped loop (CPL)
- Heatpipe radiator
- Two-phase flow heat exchanger (TPFHX)

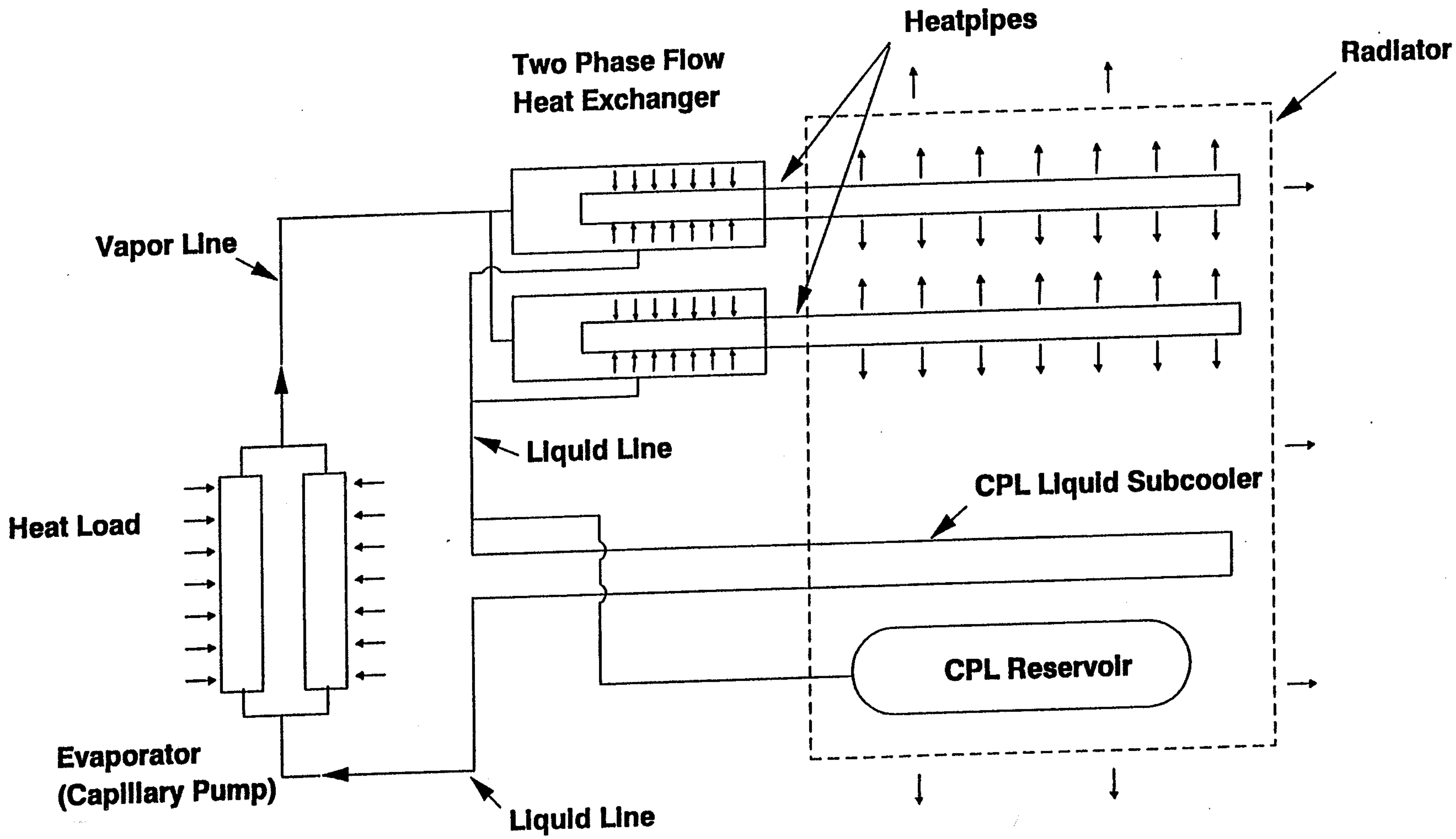
TRW

Schedule

Phase B	Start	July	1992	
	PDR	April	1993	
	NAR	June	1993	
Phase C/D	Start	Sept	1993	
	CDR	Jan	1994	
	Delivery	April	1995	
	Flight	July	1995	(OAST-3)
	Post-Flight	Oct	1995	

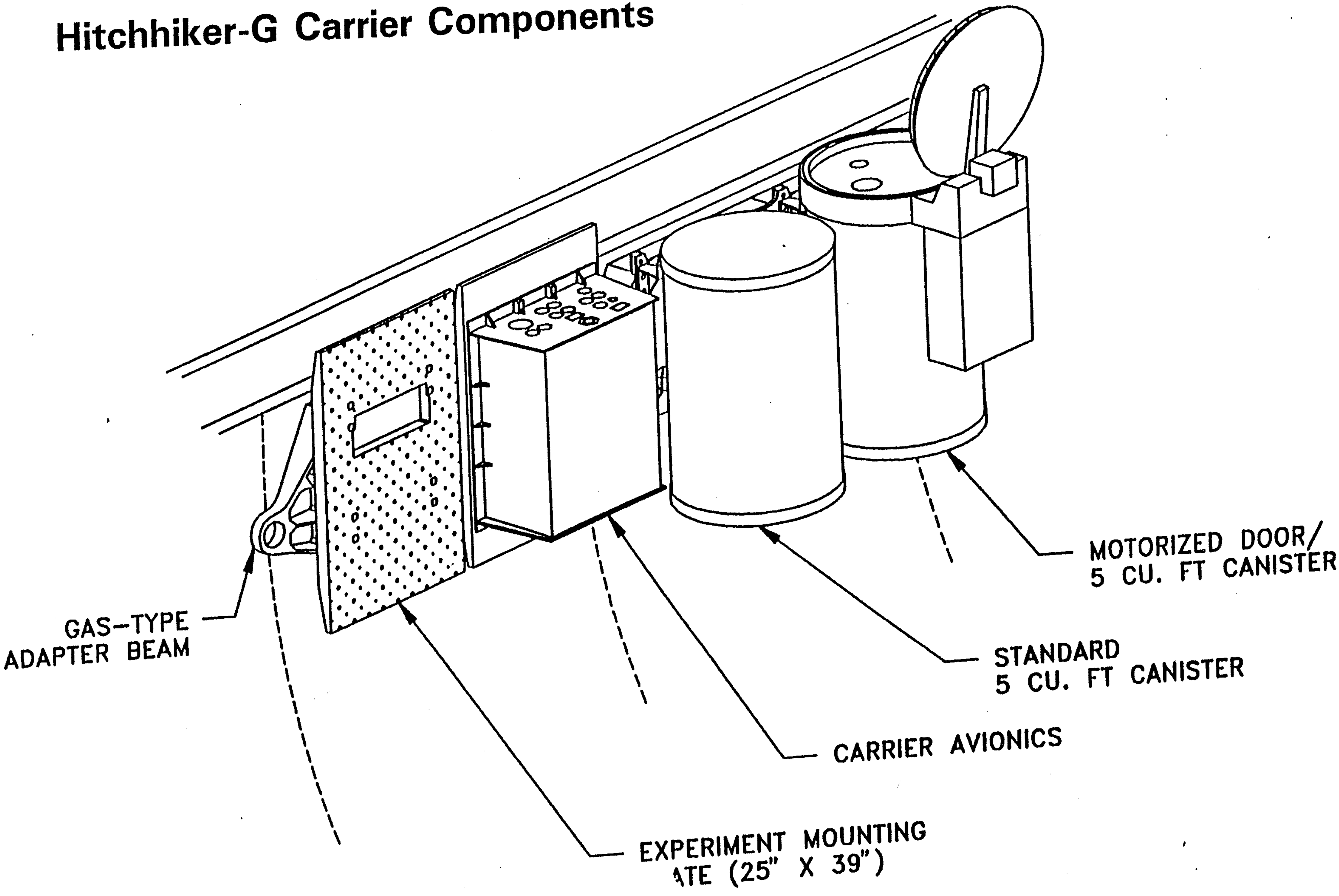
Two-Phase Flow Experiment

Thermal Control System Schematic



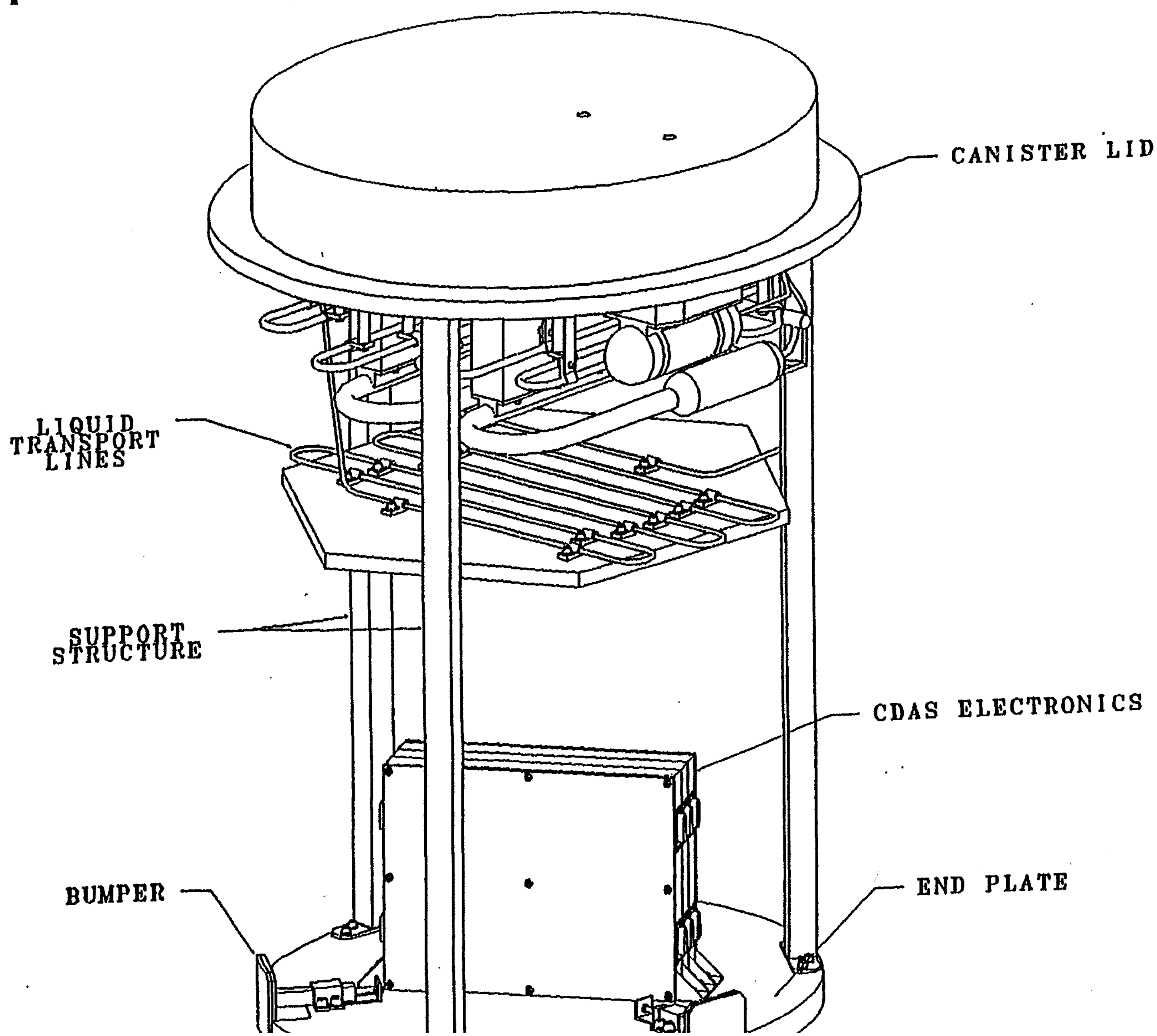
Two-Phase Flow Experiment

Hitchhiker-G Carrier Components



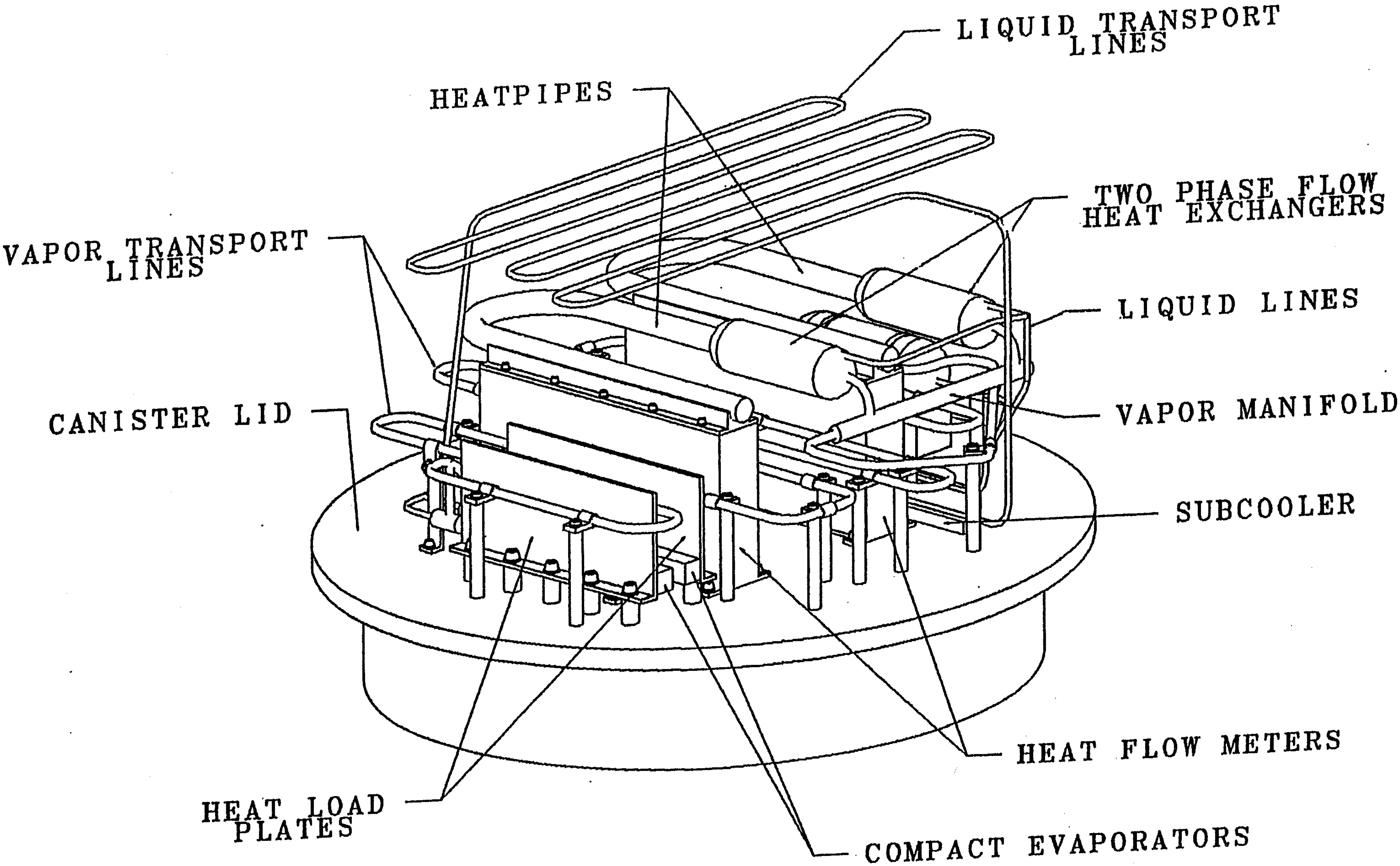
Two-Phase Flow Experiment

Flight Experiment Payload



Two-Phase Flow Experiment

Thermal Control System Mechanical Design



Flight Experiment Thermal Design Approach and Test Plan

Canister lid serves as the thermal control system heatsink and radiator with added weight to provide a high thermal capacity

Experiment instrumented with thermistors to measure temperatures required for determining the heat transfer coefficients in all components

Heat flux meters integral with the heatpipe condenser saddles will measure the heat load through the individual heatpipes

Reservoir controlled at constant temperature

Test plan includes power cycling at various levels, heat sharing, and induced deprime

Command and Data Acquisition (CDAS) system will provide

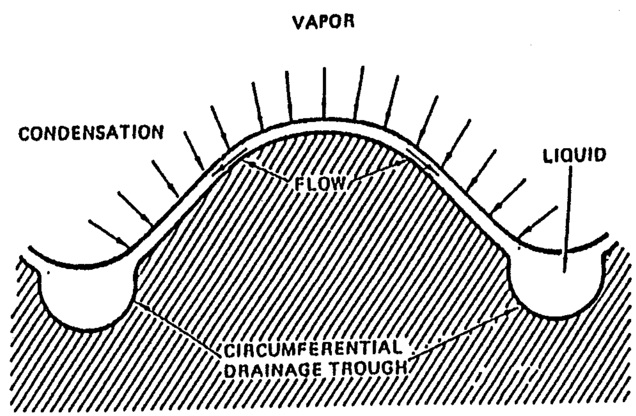
- Real time data and command capability
- Temperature measurement accuracy of $\pm 0.1\,^{\circ}C$
- Bus voltage measurement

TRW

Two-Phase Flow Heat Exchanger (TPFHX)

A small temperature drop in the heat exchanger between the capillary pumped loop and the heatpipe radiator is critical for an efficient thermal control system

Gregorig condensation grooves balance capillary and viscous forces producing a thin constant film thickness for a high condensation heat transfer coefficient

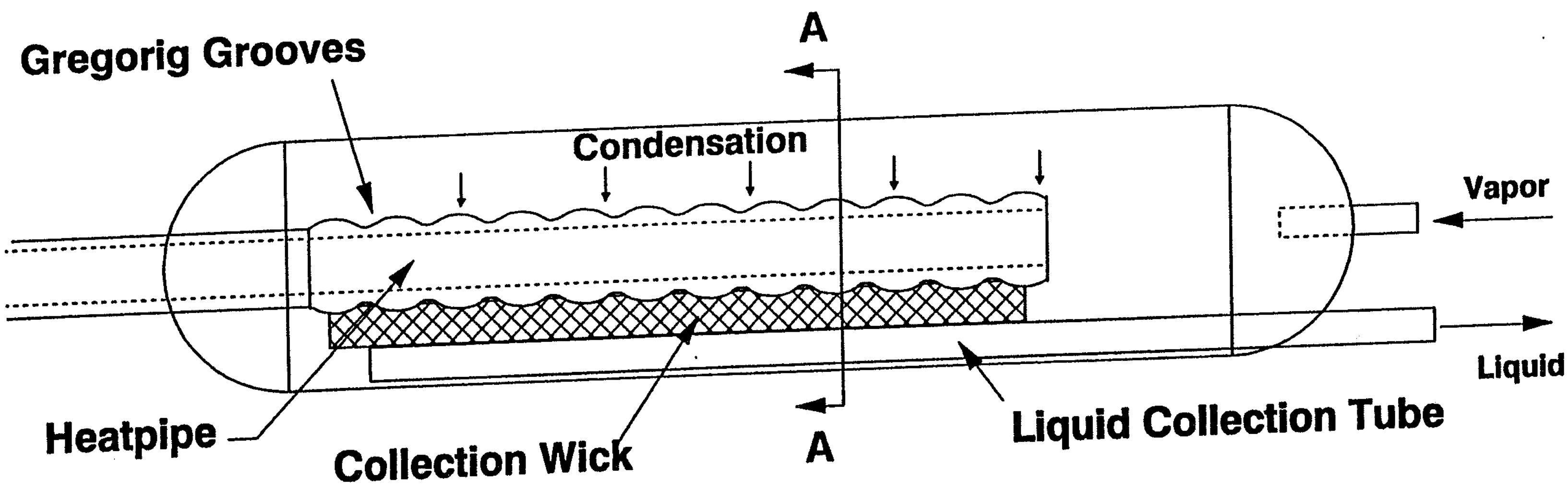


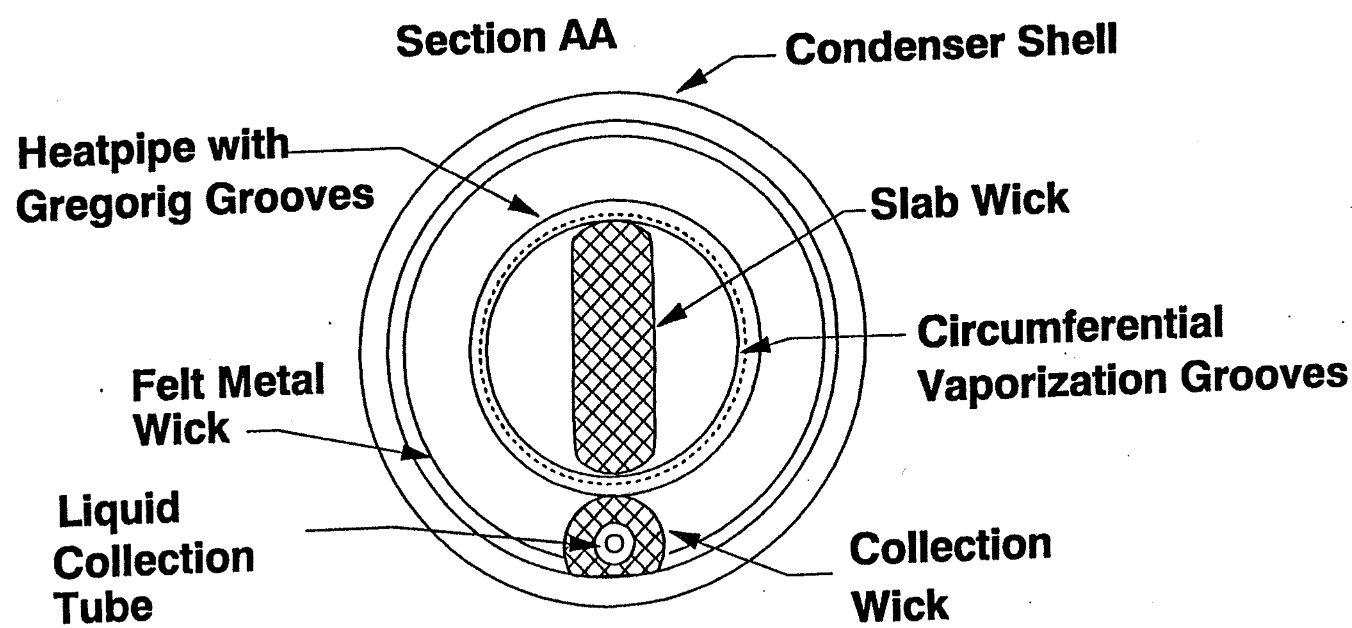
Theory predicts condensation "h" values that are an order of magnitude greater than for typical condenser sections in heatpipes

Verification requires microgravity environment of space since capillary forces which determine liquid flow patterns are dominated by gravity during ground testing

TRW

Two-Phase Flow Heat Exchanger (TPFHX)





Two-Phase Flow Experiment

TRW

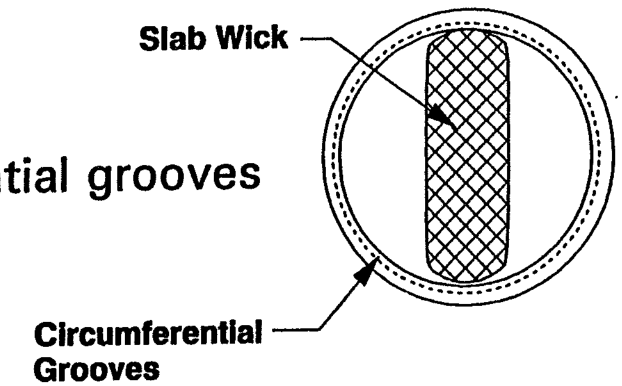
Radiator Heatpipes

Incorporating two heatpipe designs will maximize the performance data obtained from the flight experiment

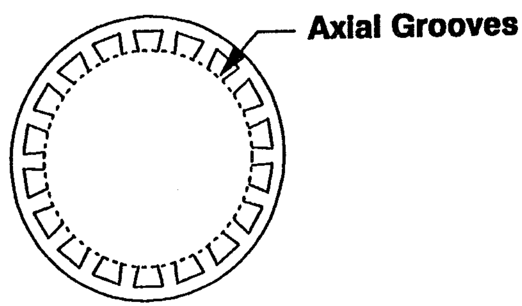
Design emphasis will be placed on achieving improved evaporation heat transfer coefficients, "h"

Candidate designs include

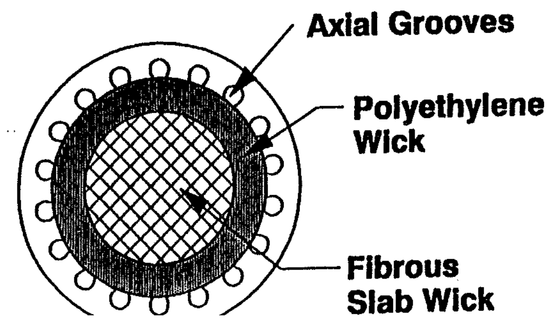
Slab wick heatpipe with high density circumferential grooves (200 grooves/inch)



Axial groove heatpipe



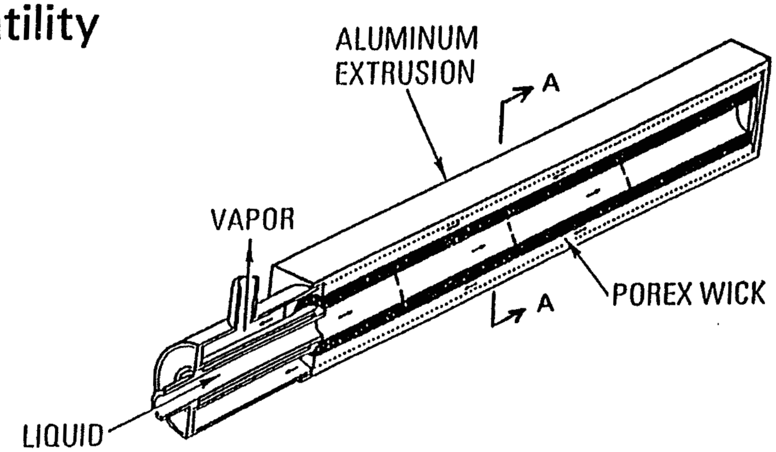
Inverted meniscus heatpipe under development at TRW



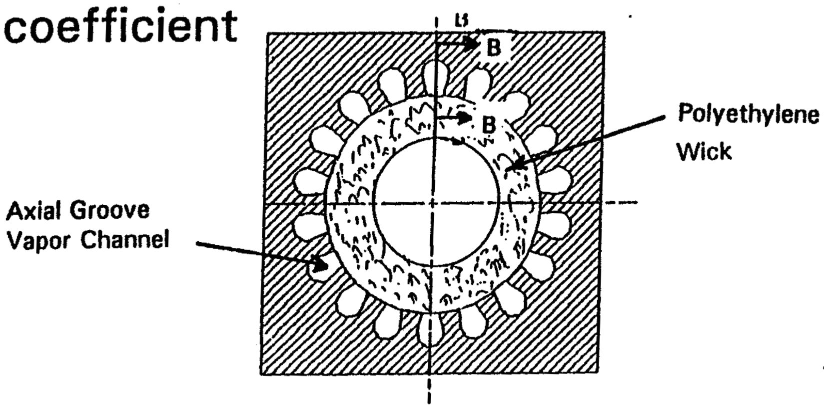
Two-Phase Flow Experiment

TRW Compact Evaporator Pump

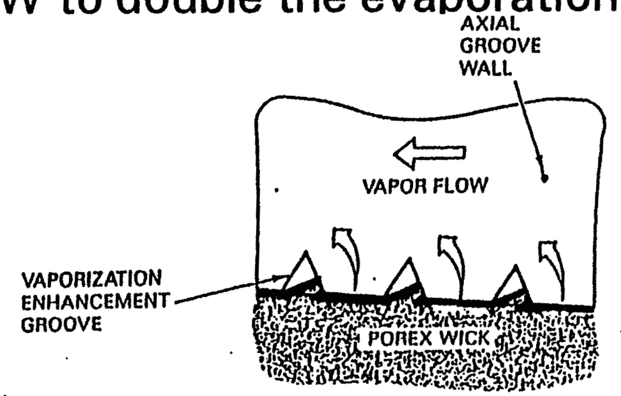
Compact mechanical design with liquid and vapor outlet at same end enhances integration versatility



Vaporization Enhancement Grooves (VEGs) machined onto the lands (flats) of evaporator extrusion have been shown by TRW to double the evaporation heat transfer coefficient



Section AA



SECTION BB

Can be verified in flight experiment by incorporating VEGs in one of the two evaporator pumps and comparing performance

Thermal Math Model of Thermal Control System

Uses SINDA Thermal Analyzer Program

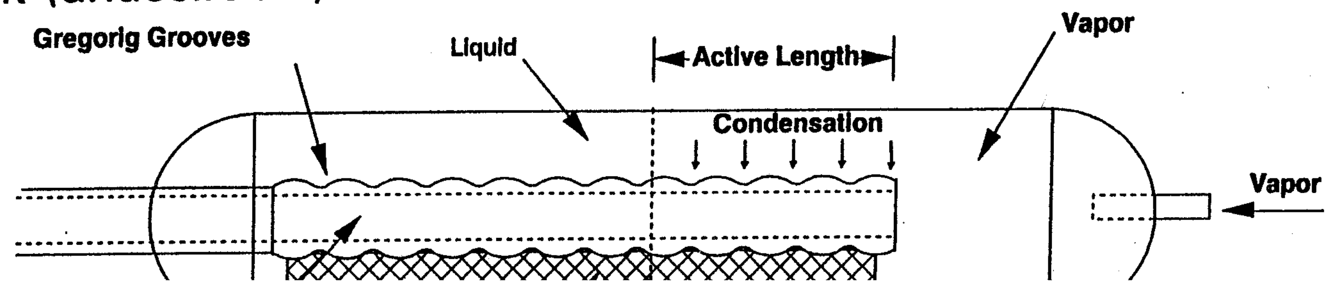
Developed to assist in component sizing and to simulate experiment operational conditions

Includes: two CPL evaporator pumps, TPFHX, subcooler, CPL reservoir, heat flow meters, canister lid

Predicts transient temperature response of all components and the active length of the heat exchanger condenser

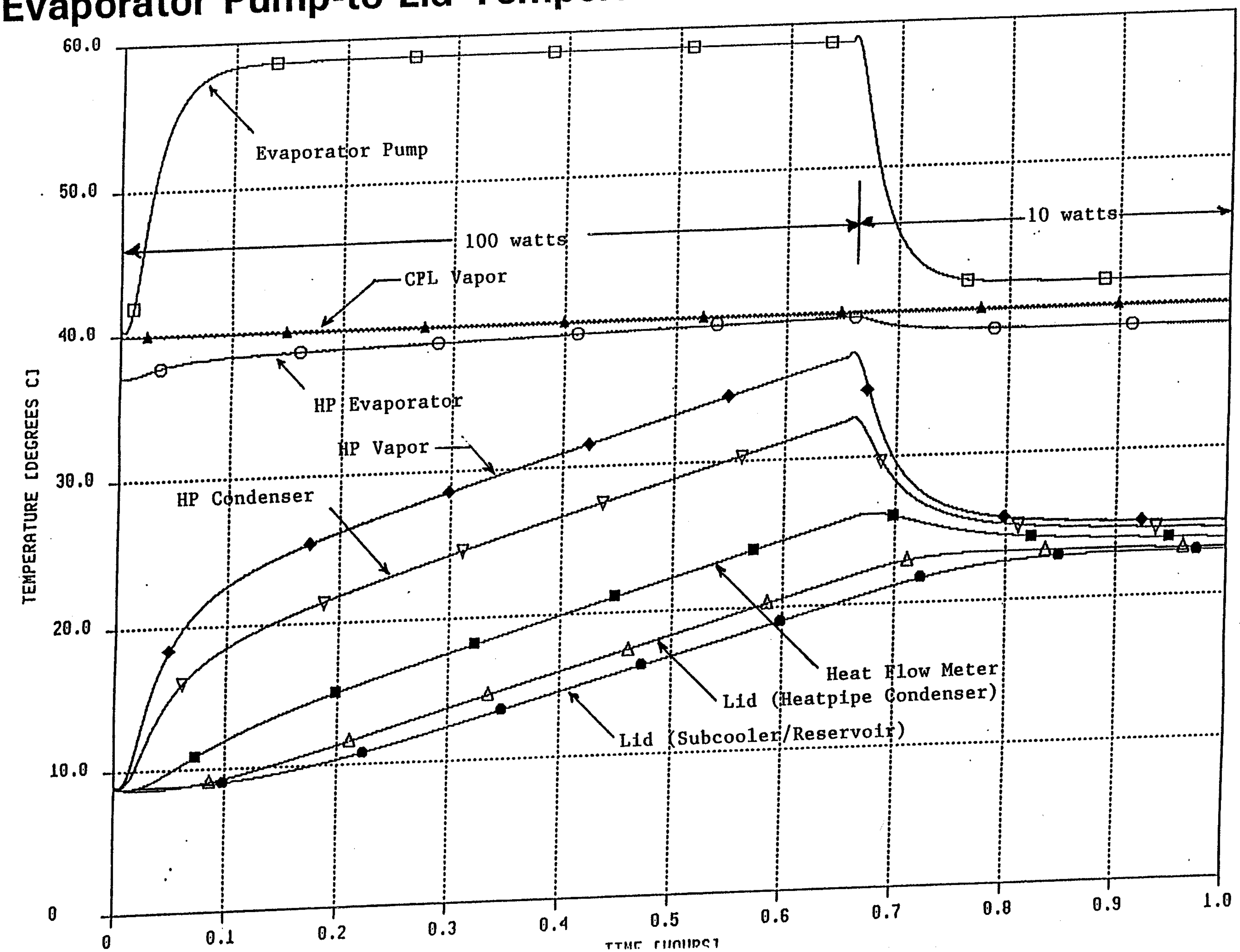
- Active length is a function of the CPL heat load and the sink temperature

- When the active length reaches the total condenser length the TPFHX becomes overdriven, i.e., CPL vapor blows by the TPFHX liquid collection wick (undesirable)



Two-Phase Flow Experiment

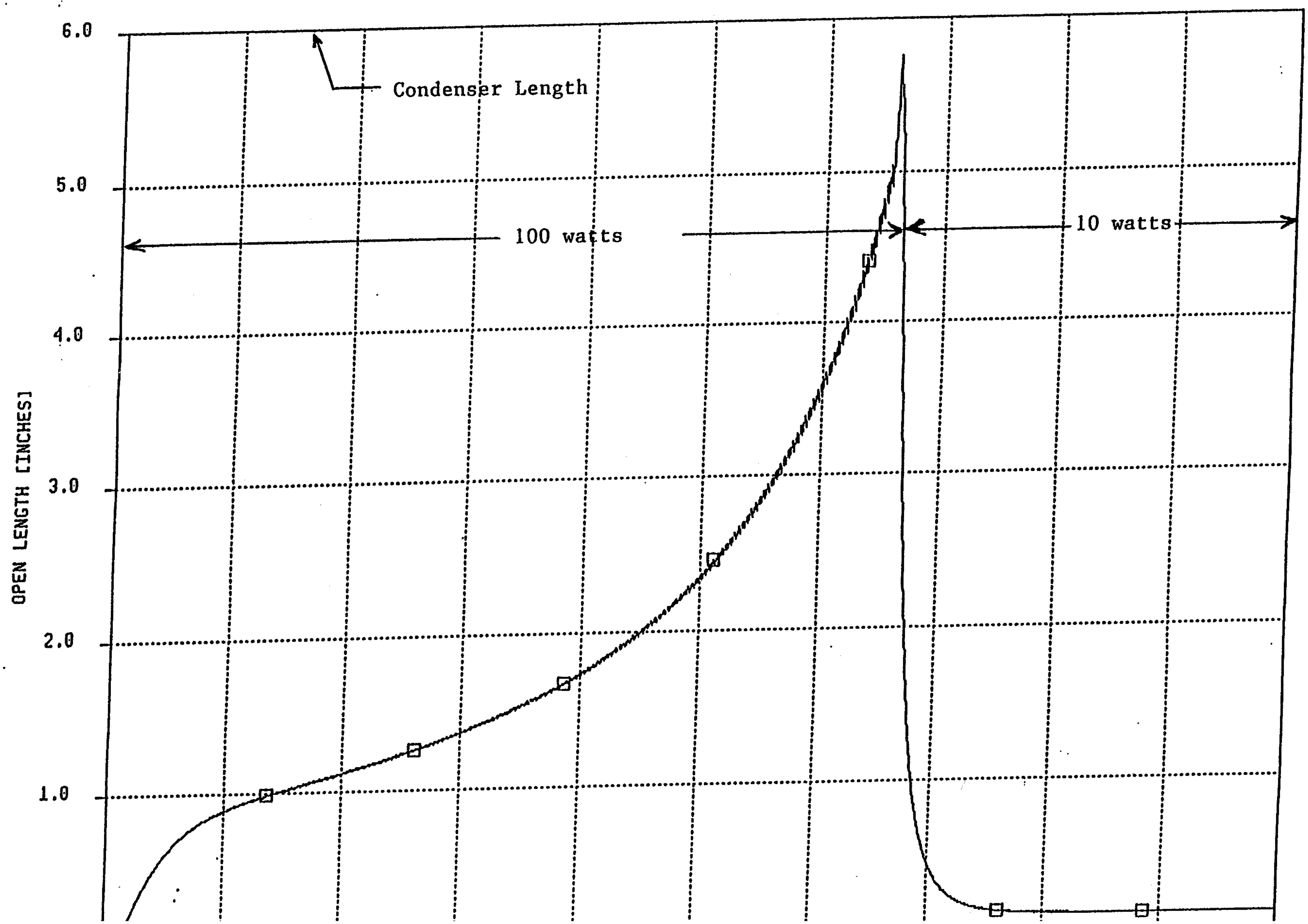
Evaporator Pump-to-Lid Temperatures



Two-Phase Flow Experiment

TRW

Heat Exchanger Condenser Open Length



One-G Measurement of Condensation "h" on Gregorig Grooves

Purpose is to bound the expected zero-g values for condensation heat transfer coefficient on gregorig-grooved surfaces

Existing hardware has been modified and incorporated into the test setup

Measurements will be made at several power levels with the grooves facing up and with the grooves facing down

- Gravity will enhance the heat transfer in "up" orientation and retard heat transfer in the "down" orientation

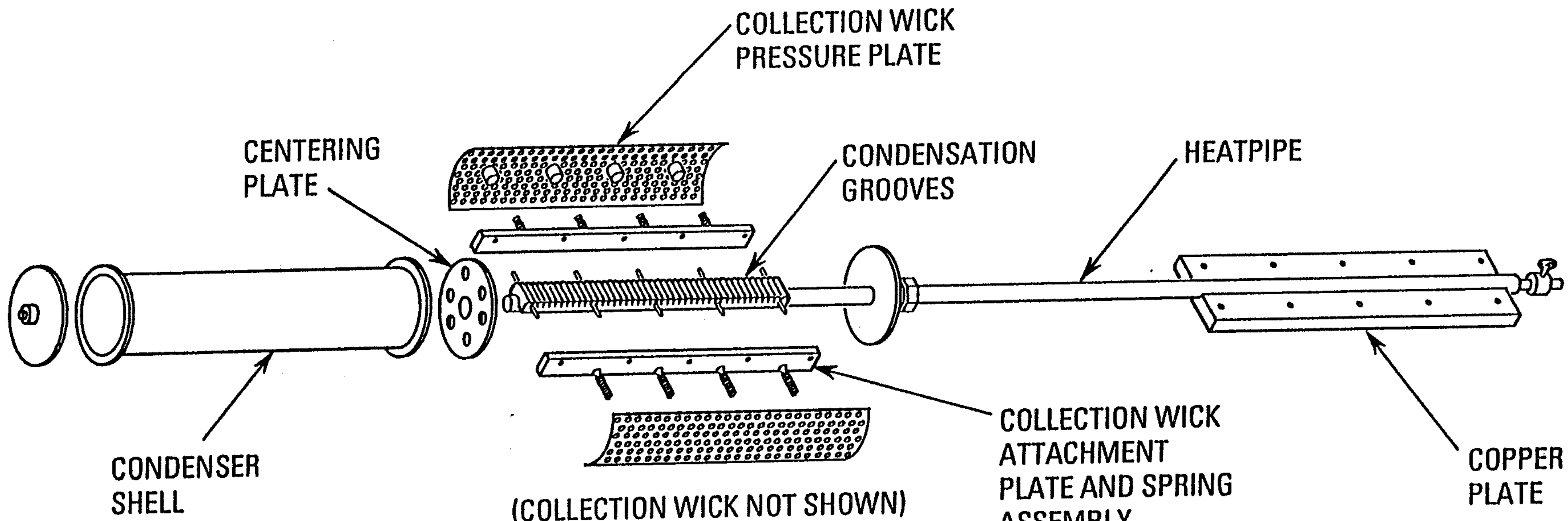
Liquid inventory in the condensation chamber will also be varied

Preliminary data has been taken and is currently under analysis

Two-Phase Flow Experiment

TRW

Exploded View of One-G Test Apparatus



CRYOGENIC HEAT PIPE EXPERIMENT BACKGROUND

- NO MICRO-GRAVITY DATA AVAILABLE FOR OXYGEN OR NITROGEN HEAT PIPES
- POOR WICKING AND LOW TRANSPORT MAKE 0-G EXTRAPOLATION DIFFICULT
- RELIABLE START UP FROM SUPER CRITICAL TEMPERATURE NEEDS TO BE DEMONSTRATED
- MICRO-GRAVITY INFORMATION ON CRYO (<100 K) HEAT PIPES IDENTIFIED AS CRITICAL TECHNOLOGY NEED BY NASA AND THE AIR FORCE - 1988 THERMAL FLUIDS IN SPACE WORKSHOP AND IN STEP 88 WORKSHOP
- OXYGEN AND NITROGEN PIPES BUILT AND EVALUATED

CRYOGENIC HEAT PIPE EXPERIMENT

OBJECTIVE

CONDUCT A SHUTTLE EXPERIMENT TO DEMONSTRATE THE RELIABLE OPERATION OF TWO OXYGEN HEAT PIPES IN MICROGRAVITY.

1. DEMONSTRATE STARTUP OF THE PIPES FROM THE SUPER-CRITICAL STATE.
2. MEASURE THE HEAT TRANSPORT CAPACITY OF THE PIPES
3. MEASURE EVAPORATOR AND CONDENSER FILM COEFFICIENTS
4. WORK SHUTTLE SAFETY ISSUES

APPROACH

- ✓ FLY TWO AXIALLY GROOVED OXYGEN HEAT PIPES ATTACHED TO MECHANICAL STIRLING CYCLE TACTICAL COOLERS
- ✓ INTEGRATE EXPERIMENT IN HITCHHIKER CANISTER
- ✓ FLY ON SHUTTLE AND CONTROL FROM GROUND

CRYOHP DESCRIPTION

- o Shuttle/HH Carrier Flight Experiment (Minus Avionics) Less Than 345 lbs

- o HH Canister
  - Modified Upper End Plate (UEP)
    - o Thermal Mass
    - o Radiator
    - o Flown on CPL/GAS and CPL/HH-1

- o Uninsulated Top Plus Sides

- o Vented Can (Valves in Lower End Plate (LEP))
  - 16 Psia Prior to Launch
  - 2 Psia Differential Pressure Relief Valves on Ascent
  - Solenoid and Butterfly Valves Provide Flight Vacuum

- o HH Avionics

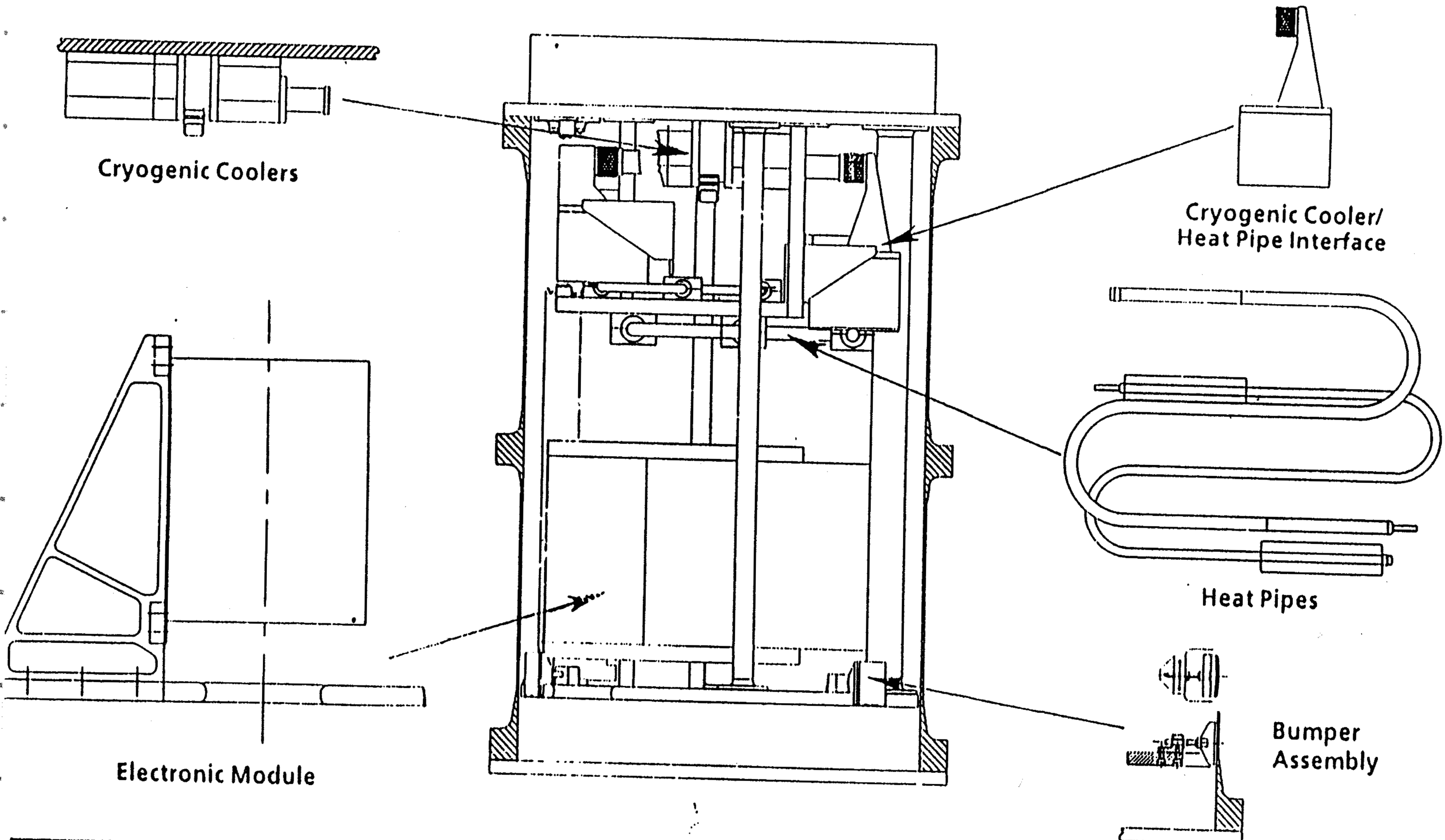
  - Provides Power, Signal, Command, and Data
  - 3 HH Ports Required

CRYOHP DESCRIPTION (cont.)

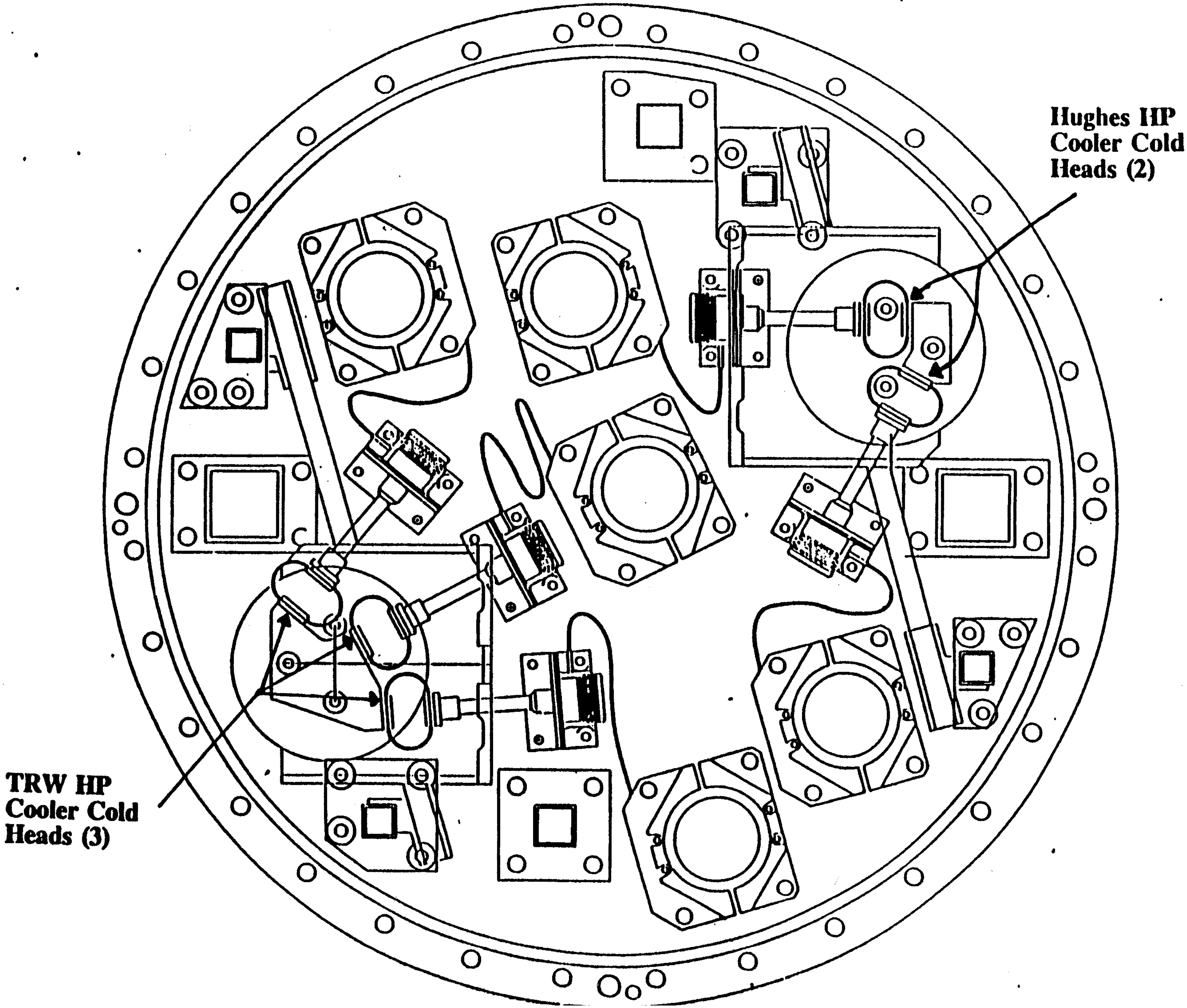
- **Heat Pipes**
  - Two Independent Designs
  - Axially Grooved Aluminum Extrusion
    - TRW
    - Hughes

- **Cryo-Coolers**
  - Five Split Stirling Cycle Coolers
    - Hughes Model No. 7044H
      - 3.5 Watts Each @ 80K
      - Mounted to HH Canister UEP
      - Helium at 450 Psia Maximum
      - 95 W Power, 7.5 Amp Startup for 100 Millisecond Max.

CRYOHP SUBSYSTEM IMPLEMENTATION



CRYOHP – FIVE COOLER ARRANGEMENT



CRYOHP OPERATIONS SCENARIO

ASCENT - Vent to 2 PSIA

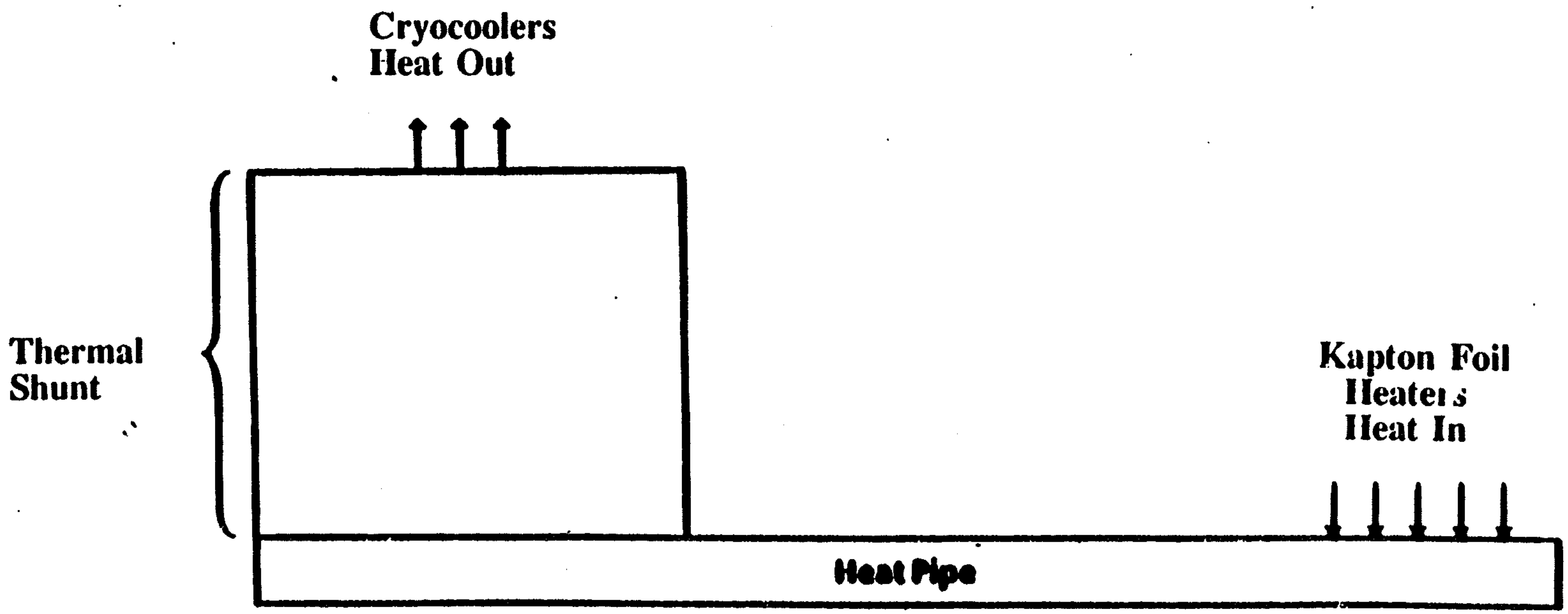
ORBIT

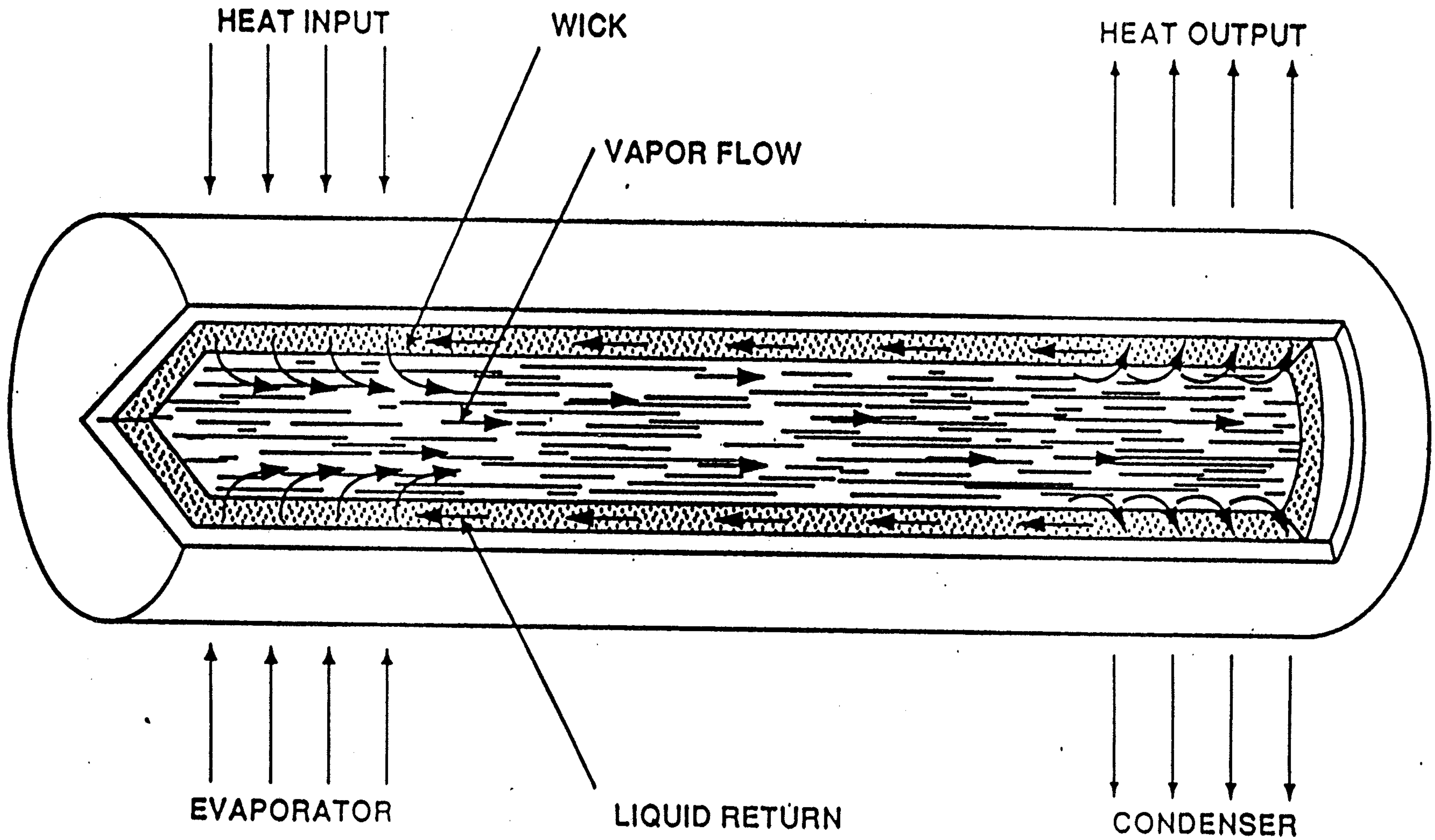
- Hitchhiker Avionics On
- Survival Heaters On
- Vent to 10^{-4} Torr or Less
- CRYOHP On
- Cooldown TRW Heat Pipe
  - o Start Up
- Cooldown
  - o Transport/Recovery
- Cooldown
  - o Transport/Recovery/Minimum Temperature
- Cooldown Hughes Heat Pipe

  Repeat
- Cooldown TRW Heat Pipe

  Repeat - Total Five Cycles Each Pipe
- CRYOHP Off
- Descent

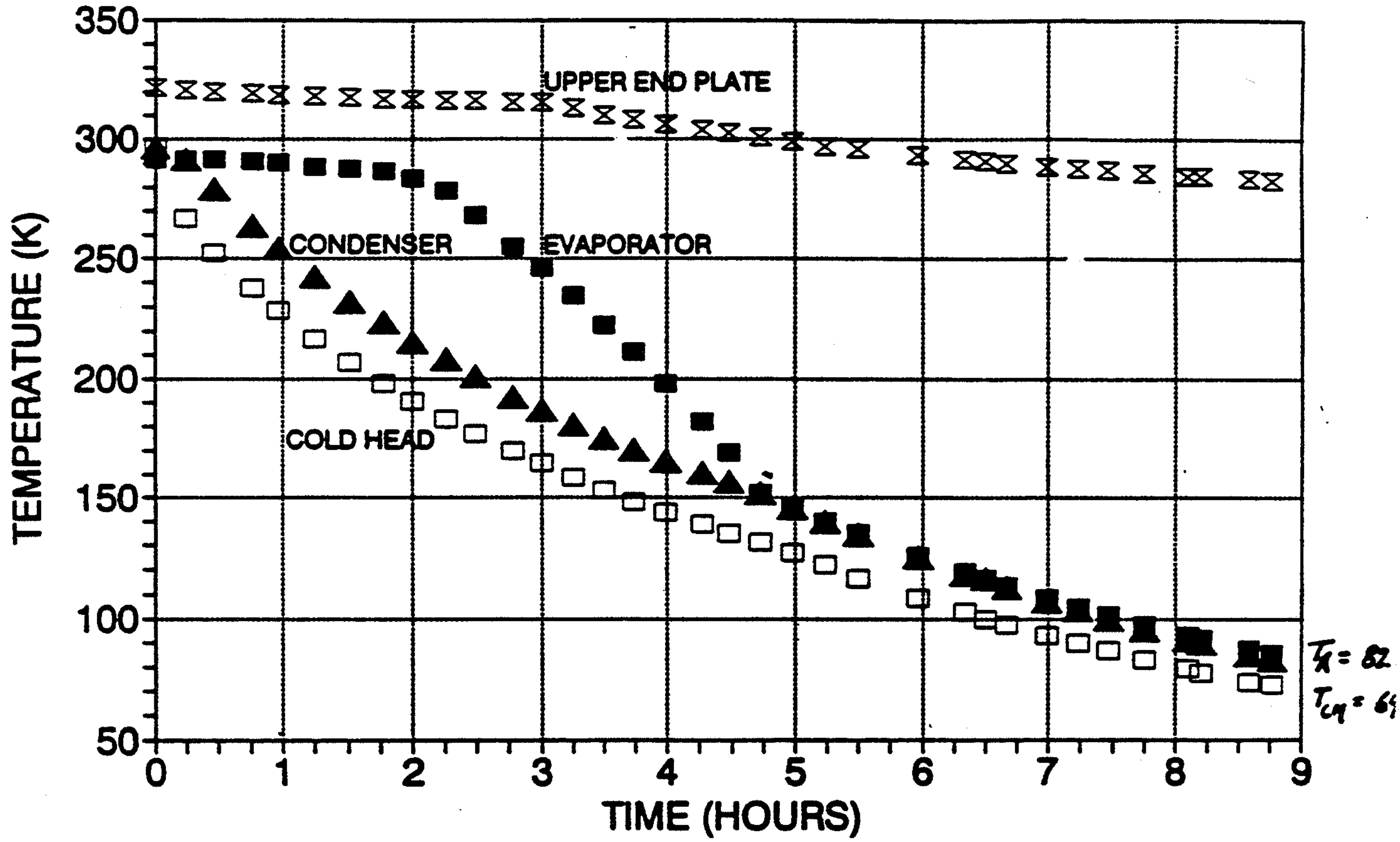
SCHEMATIC -- CRYOHP OPERATION





HEAT PIPE - CONCEPT

HAC HEAT PIPE TRANSIENT COOLDOWN
TEST DATE : 04-09-92



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CRYOHP INSTRUMENTATION

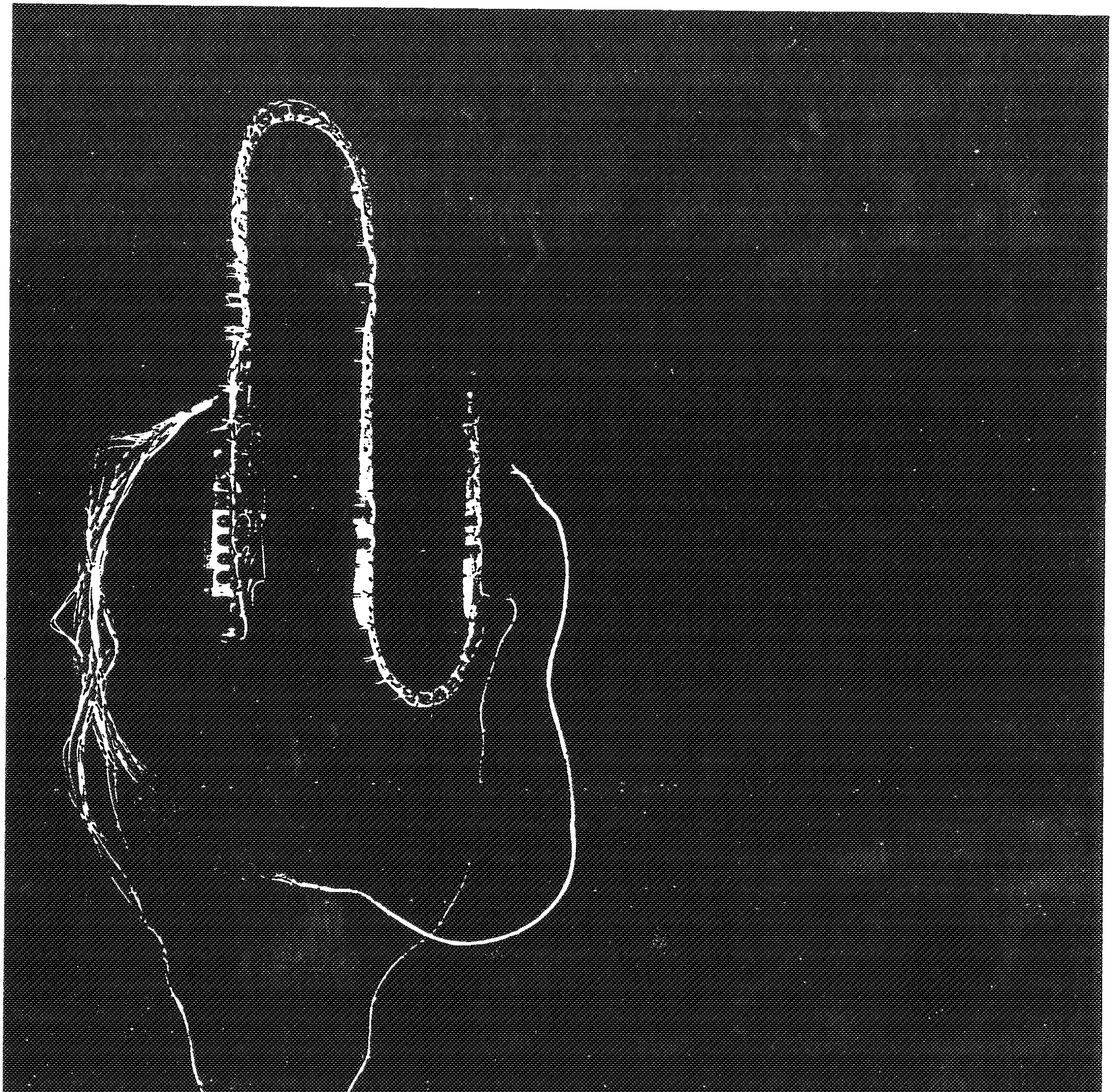
TYPE	QUANTITY	LOCATION
Platinum Resistance Thermometers (PRTs)	26	13 each heat pipe system
Thermistors	24	UEP, EBP, pillars, heat pipe structure, cryo-coolers, electronics
Thermistors	9 (HH)	EBP, Canister, & CECM Mounting Brackets
Pressure Transducers	1	Canister Internal Pressure
Current Monitors	13	CECM
Voltage Monitors	18	1 for bus voltage, 17 for temperature calibration
Heaters (Kapton foil)	11	4 per heat pipe, 3 survival
Thermostats	33	Tri-series circuit for each heater

CRYOGENIC HEAT PIPE EXPERIMENT
CURRENT STATUS

- DELIVERED TO KSC AND INSTALLED ON SHUTTLE
- FINAL INTERFACE VERIFICATION TEST COMPLETED
- ALL DOCUMENTATION COMPLETE
- LAUNCH DUE ON NOVEMBER 16, 1992

CRYOGENIC HEAT PIPE EXPERIMENT
FY 93 PLANS

- SUPPORT FLIGHT OPERATIONS
- REDUCE FLIGHT DATA AND RESOLVE ANY ANOMALIES
- PERFORM POST FLIGHT TESTS ON EXPERIMENT AND HEAT PIPES
- INCORPORATE RESULTS INTO GROOVE ANALYSIS PROGRAM AND SUBMIT TO COSMIC
- COMPLETE FINAL REPORT



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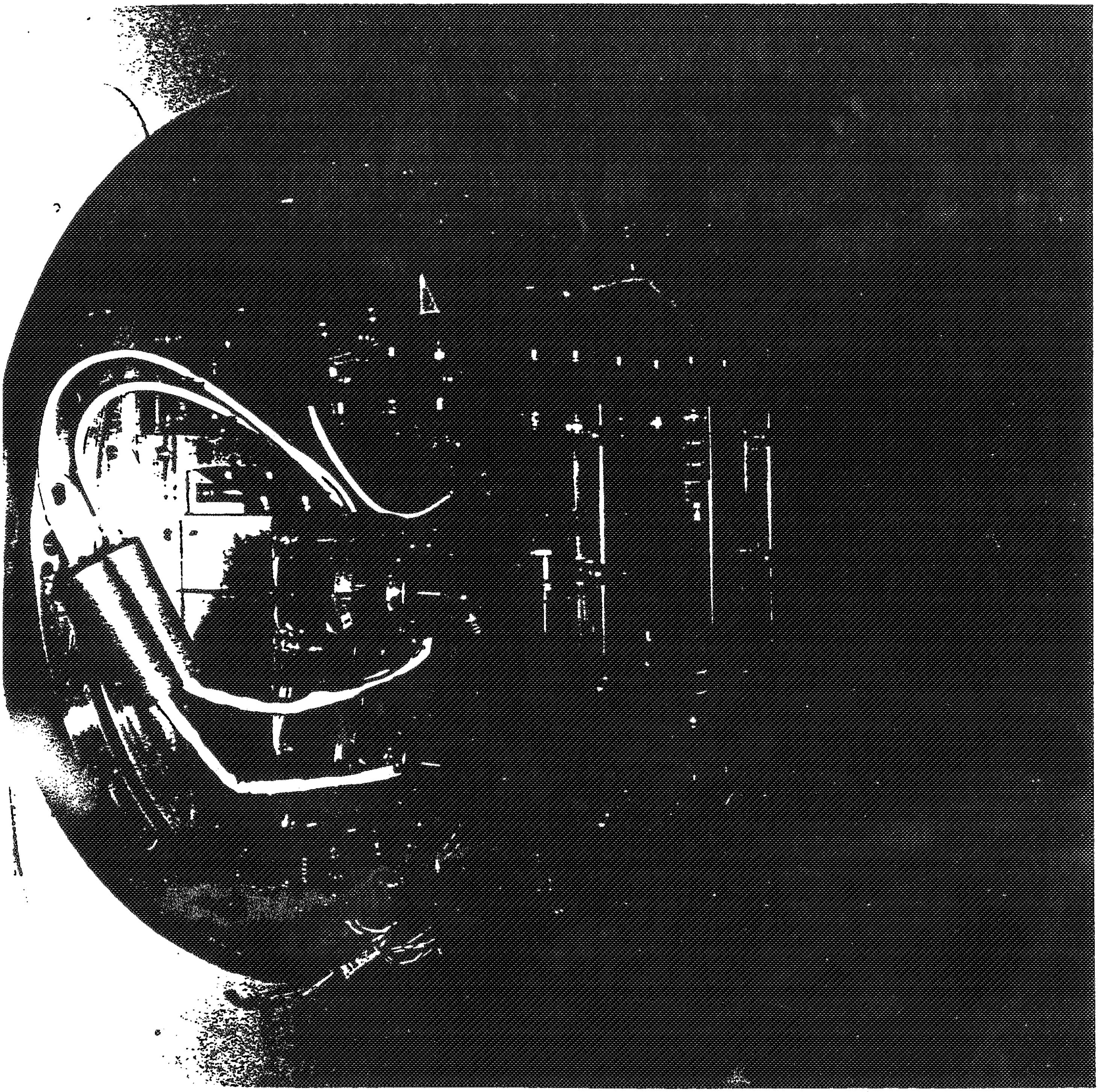

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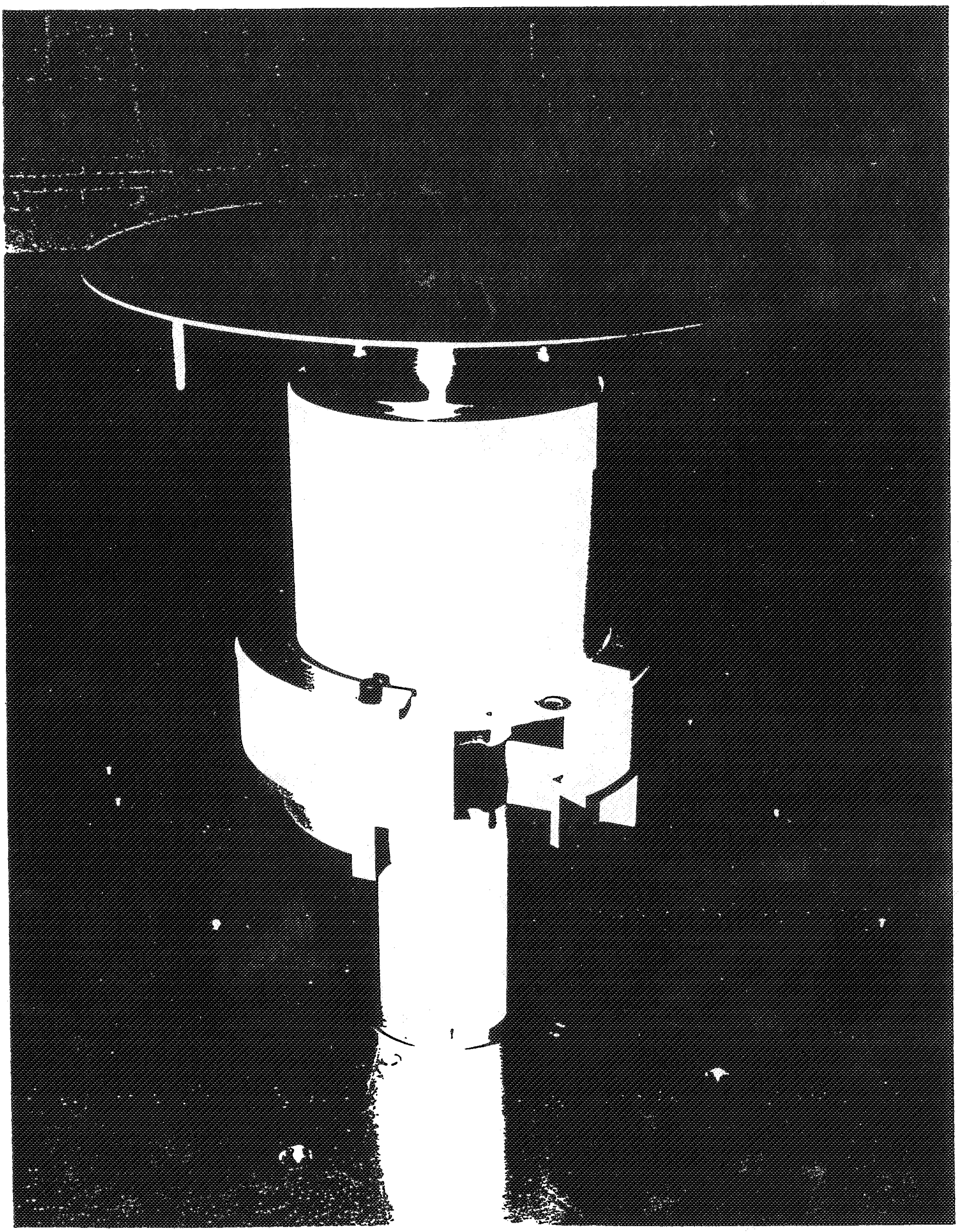
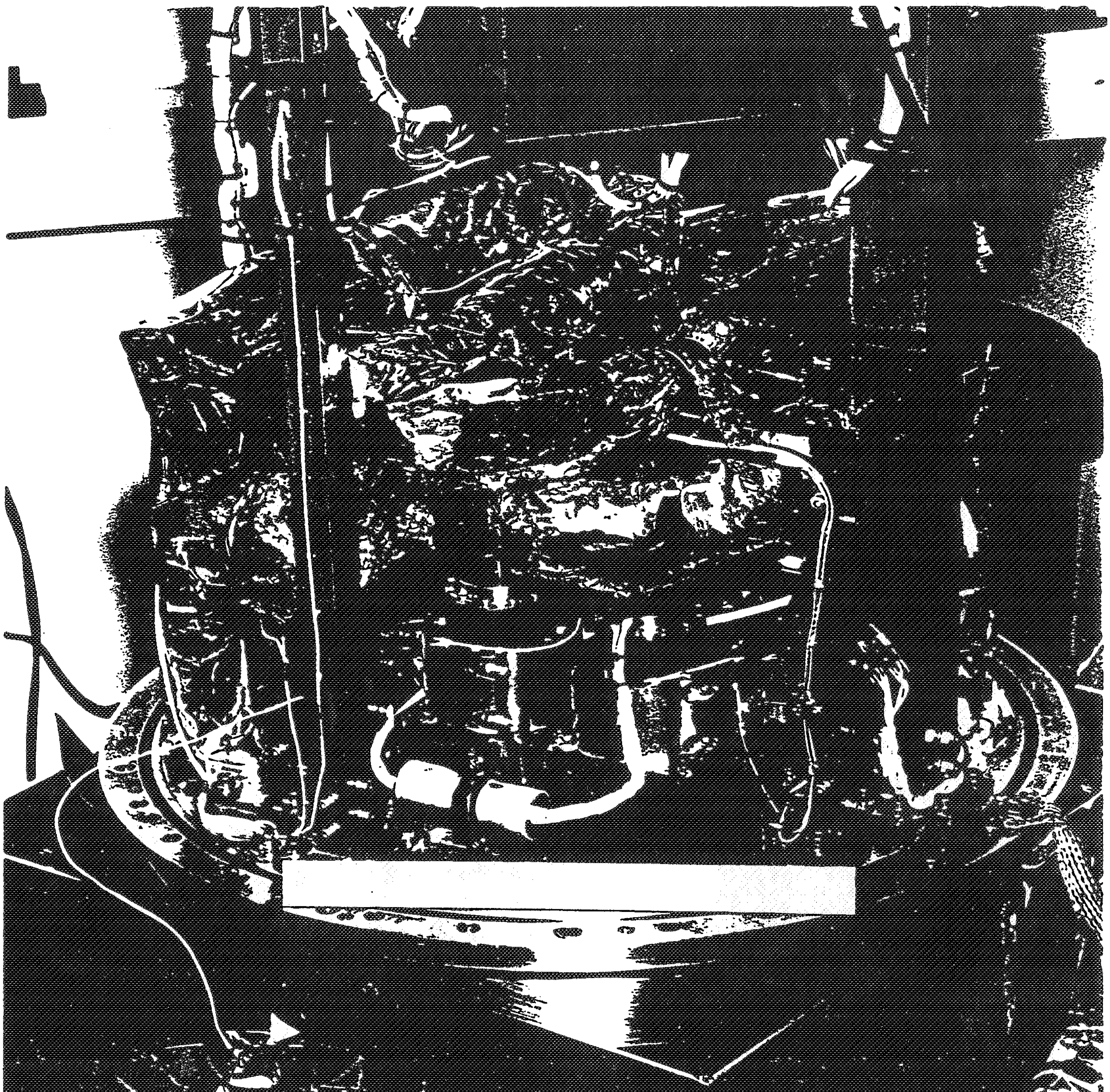
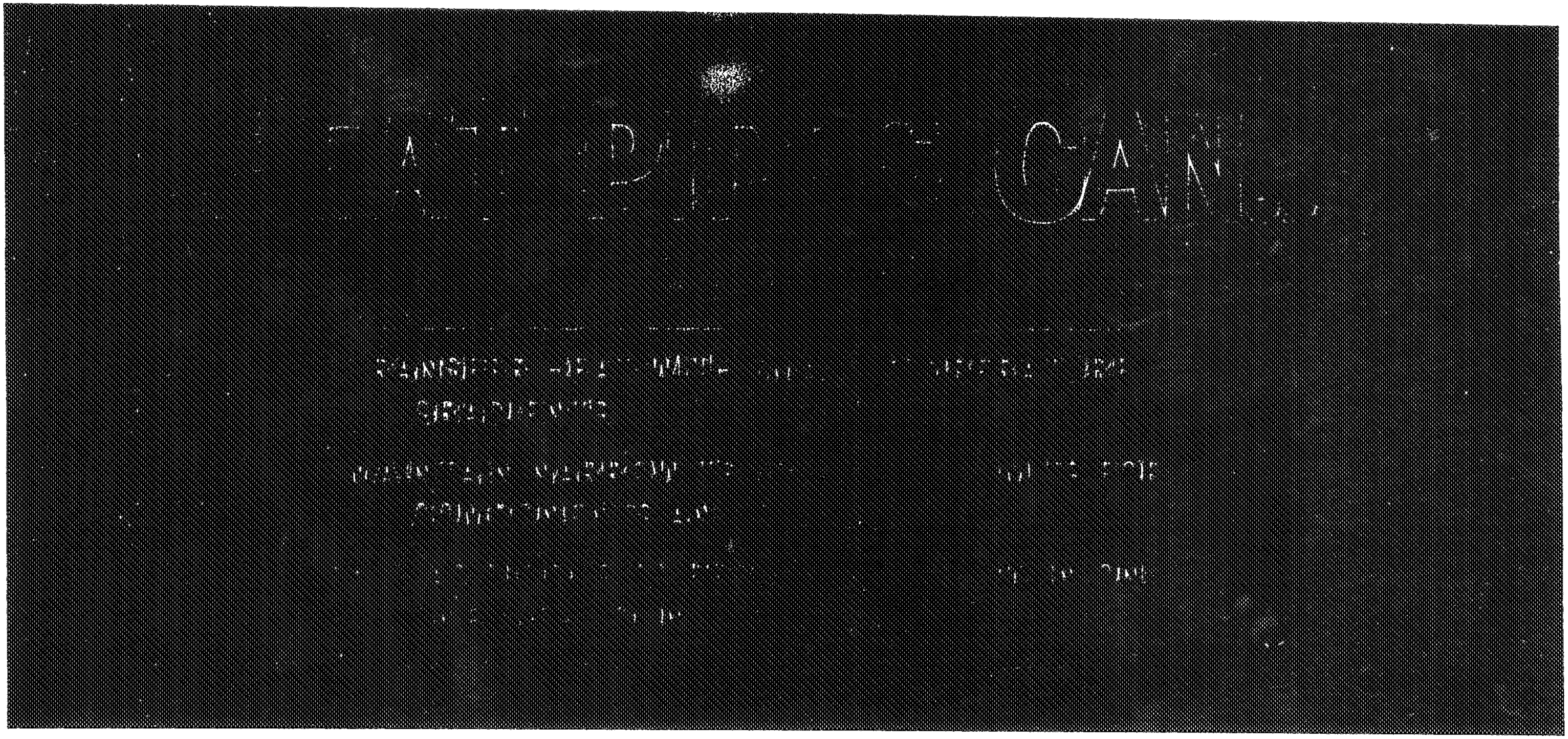






Heat Pipe Performance Experiment

Flight Experiments Technical Interchange Meeting

Sponsored by
Space Technology Interdependency Group
Flight Experiments Committee

October 5-9, 1992
Monterey, CA

George Fleischman
Hughes Aircraft Company

HUGHES

Thermal Energy Storage Flight Experiment in Microgravity

David Namkoong, Principal Investigator
Andrew Szaniszlo, Project Manager / Scientist

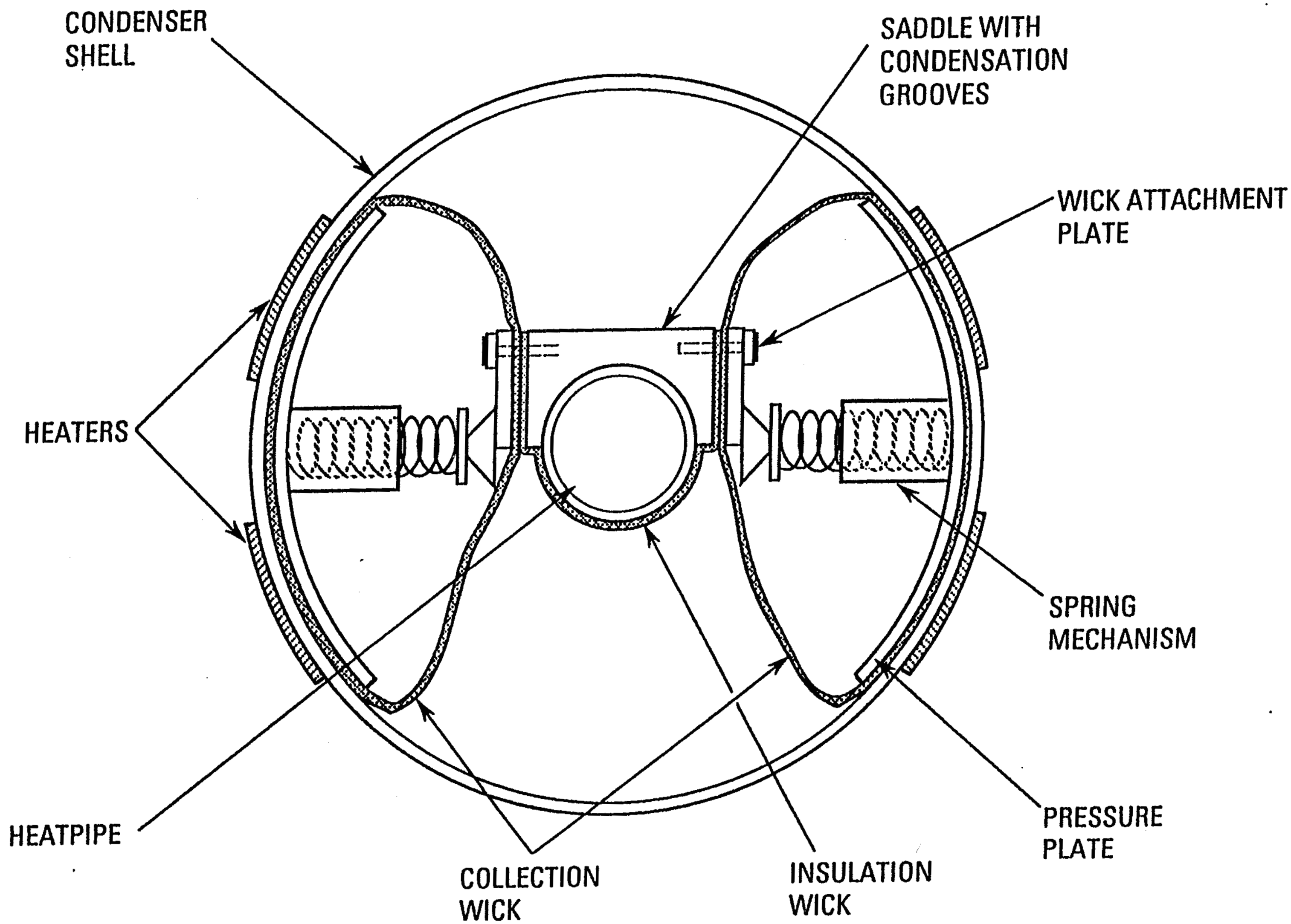
NASA Lewis Research Center
Cleveland, Ohio

Presented at the NASA / DOD Flight Experiments Technical Interchange Meeting, Monterey, California

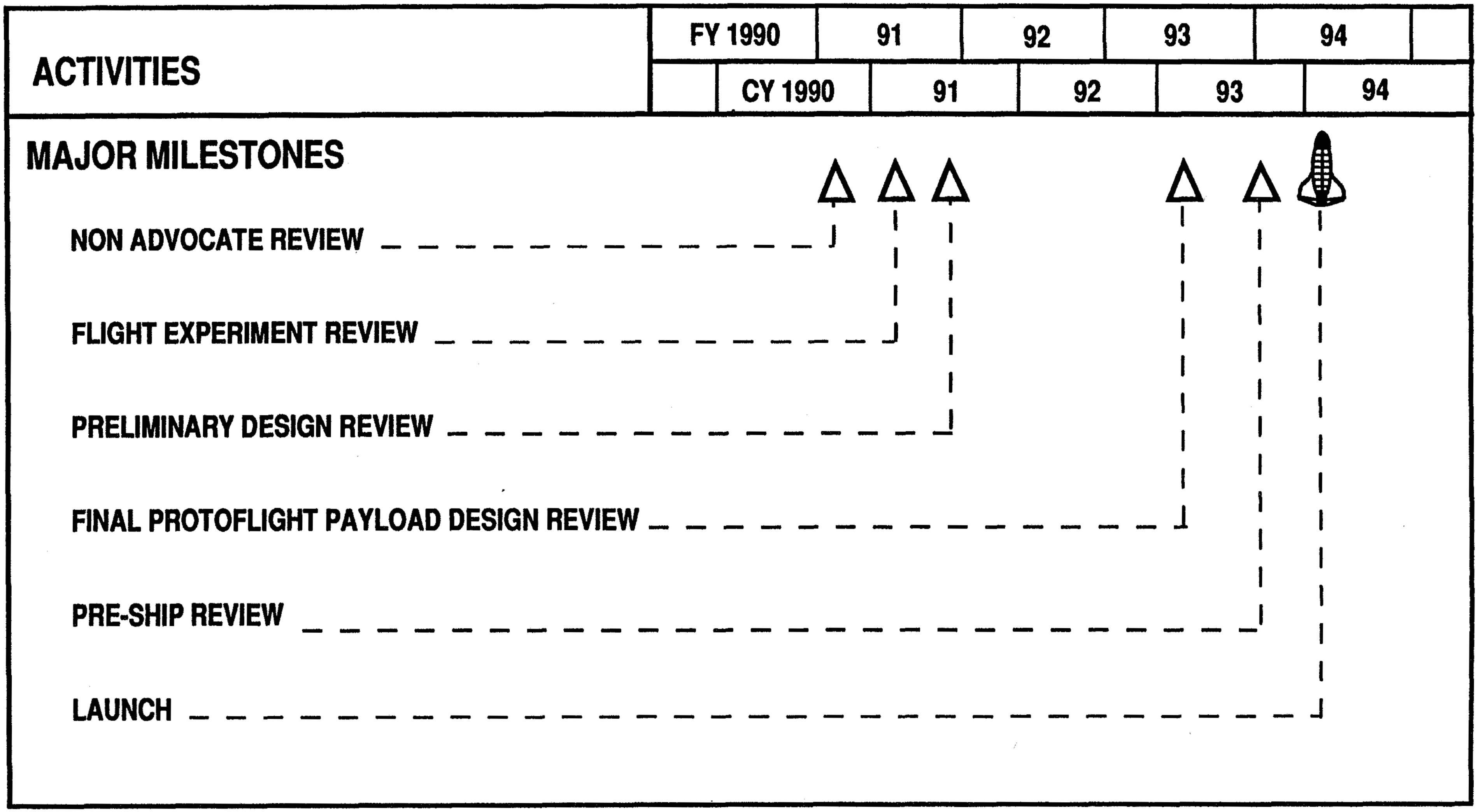
Two-Phase Flow Experiment

TRW

Cross-Sectional View of condensation Chamber



TES FLIGHT SCHEDULE



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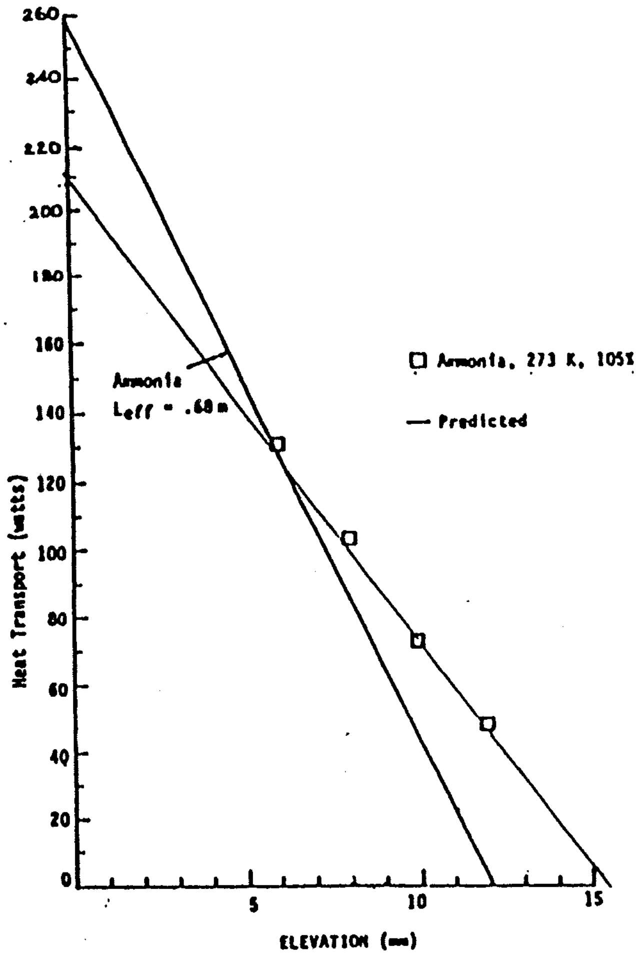
CRYOGENIC HEAT PIPE EXPERIMENT
CRYOHP

N93-28721

ROY McINTOSH
NASA GODDARD SPACE FLIGHT CENTER
OCTOBER 5, 1992

JUSTIFICATION – HEAT PIPE TECHNOLOGY



PERFORMANCE COMPARISON BETWEEN PREDICTIONS AND DATA (REF.: AIAA 77-747)

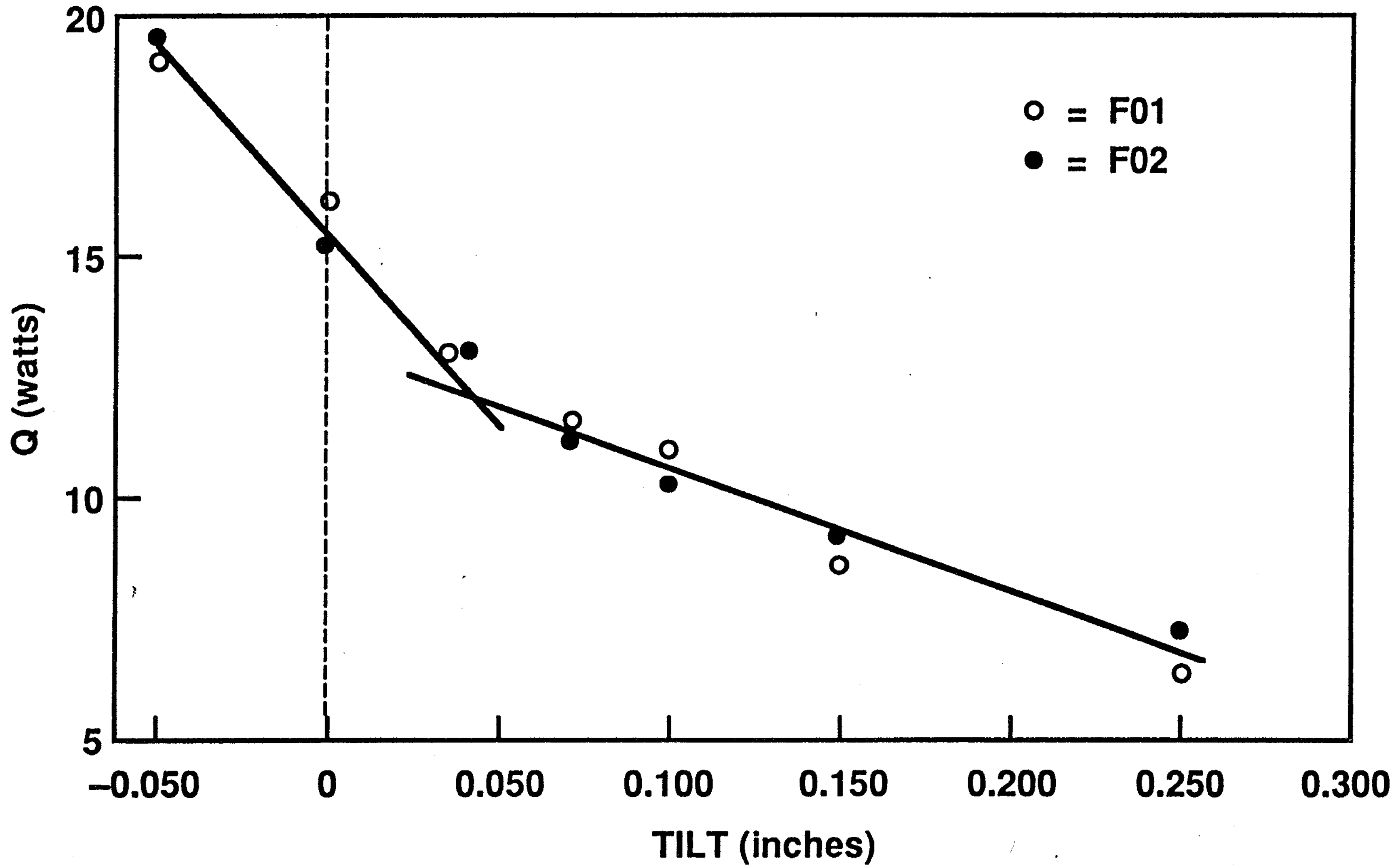
HUGHES

INVESTIGATION OF MICRO-GRAVITY EFFECTS ON HEAT PIPE THERMAL PERFORMANCE AND WORKING FLUID BEHAVIOR (IMEHP)

NASA IN-STEP HEAT PIPE PERFORMANCE (HPP)

40°C FREON HEAT PIPE GROUND TEST DATA

HUGHES



V03669-4

IN-STEP HEAT PIPE PERFORMANCE (HPP) EXPERIMENT

HUGHES

OBJECTIVES – SYSTEM LEVEL

- HOW DOES EXCESS LIQUID IN HEAT PIPES AFFECT SPACECRAFT THERMAL PERFORMANCE

- HOW DO SPACECRAFT ACCELERATIONS BETWEEN 0 AND 1-g AFFECT HEAT PIPE PERFORMANCE

- OBTAIN DATA FOR DESIGN IMPROVEMENTS IN SPACE HEAT PIPES:
  - LIGHTER WEIGHT, RELIABLE, AND MORE EFFICIENT SYSTEMS
  - HANDLING EXCESS LIQUID
  - START-UP AND REWICKING IN SPACE

V02424

IN-STEP HEAT PIPE PERFORMANCE (HPP) EXPERIMENT

HUGHES

TYPICAL SPACECRAFT ACCELERATIONS BETWEEN 0 AND 1-G:

- RENDEZVOUS AND DOCKING
- ASCENT/DESCENT
- ORBITAL MANEUVERING
- LUNAR BASE OR MARS MISSION
- SPINNING/DE-SPINNING
  - SPINNING S/C (e.g., INTERNATIONAL SOLAR TERRESTRIAL PLATFORM, ISTP–8 FT DIA. x 10 RPM)
  - STABILIZATION (TRANSFER ORBIT, GEO-SYNCHRONOUS)
  - SURVEILLANCE
  - HARDENING
- THREAT AVOIDANCE
  - SPACE DEBRIS
  - MILITARY

V02425

IN-STEP HEAT PIPE PERFORMANCE (HPP) EXPERIMENT

HUGHES

APPROACH

- TWO (2) HEAT PIPE CONFIGURATIONS
  - FIXED CONDUCTANCE HEAT PIPES (FCHPs):

    AXIAL GROOVE DESIGN; MOST COMMON DESIGN SPECIFIED FOR COMMUNICATIONS, SURVEILLANCE, SCIENTIFIC, AND OTHER SPACECRAFT

  - VARIABLE CONDUCTANCE HEAT PIPES (VCHPs):

    POROUS WICK DESIGN; CURRENTLY BEING USED ON HUGHES HS-111 SPACECRAFT, AND PROPOSED FOR FUTURE MISSIONS

- HEAT PIPE WORKING FLUIDS/MATERIALS:
  - WATER/COPPER (16 EACH)
  - FREON – 113/ALUMINUM (2 EACH)
  - VARIOUS FILL FRACTIONS (90 TO 120 %)

V02427

IN-STEP HEAT PIPE PERFORMANCE (HPP) EXPERIMENT

HUGHES

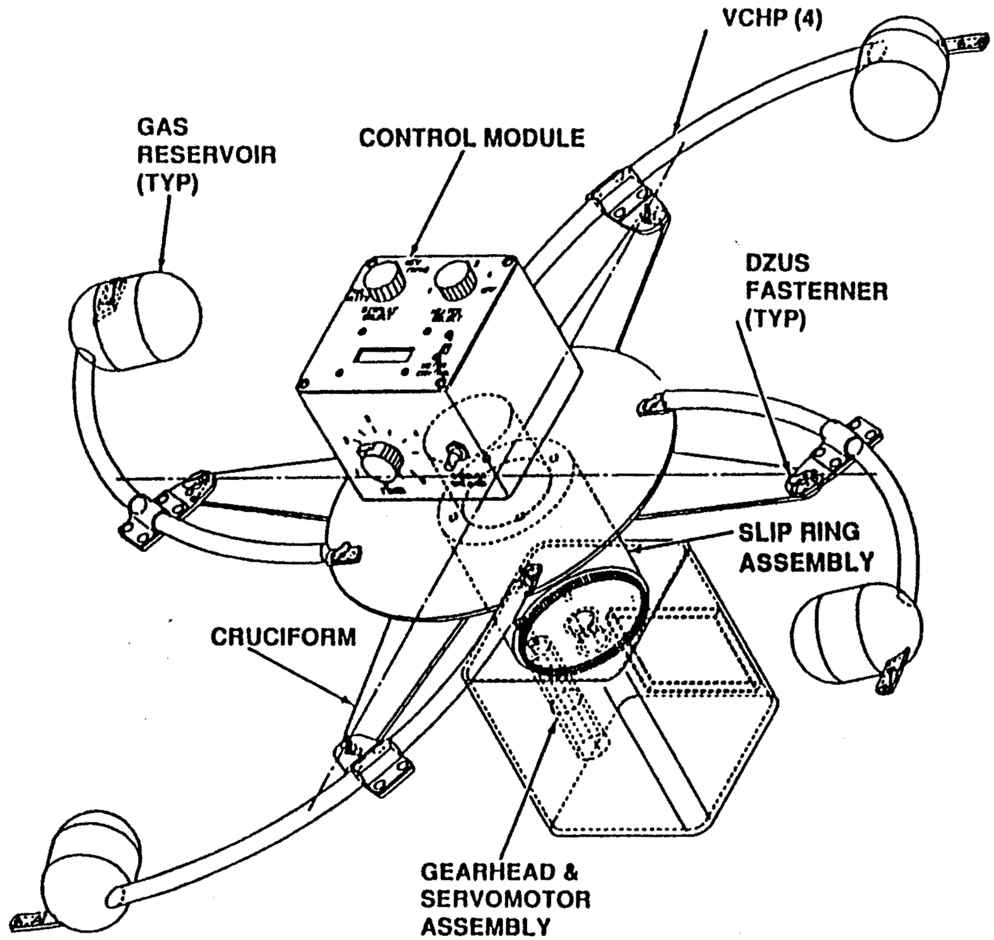
APPROACH (CONTINUED)

- THERMAL PERFORMANCE
  - STATIC
  - SPIN
  - REWICKING

V02428

IN-STEP HEAT PIPE PERFORMANCE (HPP) EXPERIMENT

HUGHES

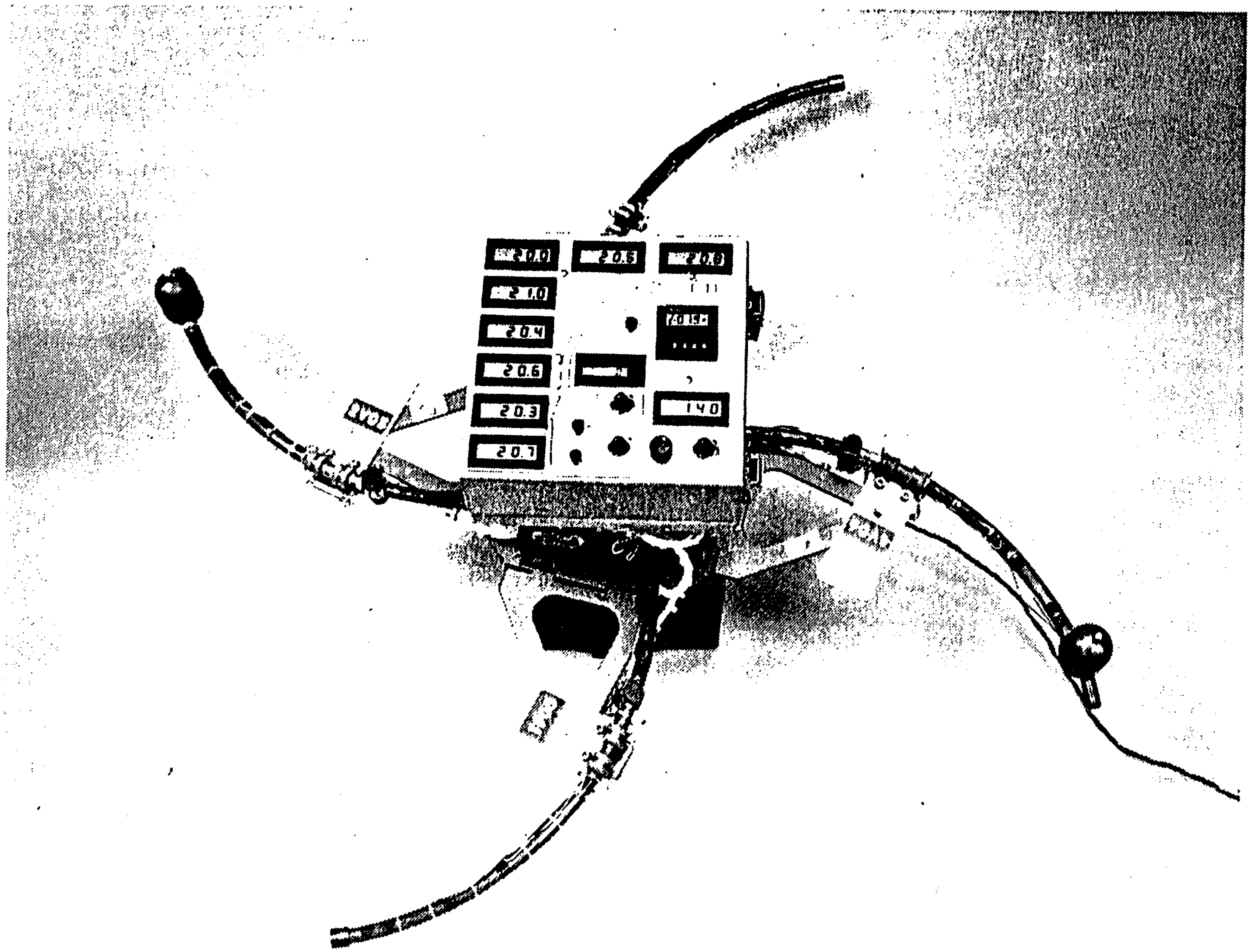


THERMAL PERFORMANCE MODEL

V0611

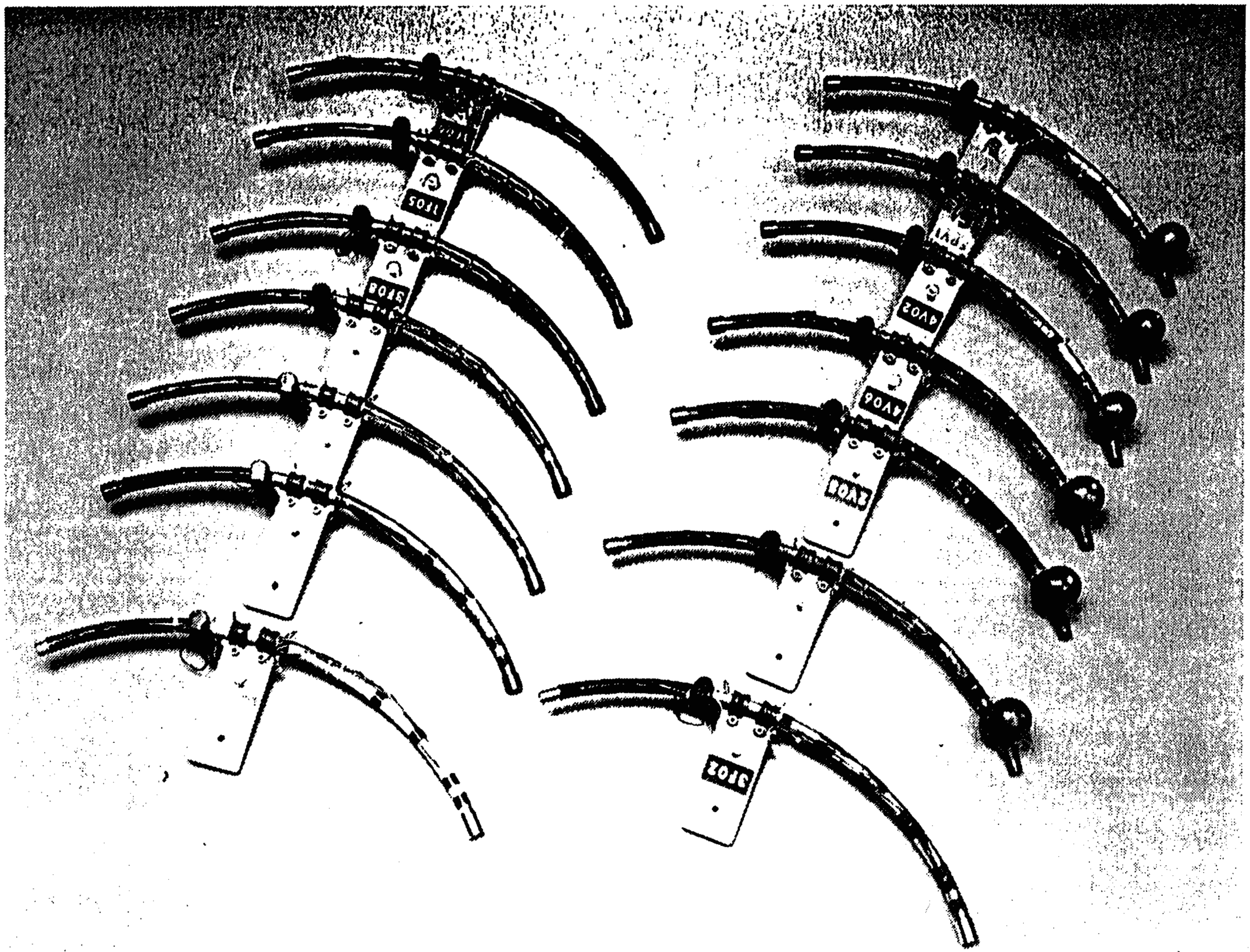
NASA IN-STEP HEAT PIPE PERFORMANCE (HPP) FLIGHT EXPERIMENT APPARATUS (WITHOUT SHROUD)

HUGHES



NASA IN-STEP HEAT PIPE PERFORMANCE (HPP) FLIGHT HEAT PIPES

HUGHES



NASA IN-STEP HEAT PIPE PERFORMANCE (HPP) FLIGHT EXPERIMENT APPARATUS

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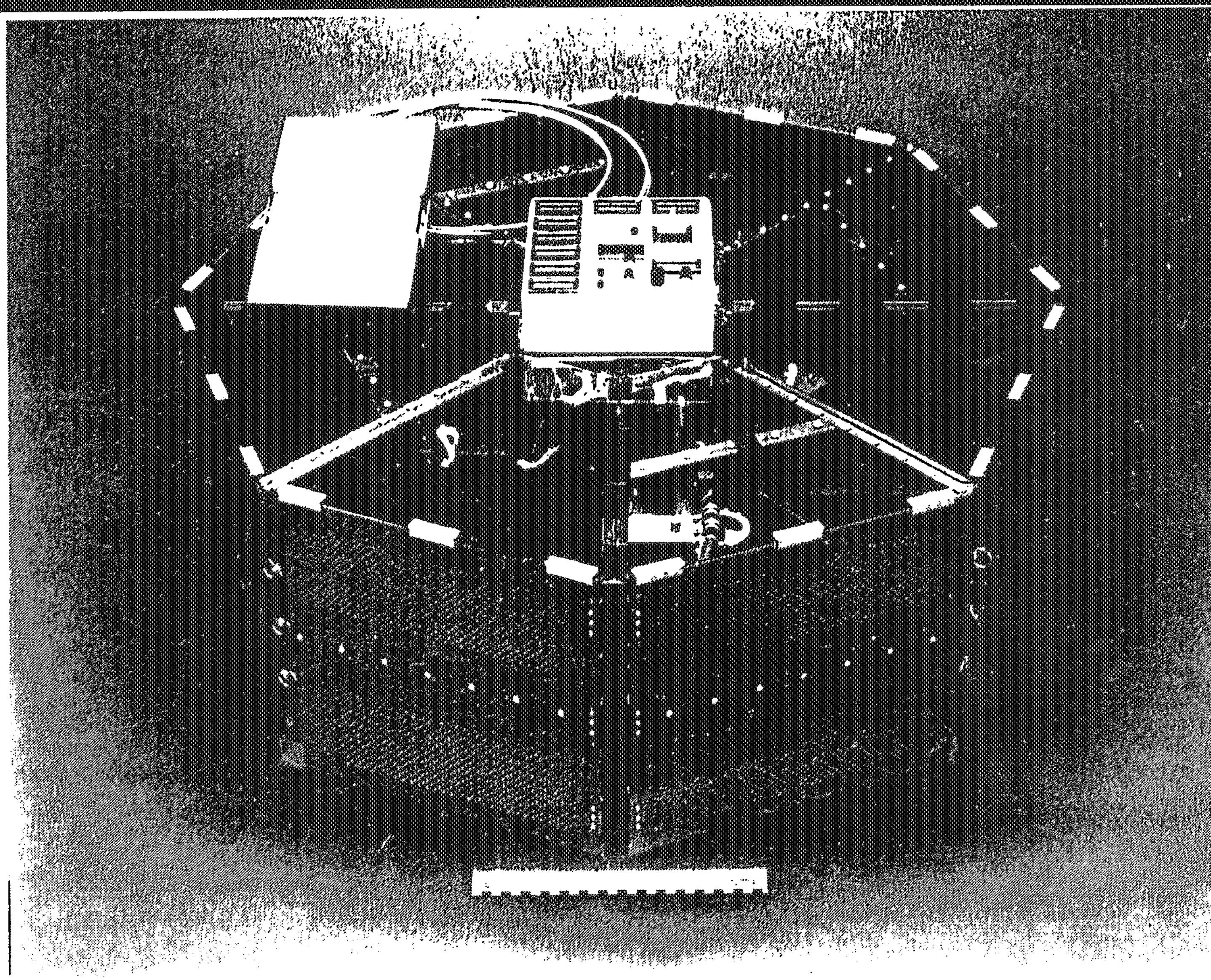
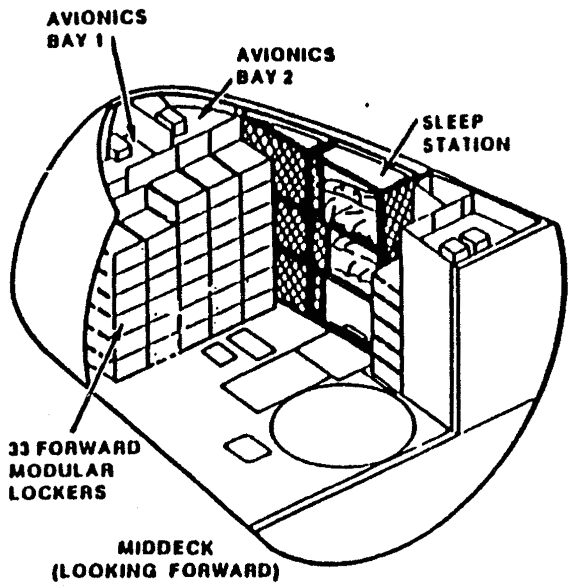
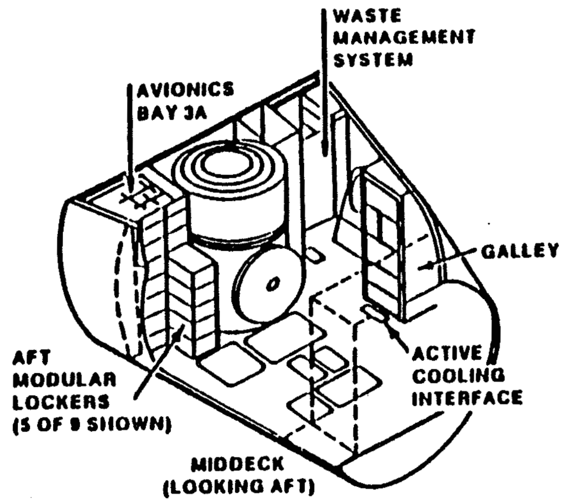
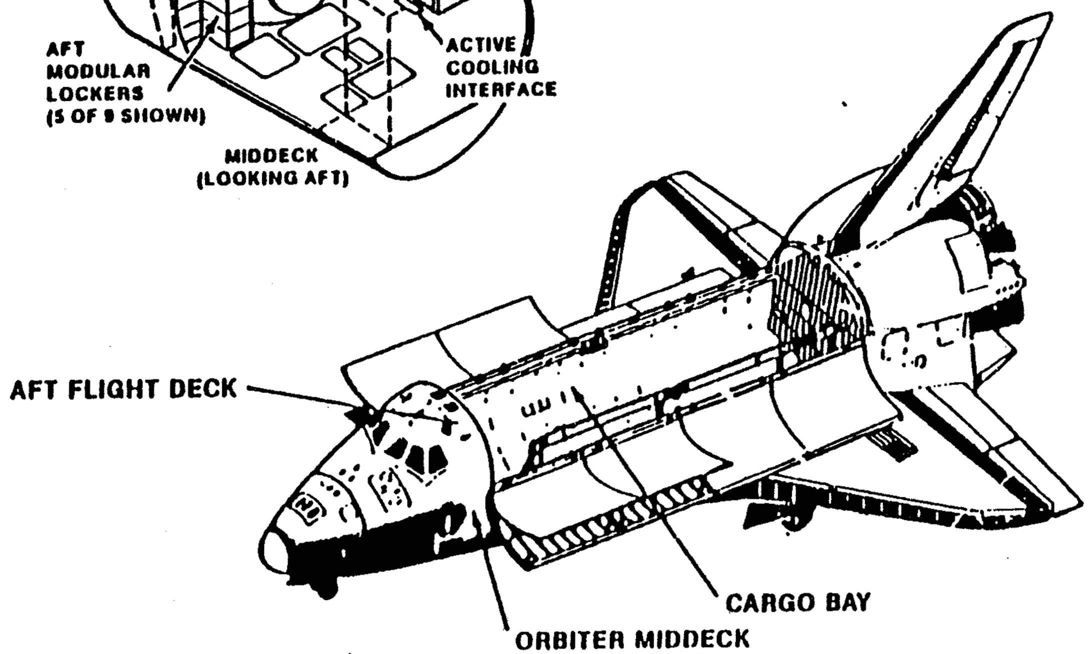


NSTS Shuttle Orbiter Middeck Configuration







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IN-STEP HEAT PIPE PERFORMANCE (HPP)

MIDDECK MODULAR LOCKERS

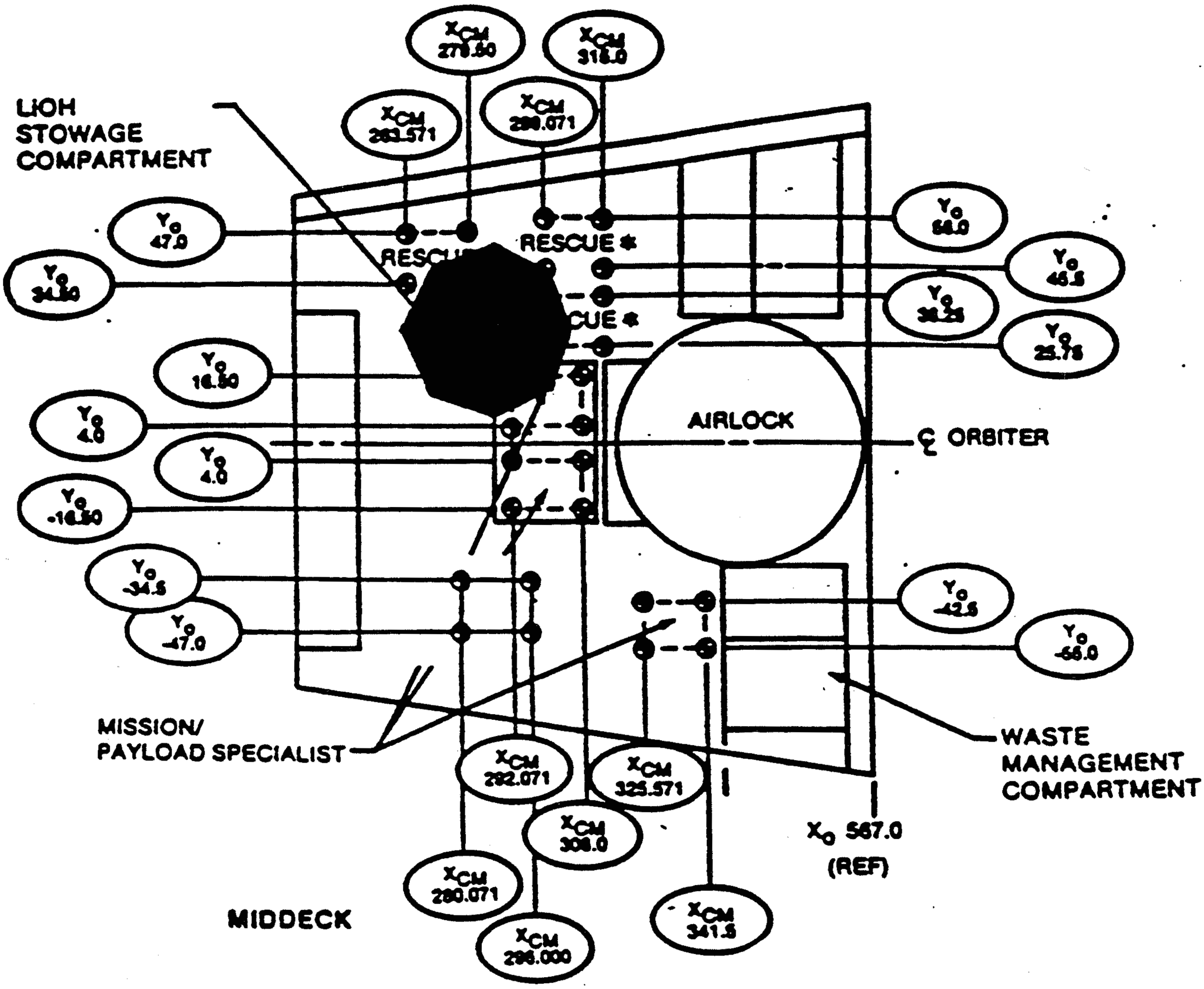
Heat Pipe Performance Experiment
Weight and Power Summary

Item	Quantity	Weight (lbs)	Orbital Power (watts)
Heat Pipe, Warmers,* and Thermostats**	14	10.9	<60.0
Control/Motor Module	1	27.7	<32.0
Safety Shroud, Quarter Pieces with Brownline Fittings	1	27.8	
Data Loggers and Cables	2 ea	9.4	
Batteries, Data Logger (Spares)	4	4.0	
HPP Tool Kit***	1	~5.0	
Crowmember Deerskin Gloves	1 pr	0.25	
DC Power Cables	1	~1.0	
35mm Film, Kodak 5017	2	0.41	
VIU with Cables	1	2.71	
Video Camcorder Assembly	1	3.3	9.6
Camcorder Videocassettes	12	2.0	
Camcorder Batteries (Spares)	8	5.0	
Total Weight		99.2	
Total Power			<103.6

*One (1) per heat pipe.
**Two (2) per heat pipe warmer.
***Includes spare circuit board interface assembly and motor amplifier board assembly.

Table 2. NASA Supplied Equipment Flown in HPP Lockers

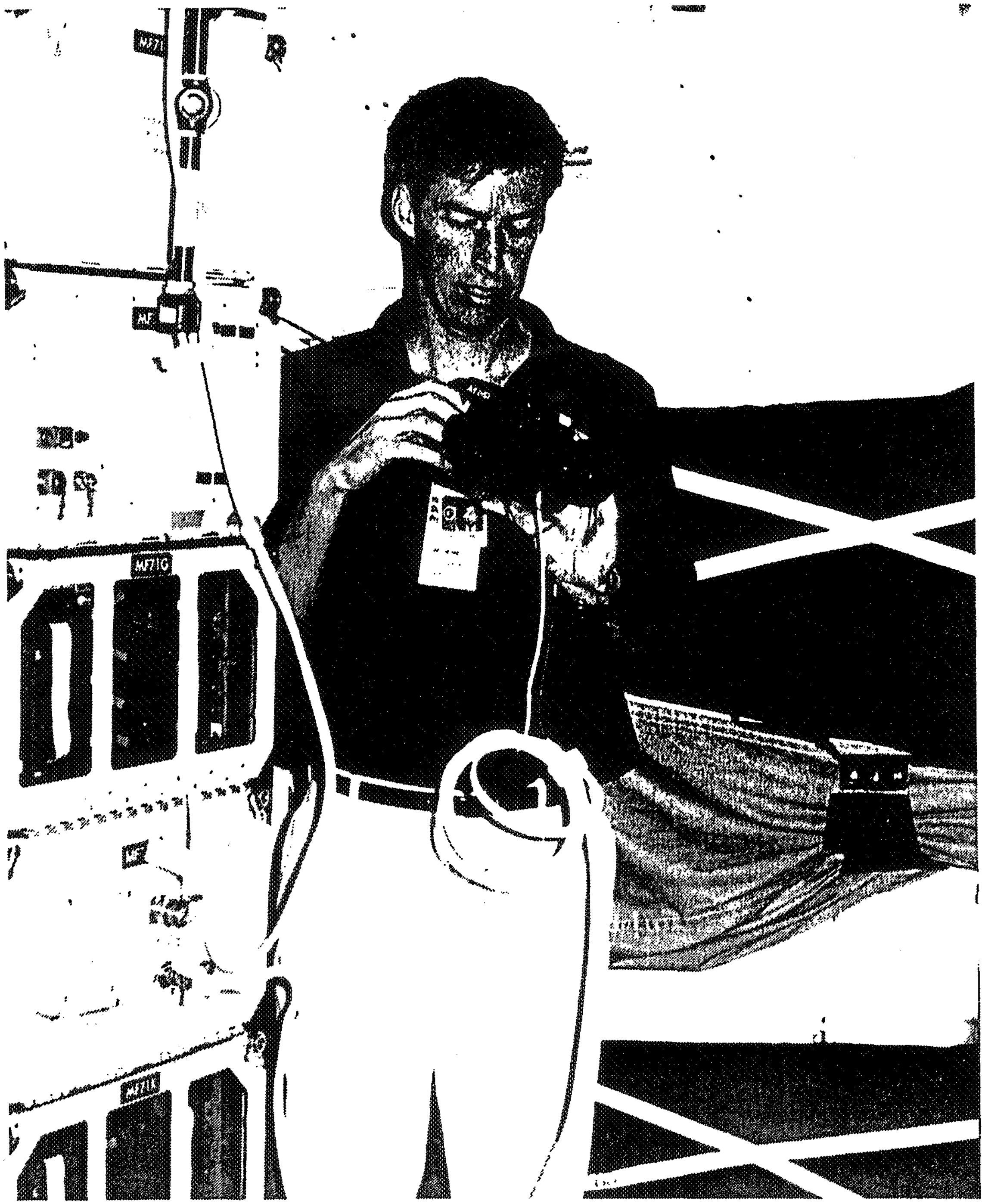
Item	JSC Part No.	Qty	Usage
1. Video Camera	SED33103370-301	1	Video Recording
2. Video Interface Unit	SED39121272-301	1	Camera Adapter
3. Video Cable	SED39122102-301	1	Camera Power/Signal
4. Deerskin Gloves	TBS	1 pr	Hand Protection



HPP Apparatus Mounted to Middeck Floor

STS-52 MISSION SPECIALIST
DR. T. JERNIGAN

HUGHES



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STS-52 COMMANDER

J. WETHERBEE

- Simple Closed Form Heat Pipe Models (Cotter's Equations):

$$\Delta P_{cap} - \Delta P_c \geq \Delta P \ + \Delta P_v$$

Where:
ΔP_{cap} = Capillary Pressure Head
ΔP_c = Centrifrugal Pressure Head Due to Rotation
ΔP = Liquid Pressure Drop
ΔP_v = Vapor Pressure Drop

- Wick Capillary Pumping Based on Eninger's Approach for Fibrous Wicks:

$$\Delta P_{cap} = 6.36 \ \frac{(1-\varepsilon)}{\varepsilon} \ \frac{\sigma}{d_w}{}^{*}$$

Where:
d_w = Wire Diameter
ε = Wick Porosity
σ = Surface Tension

*AIAA Paper No. 75-661

IN-STEP HEAT PIPE PERFORMANCE (HPP) EXPERIMENT

HEAT PIPE THERMAL PERFORMANCE ANALYSIS (CONTINUED)

HUGHES

- **Wick Pressure Drop Based on Darcy's Equation:**

$$\Delta P_\ell = \frac{\mu_\ell Q L_{eff}}{K_w A_w h_{fg} \rho_\ell}$$

Where: $K_w = \dfrac{d_w^2}{122} \dfrac{\varepsilon^3}{(1-\varepsilon)^2}$

K_w = Wick Permeability

μ_ℓ = Liquid Viscosity

h_{fg} = Latent Heat of Working Fluid

Q = Heat Transport

L_{eff} = Effective Heat Pipe Length

- **Groove and Vapor Pressure Drops Based on Channel Flow:**

$$\Delta P_{cap} = \frac{2\sigma}{w_g}$$

w_g = Groove Width

$$\Delta P_{\ell,v} = \frac{2\,(fRe)_{\ell,v}\mu_{\ell,v} Q L_{eff}}{(d_h^2)_{\ell,v} A_{\ell,v} \rho_{\ell,v} h_{fg}}$$

fRe = Constant (Laminar Flow)

$d_h = \dfrac{4A}{P_w}$

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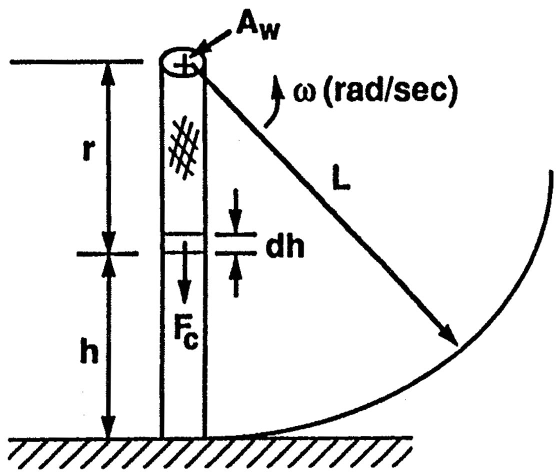
HEAT PIPE THERMAL PERFORMANCE ANALYSIS (CONTINUED)

Centrifugal Force Due to Spinning:



r = Radius About Axis of Rotation
L = Heat Pipe Length
A_W = Wick Cross-Sectional Area
ρ = Liquid Density
ω = Angular Velocity

Centrifugal Force (F_c) on Element at Radius, r:

$$dF_c = r\,\omega^2\,dm$$
$$dF_c = \rho\,\omega^2\,r\,A_W\,dh$$

Pressure Drop at Radius, r, Due to Centrifugal Force on Element:

$$\frac{dF}{A_W} = dp = \rho\,\omega^2\,r\,dh$$
$$r = L - h$$
$$dp = \rho\,\omega^2 (L - h)\,dh$$

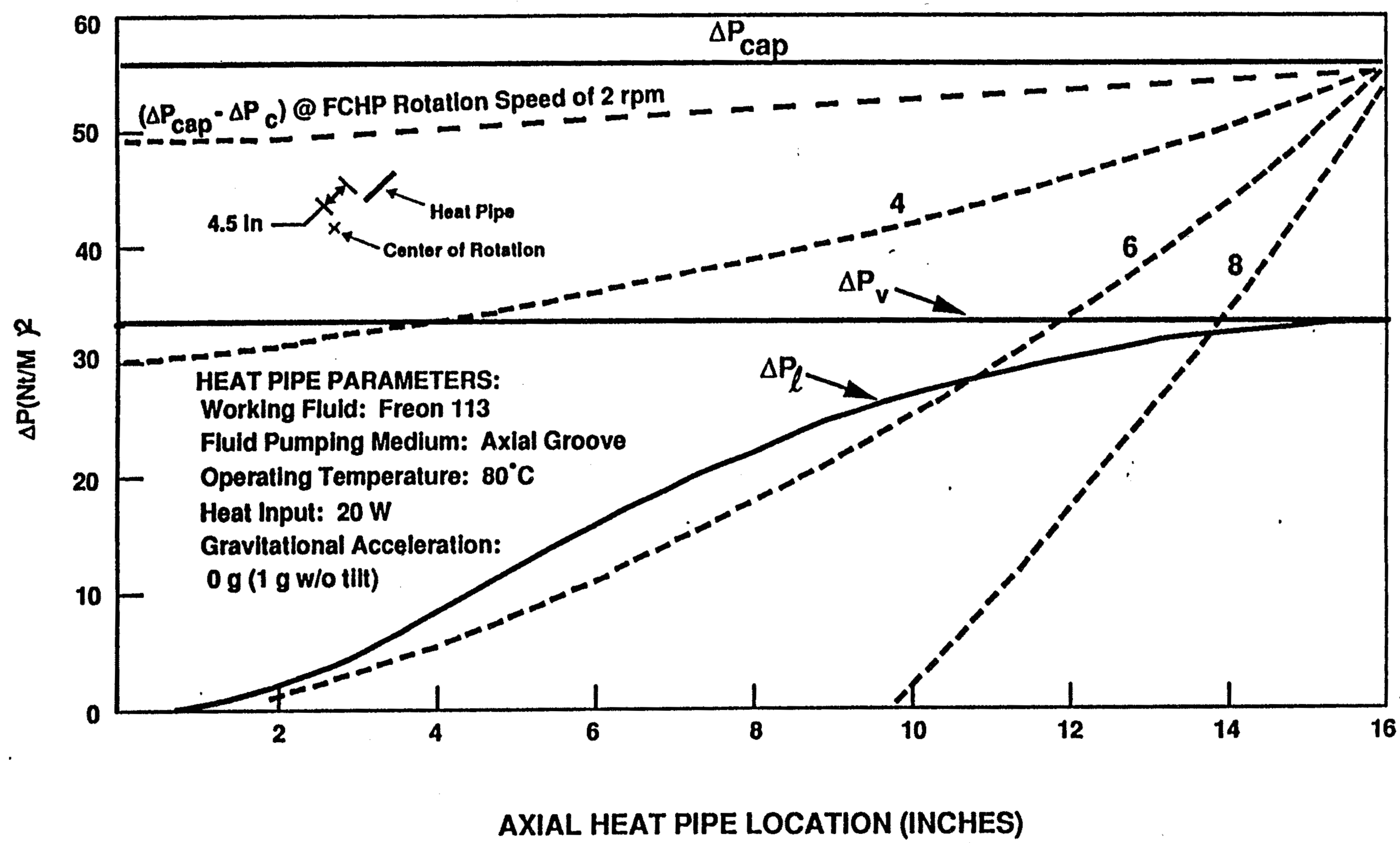
Pressure Drop at Radius, r, Due to Centrifugal Force on Liquid column of Length, h:

$$\Delta P_c = \rho\,\omega^2 \int_o^h (L - h)\,dh$$
$$\Delta P_c = \rho\,\omega^2 h\left(L - \frac{h}{2}\right)$$

IN-STEP HEAT PIPE PERFORMANCE (HPP) EXPERIMENT
HEAT PIPE THERMAL PERFORMANCE ANALYSIS

HUGHES

PRESSURE HEAD DIFFERENTIAL ARISING FROM FCHP ROTATION

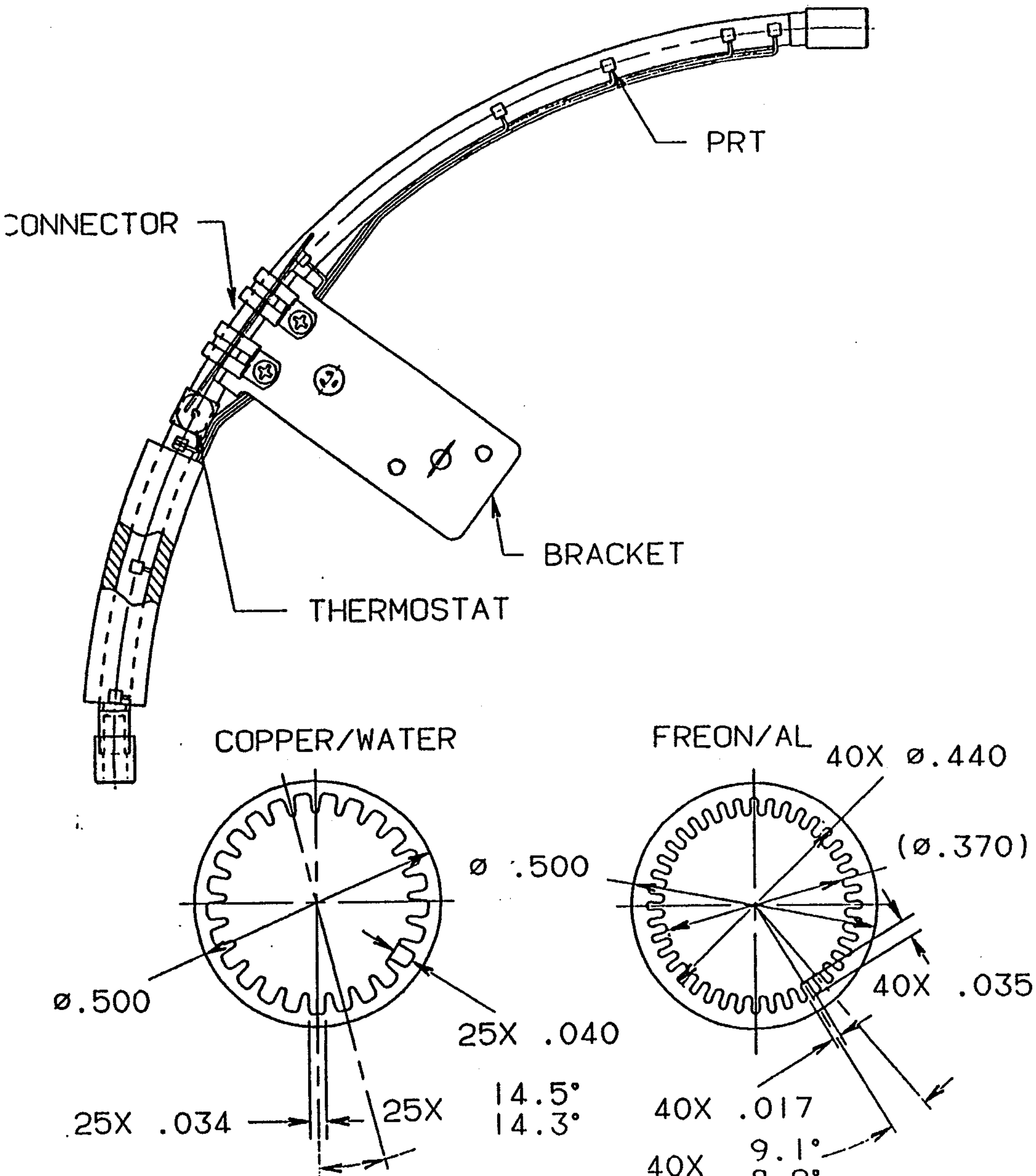


V03485-D

Description of Experimental Pipes

Item	Fixed Conductance Heat Pipes (FCHPs)		Variable Conductance
	Copper - Water	Aluminum - Freon	Heat Pipes (VCHPs)
Envelope:			
Outside Diameter (in.)	0.50	0.50	0.50
Wall Thickness (in.)	0.020	0.030	0.020
Active Length (in.)	17.0	17.0	17.0
Gas Reservoir	None	None	1.5 OD x 2.5*
Material	OFE Copper	6063-T6 Aluminum	OFE Copper
Wick:			
Description	25 axial grooves electro-discharge machined in envelop wall	40 axial grooves extruded in envelop wall.	80% porosity knitted mesh center core wick with three 70% porosity spacer wicks; 1-layer #100 mesh evaporator wall wick.
Dimension (in.)	0.040 depth 0.034 width	0.035 depth x 0.017 width	0.34 dia. center core; 0.090 square spacers
Material	N/A	N/A	Copper
Working Fluid	Triply Distilled Water	Freon-113	Triply Distilled Water
Nominal Fluid (gm)	7.7	8.8	23.2
Liquid Fill Fractions (%)	90,100,105	100	90,100,120
Weight per pipe, dry (lbm)	0.32	0.15	0.56
Quantity (Flight)	6	2	6

* Two (2) VCHPs will have non-condensible gas and four (4) will not. Reservoir has hemispherical end caps.

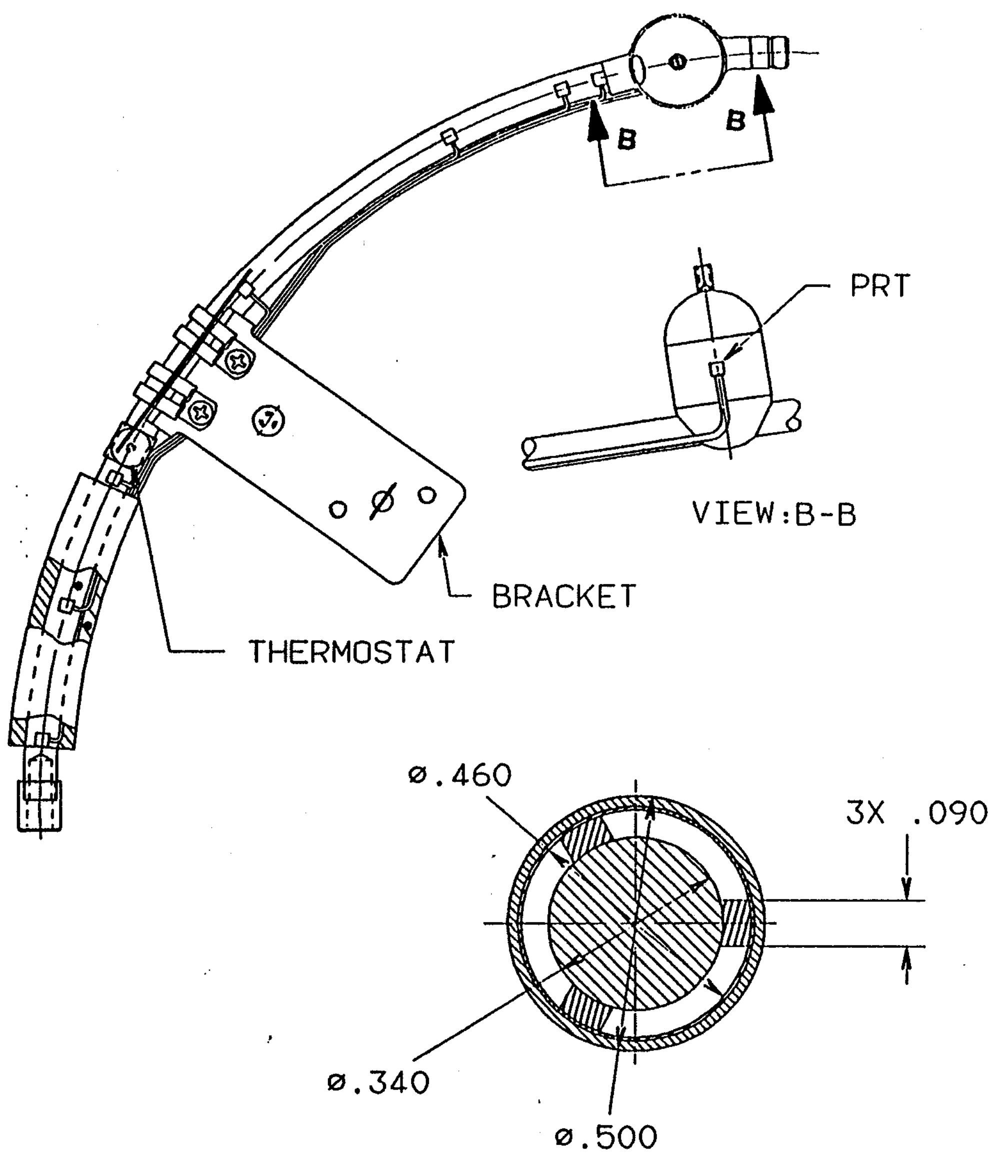


V03669-1

HUGHES

NASA IN-STEP HEAT PIPE PERFORMANCE (HPP)

FIXED CONDUCTANCE HEAT PIPE



V03669-2

HUGHES

NASA IN-STEP HEAT PIPE PERFORMANCE (HPP)

VARIABLE CONDUCTANCE HEAT PIPE

Heat Pipe Pressure Characteristics

	Nominal Operating Pressure (PSIA) at 65°C	Maximum Design Pressure (PSIA) at 85°C*	Minimum Required Burst Pressure (PSIA)**	Calculated Burst Pressure (PSIA)	Safety Factor ***	Proof Pressure (PSIG)	Qty.
FCHP Freon 113/ Al 6063-T6	24.7	44.7	111.8	1149	26	223.5	2
FCHP Water/ Copper	3.6	8.4	21.0	1667	198	42.0	6
VCHP Water/ Copper	3.6	8.4	21.0	1129	134	42.0	6

* *This pressure results from the temperature corresponding to the worst-case two-failure condition.*

** *This pressure corresponds to the MDP with a safety factor of 2.5*

*** *Based on the temperature corresponding to MDP (i.e. SF = calculated burst pressure/MDP)*

NASA IN-STEP HEAT PIPE PERFORMANCE (HPP) COMPONENT STRESS ANALYSIS SUMMARY

HUGHES

Component	Material	Max. Stress (kPa)	Ult. Stress (kPa)	Factor of Safety
Motor Mount				
Screw A	Steel	30.3	517	17.1
Screw B Mount	Aluminum	3.69	75.8	20.6
Screw C	Steel	120	517	4.3
Heat Pipe Bracket DZUS Fastener	Steel	4.44	317	71.6
Cruciform Arm Hinge Pin (2)	Steel	6.25	317	50.7
Fan Blade	Steel	21.7	193	8.9
Safety Shroud	Lexan	43.1	65.5	1.5

V03669-3

HPP EXPERIMENT FEATURES
PAYLOAD INTEGRATION

HUGHES

- Mechanical Integration
  - Modular Design Facilitates Assembly and Stowage (Nominal and Emergency)
  - Hardware Stows in Three Middeck Locker Drawers
  - HPP Complies with NSTS 21000-IDD-MDK Weight and c.g. Requirements
  - Total HPP Weight: 99.2 lbs (36 lbs Maximum in Single Locker)

- Electrical Integration
  - Total Orbiter DC Electrical Power Used by HPP: 104 Watts (Maximum)
  - Fuse (5 amp) at Payload/Orbiter Interface Protects Against HPP Failure
  - SSP-Provided Power Cord (15 ft Length) Connects HPP to Orbiter
  - Relay Prevents Inadvertant Activation Upon Power Cord Connection

HPP EXPERIMENT FEATURES THERMAL, MECHANICAL DESIGN

HUGHES

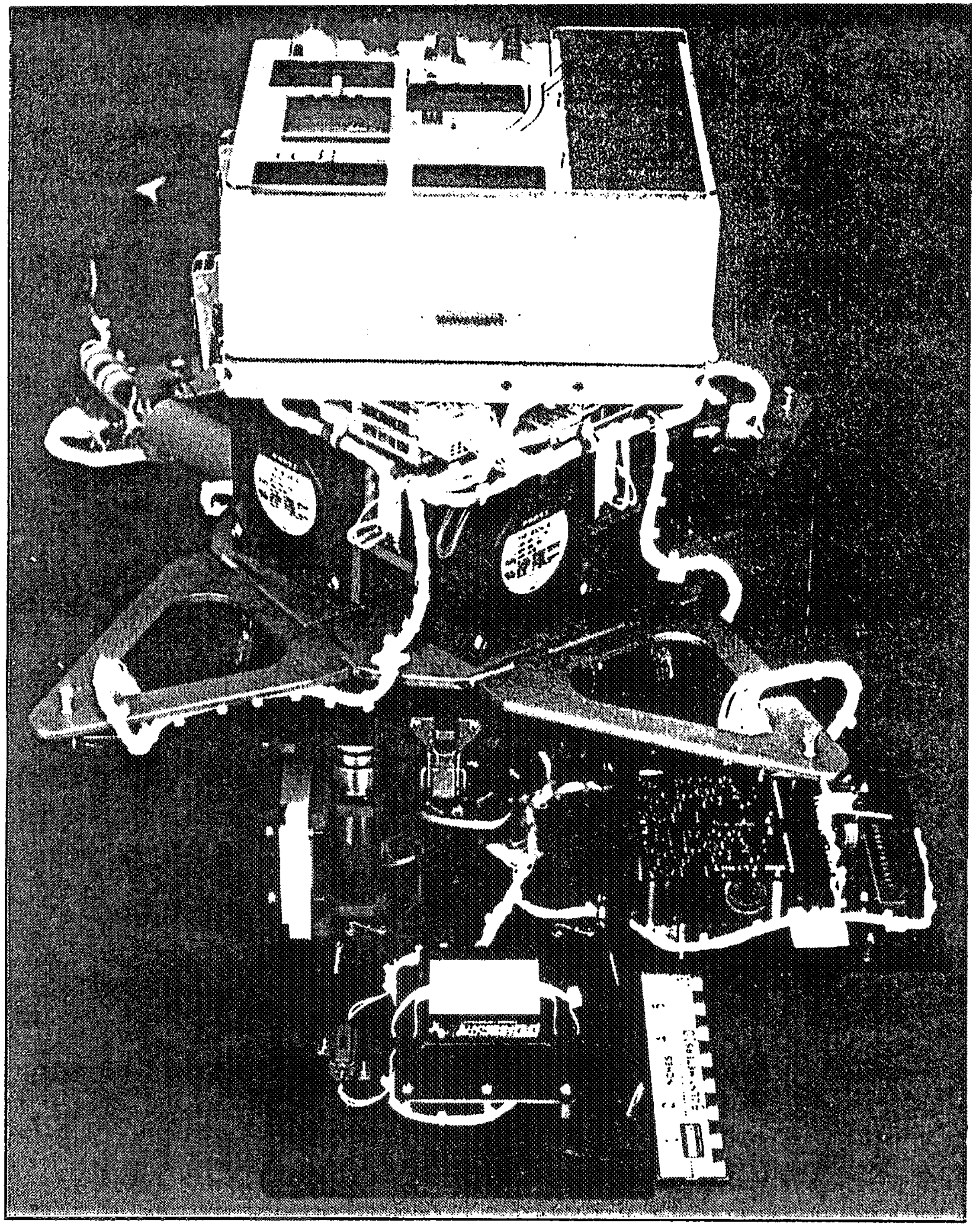
- Thermal Design Features
  - Redundant Thermostats (Elmwood 3200) Mounted on Each Heat Pipe Preclude Overheating
  - Active Cooling Uses 35 CFM Fan (Pabst 8124 G) on Each Pipe
  - Aluminum and Lexan 9600 Safety Shround Prevents Contact with Surfaces at Elevated Temperatures (up to 85°C)

- Mechanical Design Features
  - Heat Pipes Incorporate Large Safety Factors ($P_{BURST}/P_{MAX} > 25$)
  - Rotating Structure Safety Factors Exceed 1.4 Ultimate
  - Collision Hazard Prevented by Safety Shroud, Internal Clutch, and Low Angular Momentum; Safety Factor Exceed 1.4 Ultimate
  - Design Contains no Fracture Critical Components as Defined by NHB8071.1 (19,307 Joules Criterion)

IN-STEP HEAT PIPE PERFORMANCE (HPP)
ELECTRICAL DESIGN

HUGHES

- Design Constraints
  - 115 Watt Power Limit
  - 28±4 VDC Power Supply From Spacecraft Bus
  - Specified Low Conductive and Radiative EMI
- Design Features
  - Switching Voltage Regulators Versus Analog for High Efficiency, Constant Output Regardless of Input Voltage Variation.
  - LCDs for Low Power Consumption
  - Each Controller Designed with Schottky Diode EMI Suppression
  - EMI Block Filter Installed on Input Line
  - Brushless DC Motor Used on Fans and Drive Mechanism
- Safety Features
  - Specified NASA Approved Circuit Breakers
  - Warmer/Fan Interlock
  - Dual Thermostats for Heat Pipe Over-Temperature Protection
  - Fail-Safe Start-up
  - Exclusive Operation of a Single Heat Pipe and Fan at Any Time



HUGHES

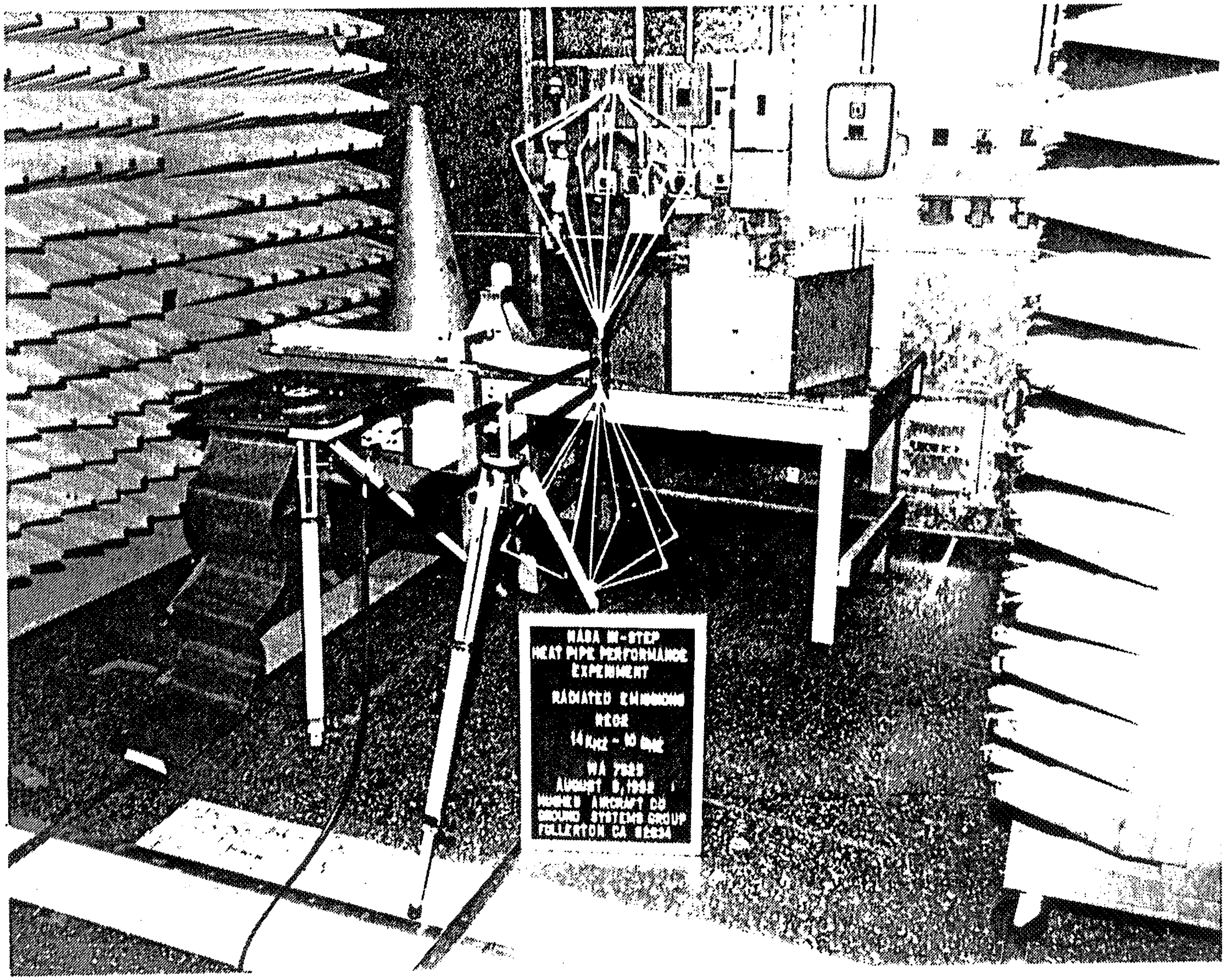
NASA IN-STEP

HEAT PIPE PERFORMANCE (HPP) EXPERIMENT

CONTROL/MOTOR MODULE

NASA IN-STEP HEAT PIPE PERFORMANCE (HPP) ELECTROMAGNETIC COMPATIBILITY TESTS

HUGHES



HEAT PIPE PERFORMANCE (HPP) THERMAL PERFORMANCE EXPERIMENT

HUGHES

- Total HPP Duration is 18 Hours not Including Set-up and Restowage Time
- HPP Includes Three Types of Experimental Runs:

1) Static Testing-at 0 RPM (no Rotation) Heat Pipes are Warmed at Increasing Power Levels Intil:

a) Dryout Occurs;

b) 40 W Maximum Power Level is Reached; or

c) Heat Pipe Temperature Exceeds 85° C

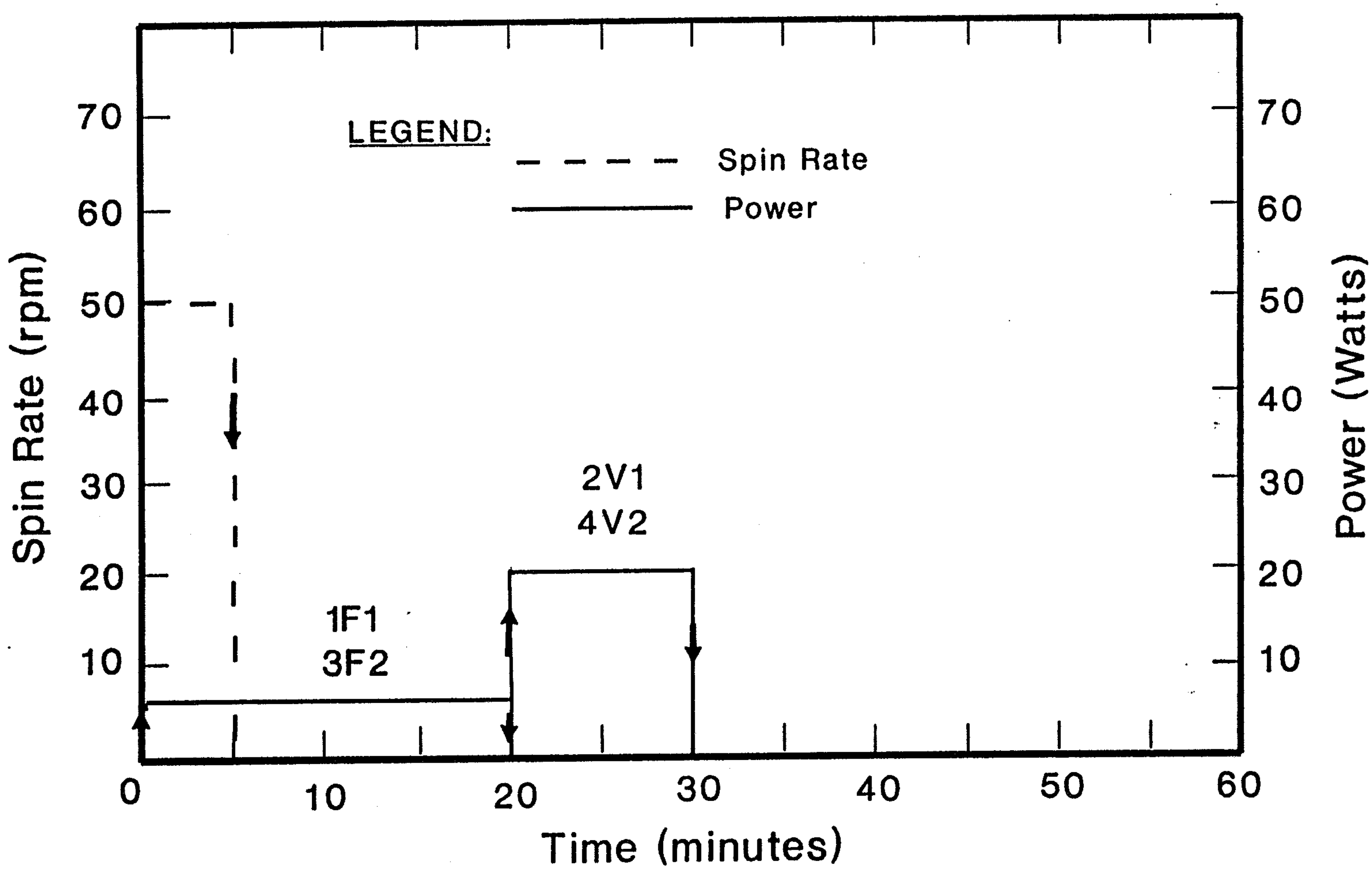
2) Spin Testing-at Warmer Power of 20 to 40 W, Spin Rates are Increased Until:

a) Dryout Occurs; or

b) 30 RPM Spin Rate is Reached

3) Rewicking Testing – See Next Viewgraph

HPP REWICKING TEST TIMELINE



HPP TEST MATRIX

RUN	TEST	HEAT PIPE	FLUID	POWER LEVEL	TEST INCREMENT	SPIN RATE	WARM TIME	RUN TIME
A1	STATIC	1F1	Freon-113	8 → 16 W	2 W	0 RPM	15 min	90 min
A2	STATIC	2V1	Water/NCG	20 → 40 W	20 W	0 RPM	15 min	30 min
A3	STATIC	3F2	Freon-113	8 → 16 W	2 W	0 RPM	15 min	90 min
A4	STATIC	4V2	Water/NCG	20 → 40 W	20 W	0 RPM	15 min	30 min
B1	SPIN	1F1	Freon-113	6 W	2 RPM	2 → 8	10 min	40 min
B2	SPIN	2V1	Water/NCG	40 W	3 RPM	20 → 29	10 min	40 min
B3	SPIN	3F2	Freon-113	6 W	2 RPM	2 → 8	10 min	40 min
B4	SPIN	4V2	Water/NCG	40 W	3 RPM	20 → 29	10 min	40 min
C1	REWICK	1F1	Freon-113	5 W	N/A	50 RPM	20 min	20 min
C2	REWICK	2V1	Water/NCG	20 W	N/A	50 RPM	10 min	10 min
C3	REWICK	3F2	Freon-113	5 W	N/A	50 RPM	20 min	20 min
C4	REWICK	4V2	Water/NCG	20 W	N/A	50 RPM	10 min	10 min
D1	STATIC	1F3	Water	20 → 40 W	20 W	0 RPM	15 min	30 min
D2	STATIC	2V3	Water	20 → 40 W	20 W	0 RPM	15 min	30 min
D3	STATIC	3F4	Water	20 → 40 W	20 W	0 RPM	15 min	30 min
D4	STATIC	4V4	Water	20 → 40 W	20 W	0 RPM	15 min	30 min
E1	SPIN	1F3	Water	40 W	2 RPM	8 → 14	10 min	40 min
E2	SPIN	2V3	Water	40 W	3 RPM	20 → 29	10 min	40 min
E3	SPIN	3F4	Water	40 W	2 RPM	8 → 14	10 min	40 min
E4	SPIN	4V4	Water	40 W	3 RPM	20 → 29	10 min	40 min
F1	REWICK	1F3	Water	20 W	N/A	50 RPM	20 min	20 min
F2	REWICK	2V3	Water	20 W	N/A	50 RPM	10 min	10 min
F3	REWICK	3F4	Water	20 W	N/A	50 RPM	20 min	20 min
F4	REWICK	3V4	Water	20 W	N/A	50 RPM	10 min	10 min

HPP TEST MATRIX (CONT.)

RUN	TEST	HEAT PIPE	FLUID	POWER LEVEL	TEST INCREMENT	SPIN RATE	WARM TIME	RUN TIME
G1	STATIC	1F5	Water	40 W	N/A	0 RPM	15 min	15min
G2	STATIC	2V5	Water	40 W	N/A	0 RPM	15 min	15 min
G3	STATIC	3F6	Water	40 W	N/A	0 RPM	15 min	15 min
G4	STATIC	4V6	Water	40 W	N/A	0 RPM	15 min	15 min
H1	SPIN	1F5	Water	40 W	2 RPM	8 → 14	5 min	20 min
H2	SPIN	2V5	Water	40 W	3 RPM	20 → 29	5 min	20 min
H3	SPIN	3F6	Water	40 W	2 RPM	8 → 14	5 min	20 min
H4	SPIN	4V6	Water	40 W	3 RPM	20 → 29	5 min	20 min
I1	REWICK	1F5	Water	20 W	N/A	50 RPM	20 min	20 min
I2	REWICK	2V5	Water	20 W	N/A	50 RPM	10 min	10 min
I3	REWICK	3F6	Water	20 W	N/A	50 RPM	20 min	20 min
I4	REWICK	4V6	Water	20 W	N/A	50 RPM	10 min	10 min
J1	STATIC	1F7	Water	40 W	N/A	0 RPM	15 min	15 min
J2	STATIC	3F8	Water	40 W	N/A	0 RPM	15 min	15 min
K1	SPIN	1F7	Water	40W	2 RPM	8 → 14	5 min	20 min
K2	SPIN	3F8	Water	40 W	2 RPM	8 → 14	5 min	20 min
L1	REWICK	1F7	Water	20 W	N/A	50 RPM	20 min	20 min
L2	REWICK	3F8	Water	20 W	N/A	50 RPM	10 min	10 min
					TOTAL =			1100

HEAT PIPE PERFORMANCE (HPP)
THERMAL PERFORMANCE EXPERIMENT

HUGHES

- Four Heat Pipes are Mounted at Once in "Pinwheel"; Only One of the Four is Thermally Active at Any Time During the Flight
- Temperature Control is Maintained by Power Regulation Circuitry and by Fans Mounted on Rotating Platform
- Thermostatic Switches Mounted on Heat Pipes Prohibit Overheating
- Thin Film Warmers are Attacheed to the Evaporator Section of Pipes; Maximum Temperature (Inaccessible Surfaces) is 85°C
- Safety Shroud with Aluminum Sides, Lexan 9600 Top Provides Thermal and Collision Protection for Crew
- HPP Operation is Controlled from Raised Panel of "Control Module"
- Data Recorded Using Solid State Data Loggers and Videotape Recording of Control Module Panel Meters
- Orbiter Power Used for Warmers; Power and Data Transferred Through Slip Rings to HPP Rotating Platform
- HPP Attached to Middeck Floor Using Fittings Mounted to Seat Studs

IN-STEP HEAT PIPE PERFORMANCE (HPP) EXPERIMENT

HUGHES

OBJECTIVES – HEAT PIPE TECHNOLOGY

- HOW DOES HEAT PIPE PERFORMANCE IN SPACE DIFFER FROM PERFORMANCE ON THE GROUND?
  - GRAVITY DOMINATES IN GROUND TESTING
  - SURFACE TENSION DOMINATES IN MICRO-GRAVITY ENVIRONMENT
  - EFFECT OF UNDERFILL AND OVERFILL?
- OBTAIN QUANTITATIVE DATA FOR AXIAL GROOVE AND POROUS WICK HEAT PIPES IN A MICRO-GRAVITY ENVIRONMENT FOR:
  - COMPARISON WITH ANALYTICAL MODELS
  - COMPARISON WITH GROUND TEST DATA
  - COMPARISON WITH EXISTING FLIGHT DATA

HEAT PIPE PERFORMANCE (HPP) BACKGROUND

HUGHES

- Primary Objectives of HPP Middeck Experiment:

  1) Obtain Quantitative Data on Thermal Performance of Heat Pipes in a Microgravity Environment for Comparison with Ground Testing

  2) Develop an Increased Understanding of Heat Pipes Subjected to Accelerations in Space

  3) Results will be Used to Improve Design of Spacecraft Thermal Control Systems

- HPP Design Heritage Includes Hughes Fluid Dynamics Experiment (FDE), and Several KC-135 Experiments

Overview of Hughes Space Heat Pipe Microgravity Experiment

HUGHES

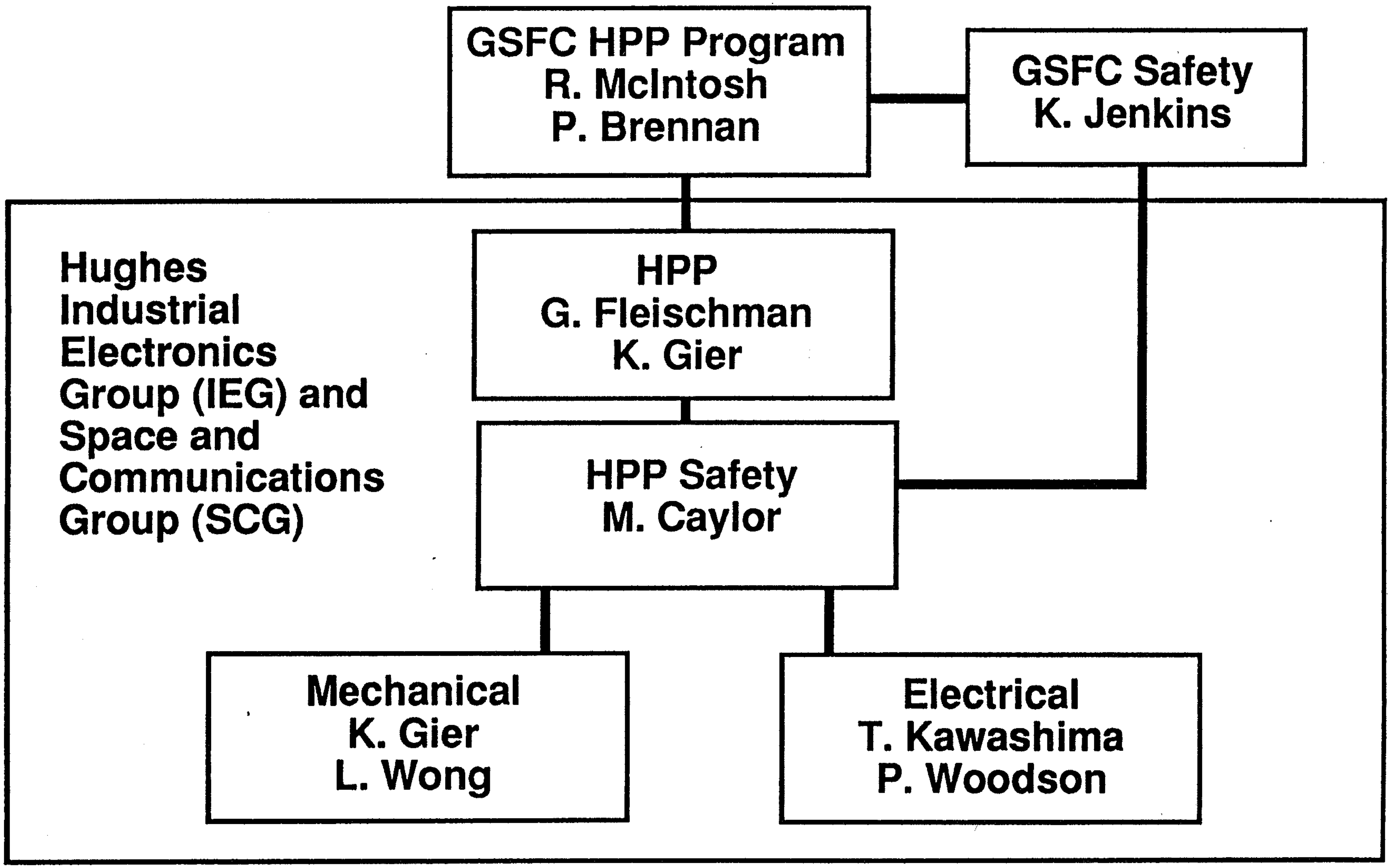
Outline

- Objectives
- Approach
- Heat Pipe Performance (HPP) Experiment Features
- Heat Pipe Design
- Thermal Performance Experiment

N93-28722

HEAT PIPE PERFORMANCE (HPP) BACKGROUND

HUGHES



V0363[redacted]

PMA 1503	Task 9 - Solar	21 July 1992

STRV1-b Flight Experiment

EXISTING		NEW	√	CONTRACT	√	IN-HOUSE	

Description: STRV1-b is a microsat built by the Defense Research Agency UK. PL and SDIO/TNK are co-funding a set of power panels for the satellite in order to fly a set of experimental PV cells.

PROJECT MANAGER: Lt Joel Boswell (505) 846-2637
CONTRACTOR: Various

PROJECT GOALS & OBJECTIVES	BENEFITS	APPLICATION
- Assemble up to 20 next generation PV cells for flight testing - Provide three power panels populated with advanced PV cells - Generate relations with UK government and contractors	- Generate first look at space performance of new R&D PV cells - Flight hardware experience for PL/VTPC - Enhance cooperation with NASA, DRA, etc.	- Data generated will help guide future R&D efforts - Public use of GaAs as main power

p. 7
152228

STRATEGIC DEFENSE INITIATIVE
DEPARTMENT OF DEFENSE

N93-28723

Space Technology Research Vehicle

(STRV-1b)

- Small (100 watt) satellite flown by the British Defense Research Agency

- Mission goal is to fly new and emerging space technologies
  - At a more reasonable cost
  - In a short term timeframe

- Orbit GTO

GTO	Perigee	200 km	
	Apogee	36,000 km	(GEO)
	Period	10.5 hours	

- Mission lifetime: 1 year requirement, 3 year goal

- Various participants include US, UK, and French governments

Space Technology Research Vehicle

SDIO/TNK - PL/VTPC INVOLVEMENT

- Up to 20 advanced experimental cell types to be flown
  - --Thin film cells funded by PL
  - -- Multi-bandgap cells funded by SDIO
  - -- High efficiency cells funded by NASA LeRC, flown as a cooperative venture
  - -- Other cells of interest to SDIO and AF SPO's

- Three of four prime power panels supplied in return (78 watts total)
  - -- Two panels made of thin GaAs/Ge MANTECH cells from ASEC
  - -- One panel and a spare made of GaAs/Ge cells from Spectrolab
  - -- All four panels will be integrated by Spectrolab
  - -- PL/VTPC working directly with DRA on requirements, definitions, etc.

Space Technology Research Vehicle

SDIO/TNK - PL/VTPC INVOLVEMENT (cont.)

- Schedule
  - -- Flight Panel design complete — Nov 92
  - -- Engineering Panel fabrication complete — Nov 92
  - -- Experimental Cells delivered to PL — Jan 93
  - -- Flight Panel fabrication complete — May 93
  - -- Delivery of Flight Panels to DRA (UK) — Sep 93
  - -- Launch — CY94

- Associated ground testing
  - -- Up to 30 of each cell type will undergo proton and electron radiation testing in the UK
  - -- Part of a larger joint US-ESA equipment and procedure comparison

Space Technology Research Vehicle

STRV-1b EXP. CELL LIST

CELL TYPE	SOURCE	CELL TYPE	SOURCE
Si (standard)	Spectrolab	GaInP (top cell)	Spire
GaAs (standard)	ASEC	GaInP (top cell)	Spectrolab
CdTe	Martin Marietta	AlGaAs (top cell)	RTI/ASEC
CIS	Martin Marietta	AlGaAs + CIS	Boeing/Kopin
CIS	Boeing	GaAs/Ge	Spectrolab
MANTECH GaAs/Ge	ASEC	a-Si	TRW
ITO/InP	NREL	GaAs	EML/TST
InP	Spire		

- SDIO

- Air Force

- NASA LeRC

Space Technology Research Vehicle

Programmatics

- **Budget**

	SDIO/TNK	PL/VTPC
JON	150309ST	682JTDAJ
$K	151	50

Total experiment cost: $201K

- **Schedule**
  - -- Launch will be early 94
  - -- Our delivery date is Sep 93

- **Status**
  - -- On schedule
  - -- Sufficient funding

NEP Space Test Program Objective

The Objective Of The NEP Space Test Program Is To Launch A NEP Satellite Powered By A Russian Topaz II Reactor By December 1995

23

N93- 28724

Space Technology Research Vehicle

Anticipated Benefits

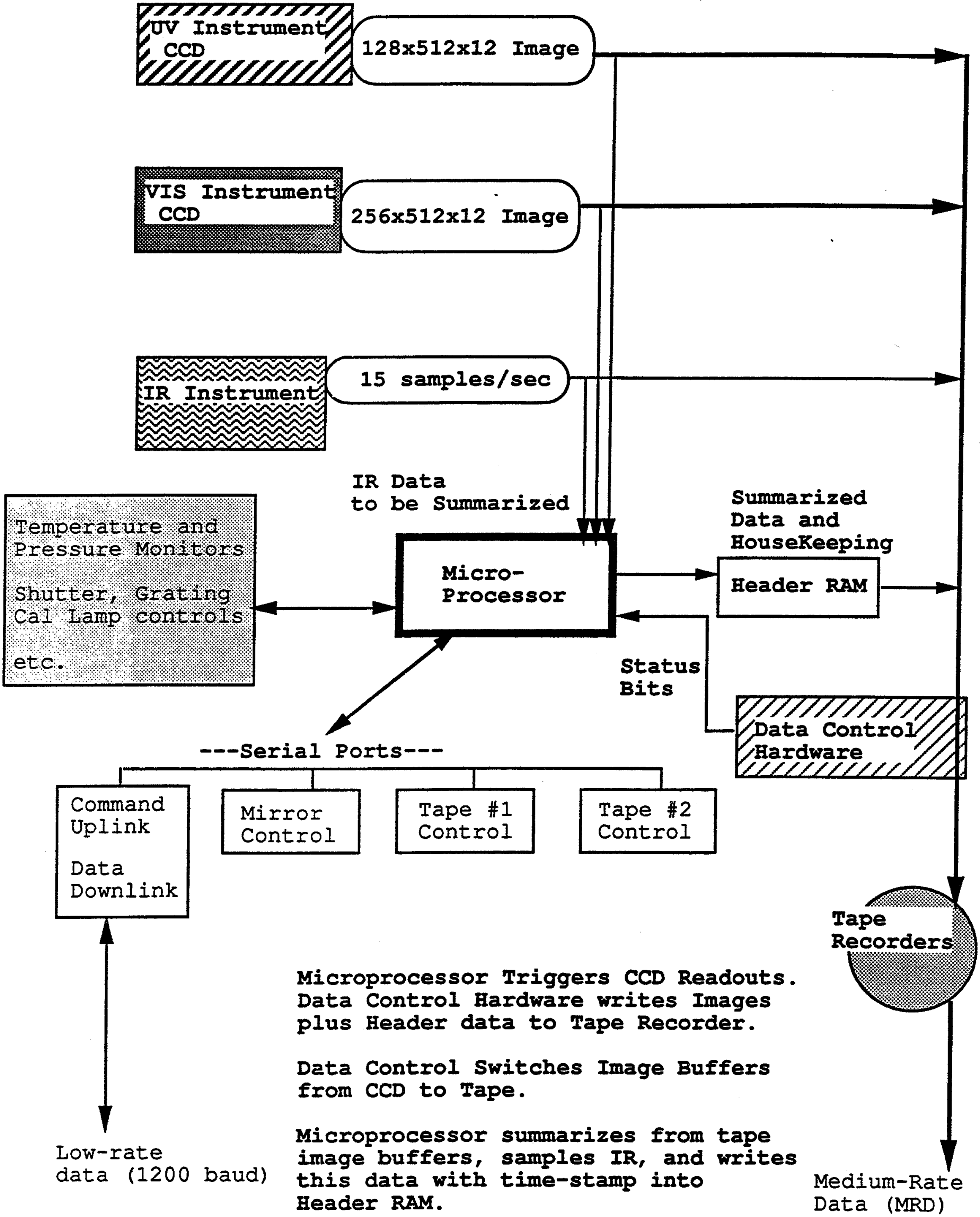
- Experiments
  - -- Flight testing of recent advances in the PV area
  - -- Helpful "industry survey" for PL/VTPC
  - -- Increase cooperation with NASA LeRC and DRA

- Power Panels
  - -- Useful "space hardware" experience
    - --- Procurement
    - --- Testing & flight qualification
  - -- Excellent test of MANTECH GaAs solar cell technology
    - --- Demonstration of the MANTECH cells in a hostile environment
    - --- Laydown by an outside contractor

- Ground testing
  - -- Confirmation of space data and other ground testing
  - -- Participation in the joint US-ESA calibration effort

C-6

EISG Data Flow



Microprocessor Triggers CCD Readouts. Data Control Hardware writes Images plus Header data to Tape Recorder.

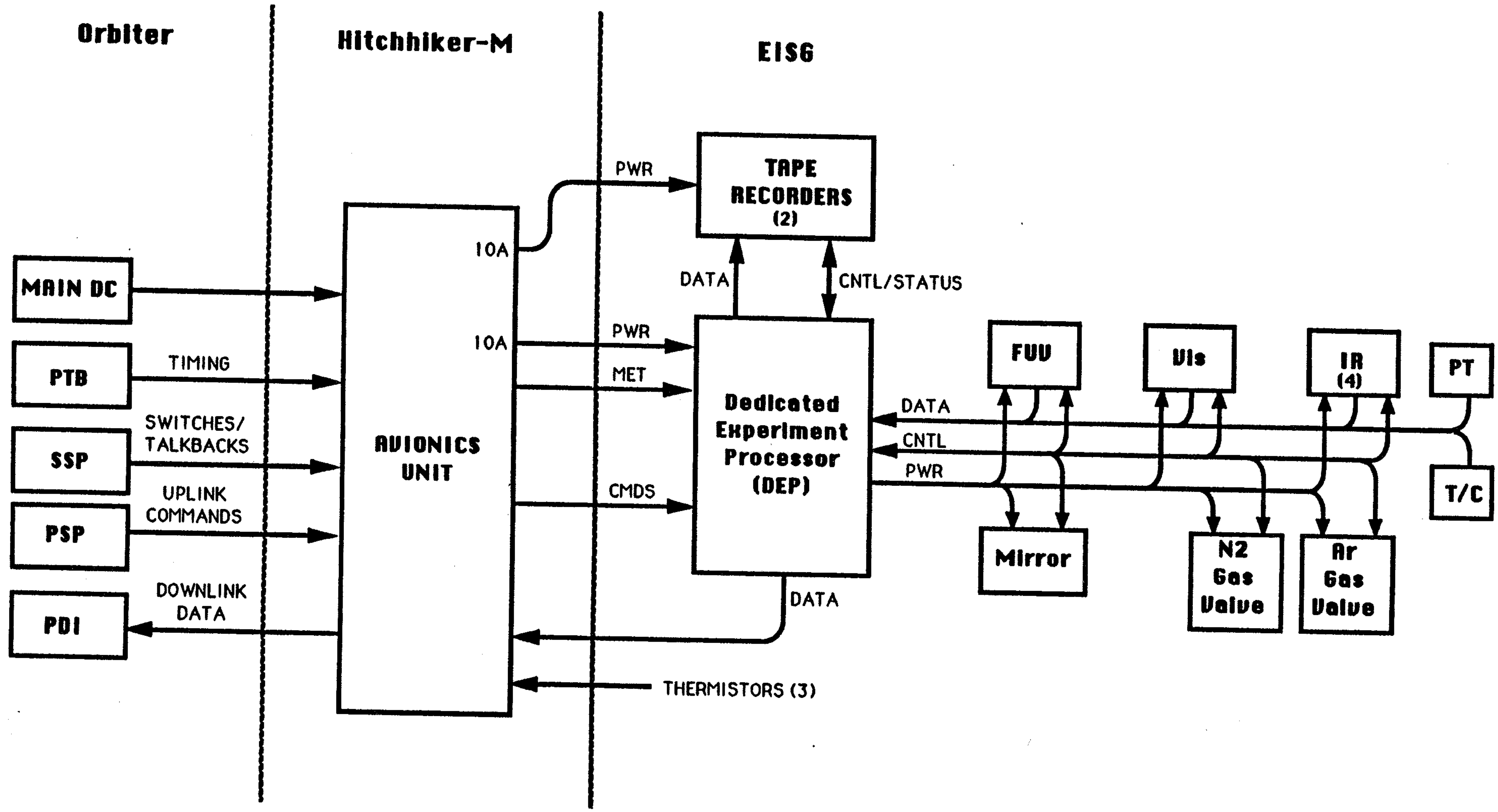
Data Control Switches Image Buffers from CCD to Tape.

Microprocessor summarizes from tape image buffers, samples IR, and writes this data with time-stamp into Header RAM.

[redacted]

Lockheed	Experimental Investigation of Spacecraft Glow	Preliminary Design Review
		Alan Epperson — 3-5-91

Avionics Block Diagram



IMPORTANCE OF COMBINING SKIRT WITH EISG (con't)

- SKIRT will allow EISG to measure other contributors to the infrared glow: OH, NO^+, NO_2, ...
- SKIRT will obtain data during both day and night sides of orbits, thereby enhancing EISG night-only data.
- Comparing SKIRT data from STS-39 and STS-62 will show the dependence of the glow on phase in the solar activity cycle.
- EISG/SKIRT will expand and enhance the technology base pertaining to shuttle glow.

IMPORTANCE OF COMBINING SKIRT WITH EISG

- Only by including SKIRT as part of EISG will it be possible to investigate all aspects of the glow process. SKIRT gives IR spectroscopic coverage to EISG.

- The elliptical orbits planned for STS-62 will provide a unique opportunity to measure the IR, visible, and UV glow as a function of altitude. Altitude information was not obtained by SKIRT on STS-39.

- EISG nitrogen gas releases will provide the first combined IR, visible and UV test of the role nitrogen plays in the glow chemistry.

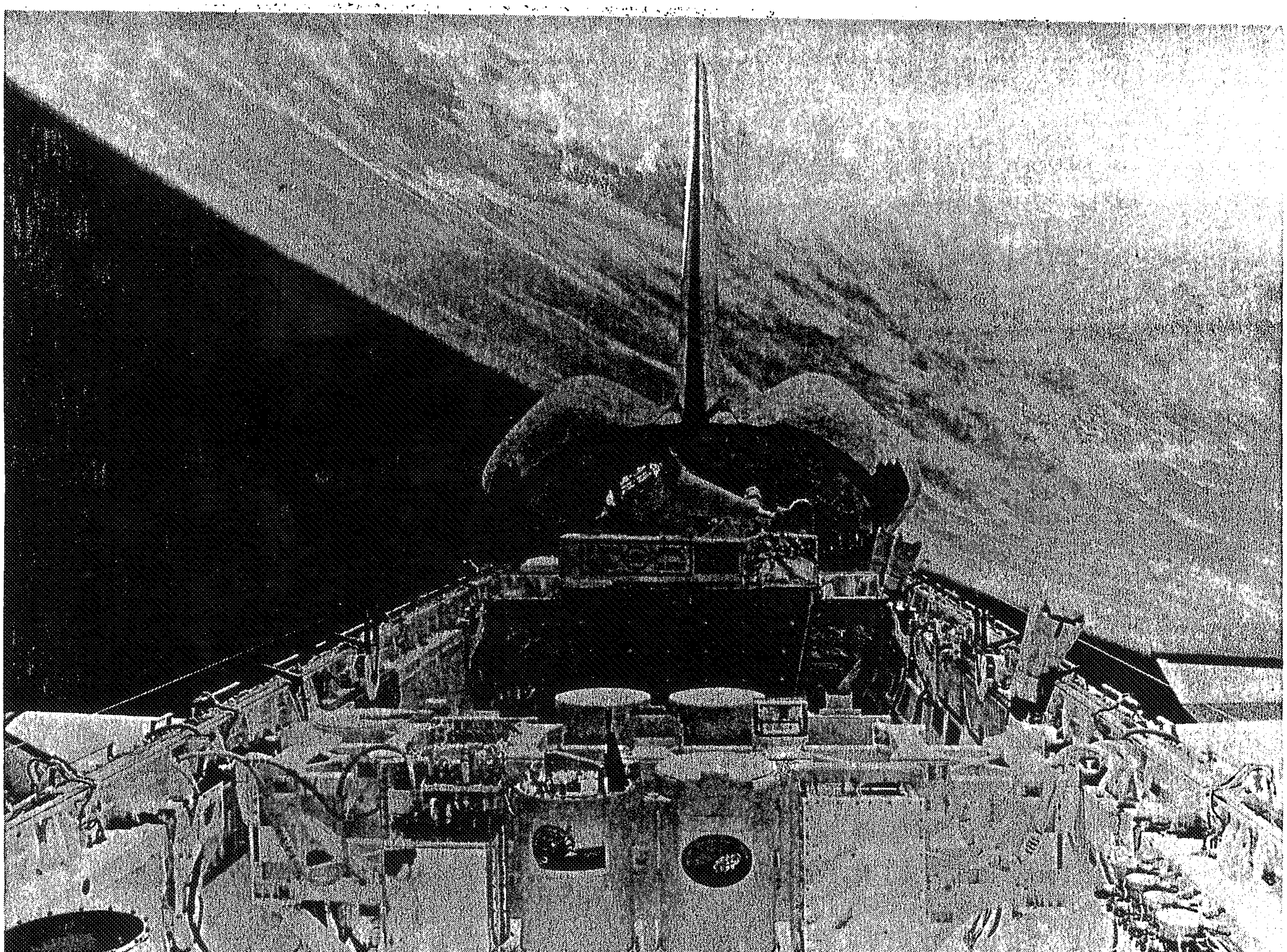
THE ROLE OF INFRARED SPECTRA IN INVESTIGATING THE GLOW PROCESS

- Spacecraft-atmospheric interaction involves three reactions:
    1) $N_2 + O \rightarrow NO + N$
    2) $NO_{surface} + O \rightarrow NO_2^*$
    3) $N_{surface} + N \rightarrow N_2^*$

- Reaction 1) is monitored by observing NO spectral emission in the infrared. (This was first demonstrated by SKIRT on STS-39.)

- Reactions 2) and 3) are monitored by observing NO_2^* and N_2^* spectral emissions in the visible and ultraviolet, respectively.

- Only by observing spectra in all three regimes - IR, visible, and UV - can the entire glow process be studied.



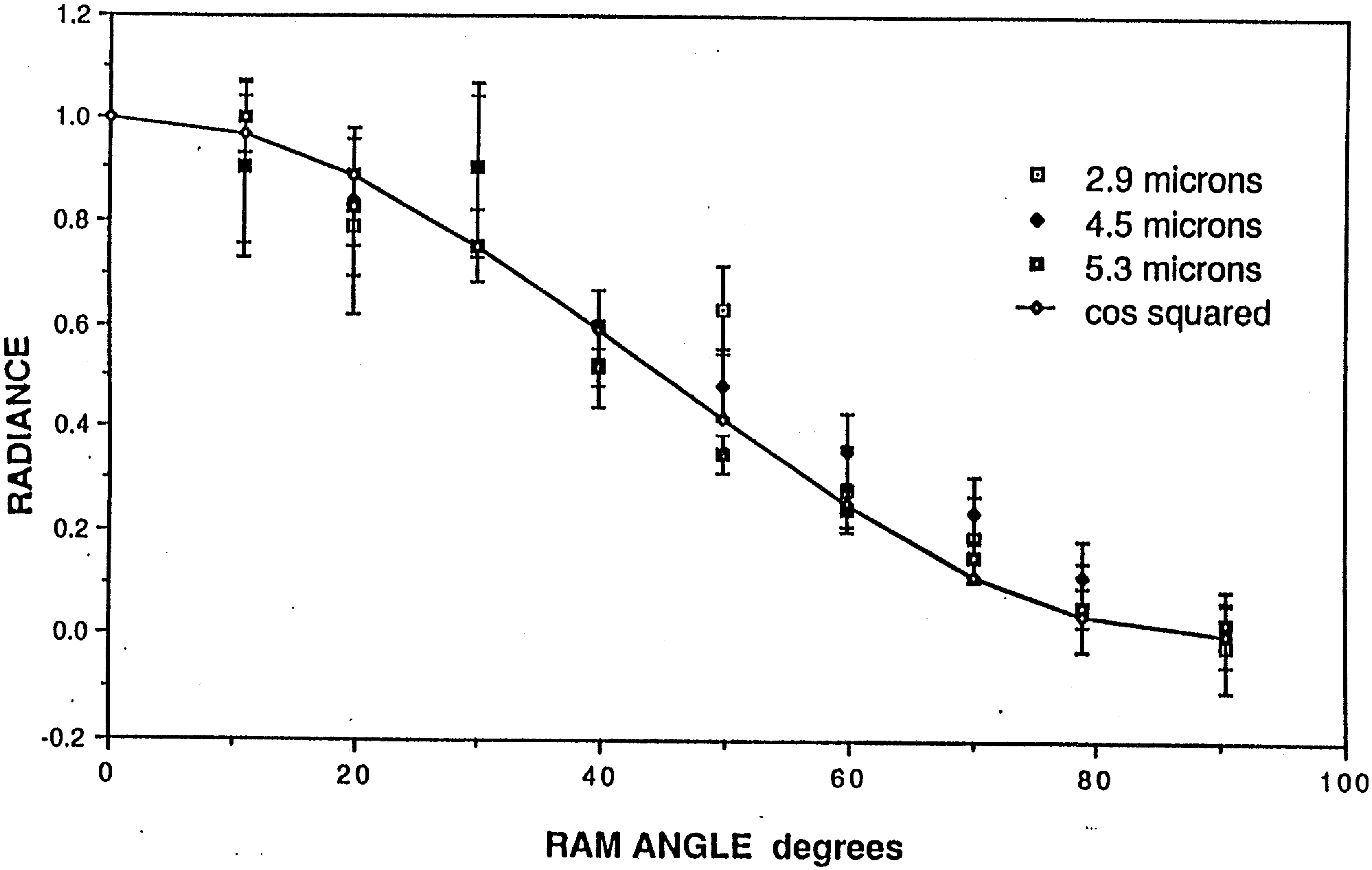
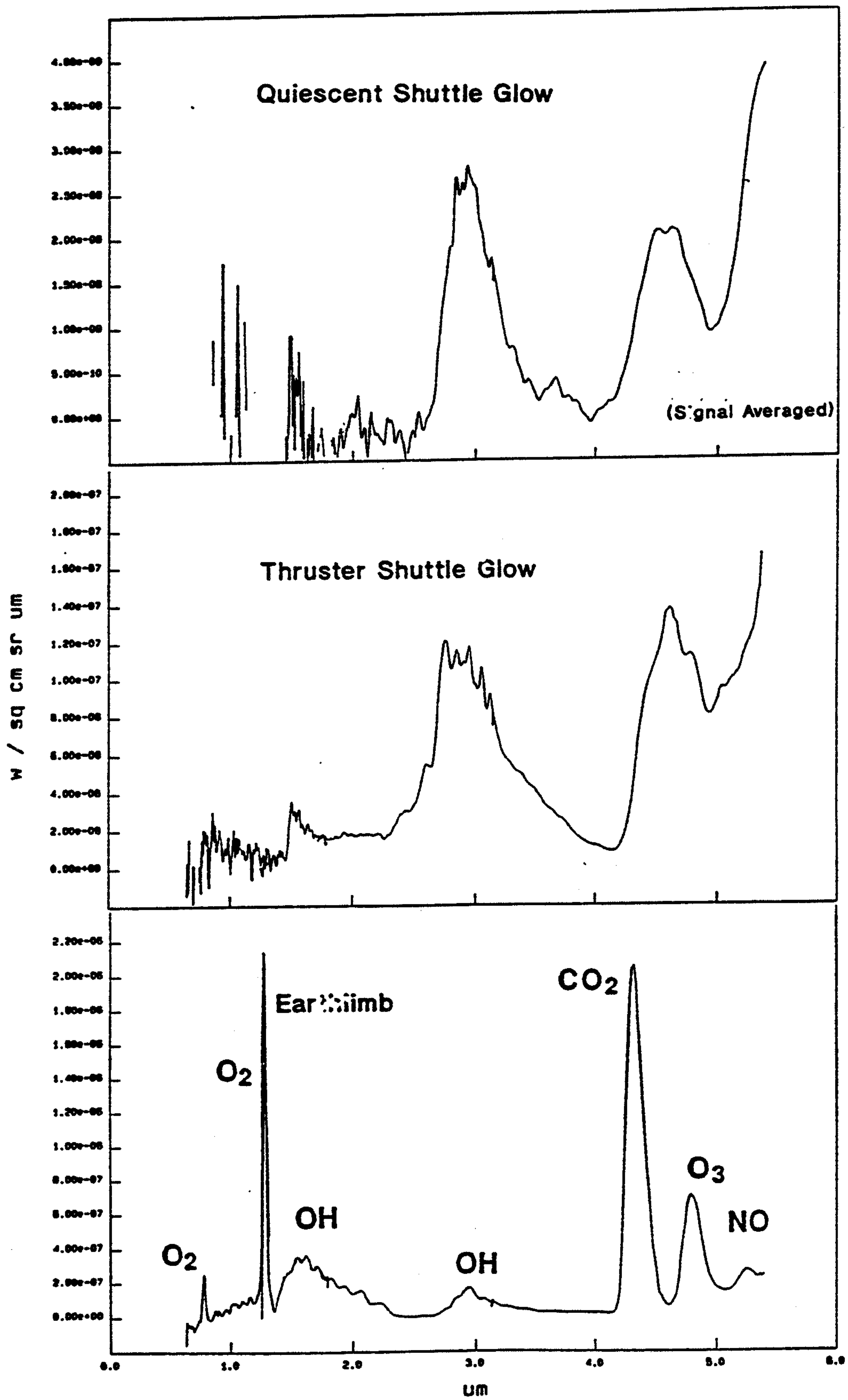
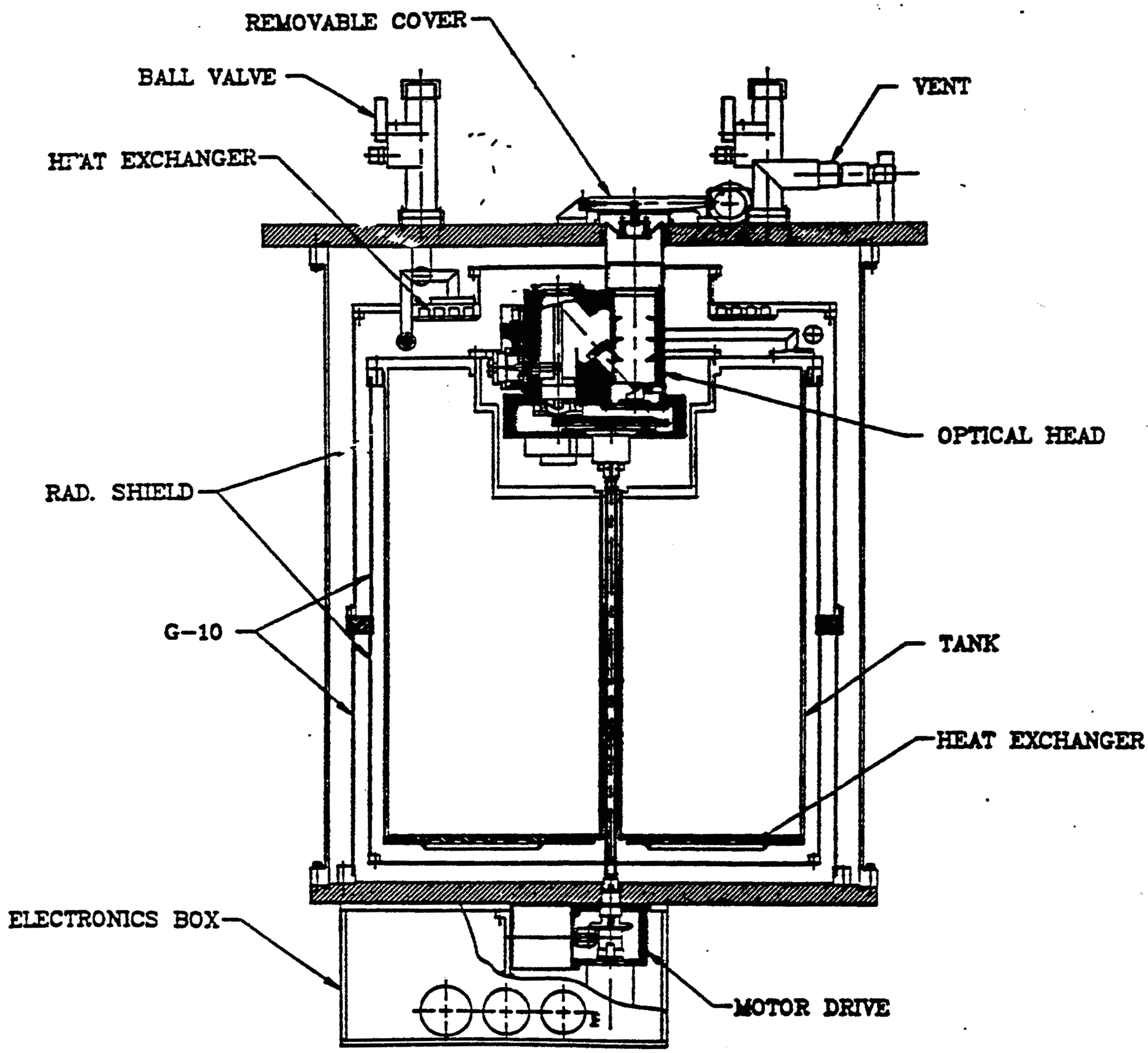


Figure 36. 2.9 μm, 4.5μm, and 5.5 μm Glow Intensities vs Ram Angle.[25]


Quiescent Shuttle Glow
(Signal Averaged)
Thruster Shuttle Glow
Earthlimb
O_2
OH
CO_2
O_3
NO
w / sq cm sr um
um



SKIRT CVF payload configuration. This assembly fits into a NASA Get Away Special (GAS) can modified for a Hitchhiker mechanical and electrical interface.

Instruments - Optical Properties

VISIBLE IMAGING SPECTROMETER (VIS)

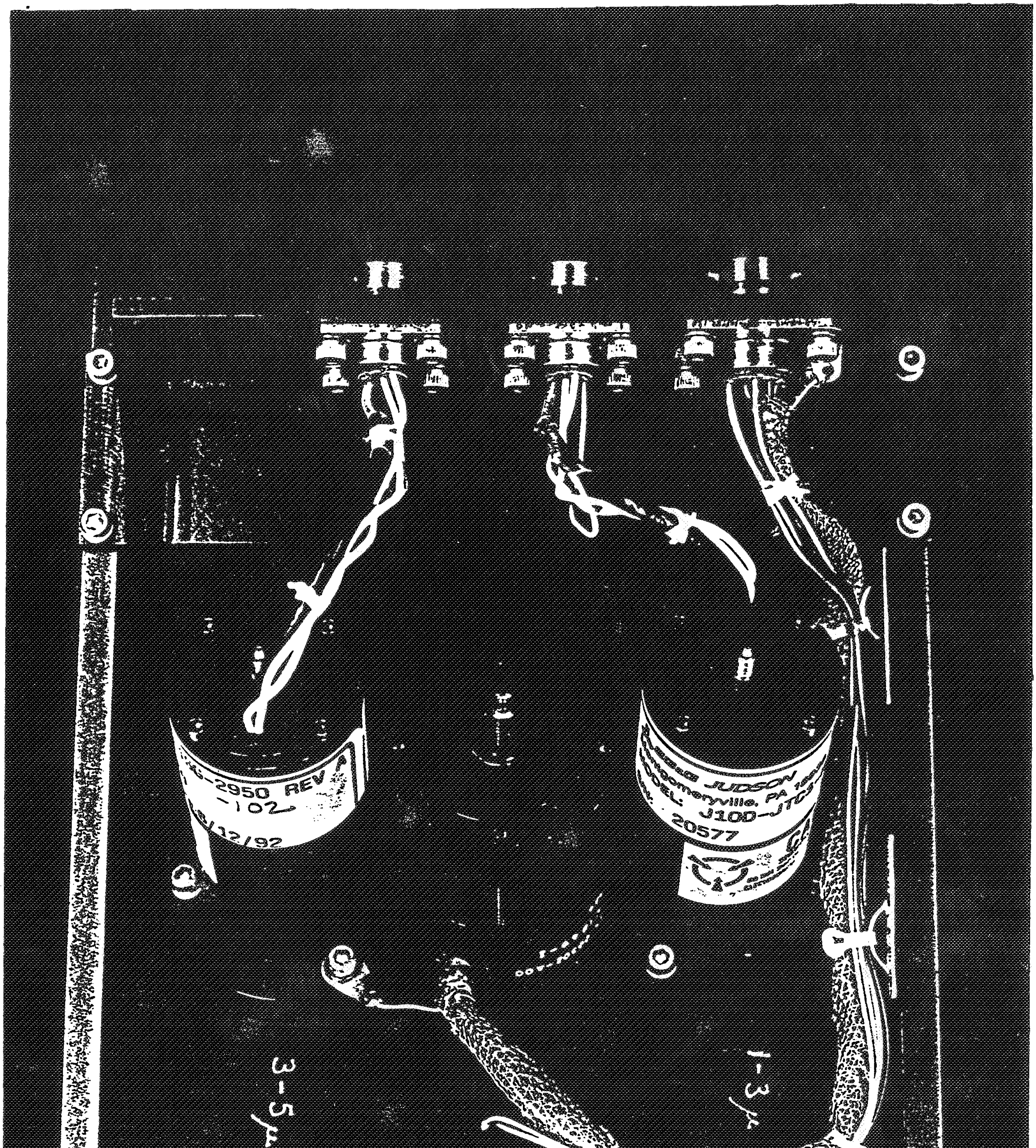
Optics:	Field of view	60°
	Transmission Grating	600 lp/mm
	Throughput	f/1.8
	Spectral range	4500-8400Å
	Spectral Resolution	15Å
Image plane:	X axis - spatial	60° FOV
	Y axis - spectral	4500-8400Å
Detector:	ICCD camera with S20 R photocathode intensifier	
	Integration time	1/60-180 s

FAR ULTRAVIOLET IMAGING SPECTROMETER (FUV)

Optics	Field of View	20°
	Transmission Grating	1800 lp/mm
	Throughput	f/4.8
	Spectral range	1100-2000 and 1900 3000Å [2 step gratin; cam]
	Spectral resolution	15Å
Image plane:	X axis - spatial	20°
	Y axis - spectral	15Å
Detector:	ICCD camera with RbTe photocathode intensifier	
	Integration time	1/60-180 s

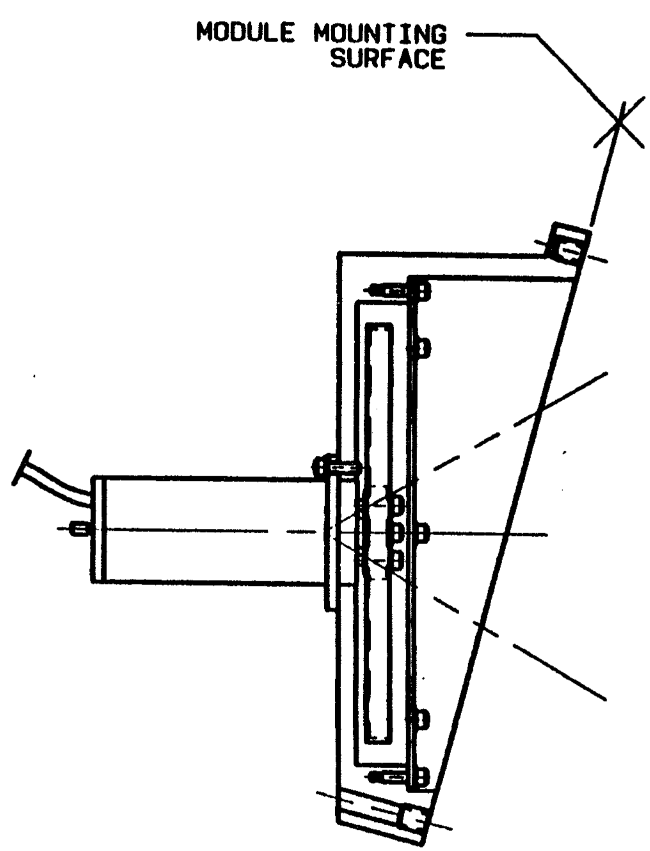
INFRA RED DETECTORS (2 each Sample)

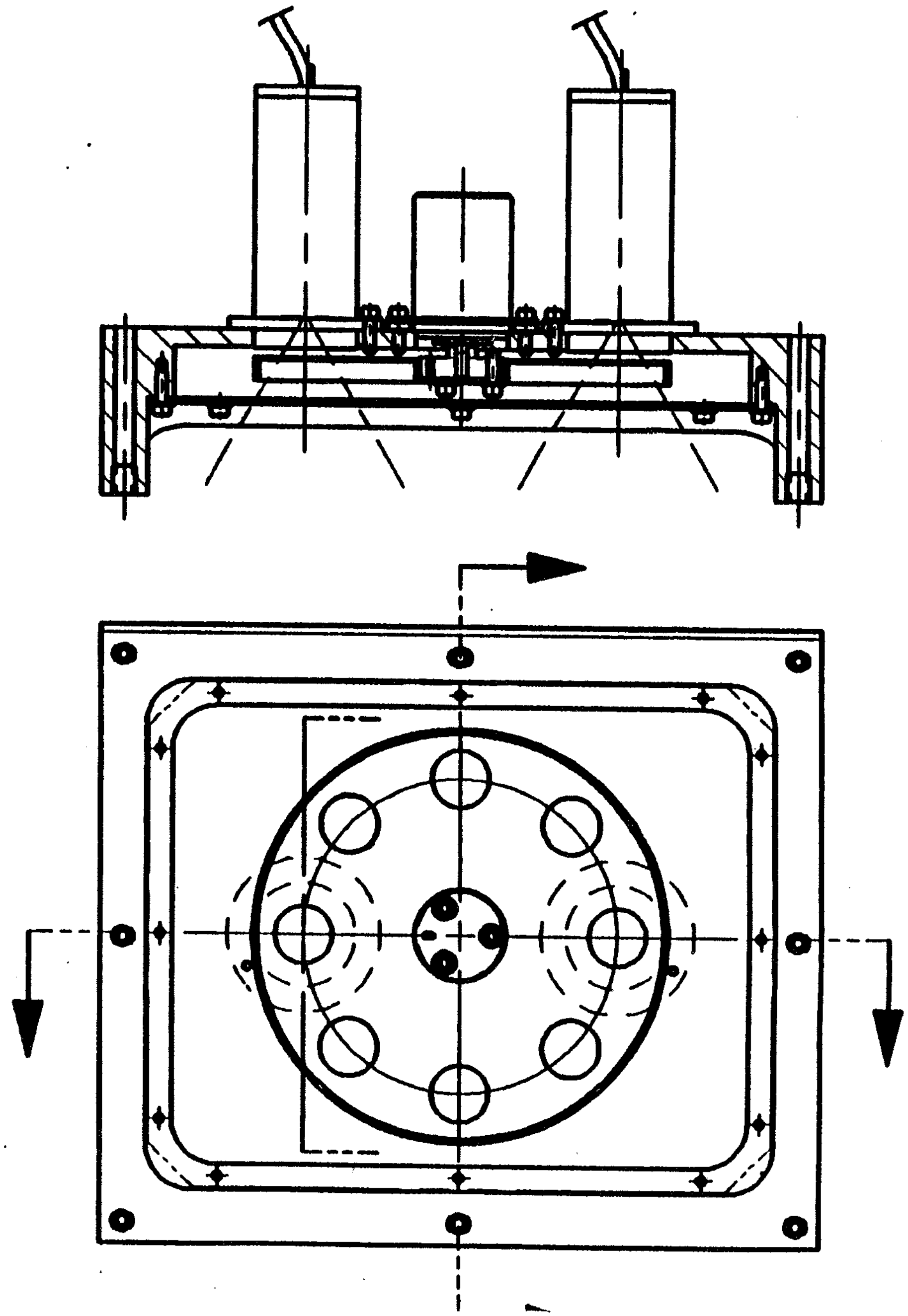
Optics:	Field of view [Cold shield sets field stop, without lenses]	60°
	Filters	1-3 micron (#1) 3-5.4 microns (#2)
Detectors:	InSb, Joule-Thomson cooled to 84 K with Argon cryostat. Supply bottle at 3000 PSI, flows at .5 liter per minute, STP, while active.	
	Sample rate is 15 Hz for each channel	


2950 REV A
-102
JUDSON
20577
3-5 μ
1-3 μ

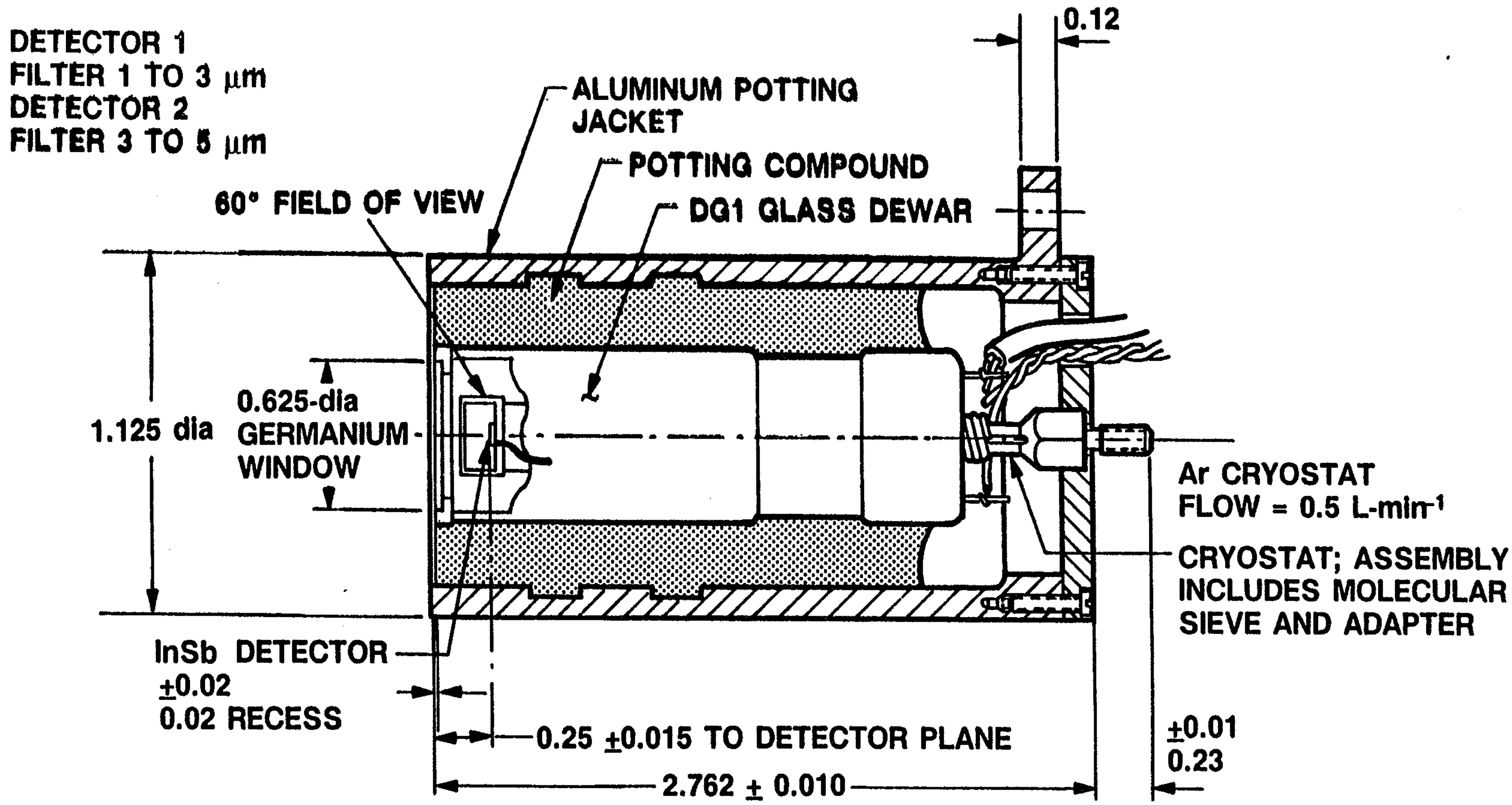
Lockheed	Experimental Investigation of Spacecraft Glow	Preliminary Design Review
		Jeff Ruddock — 3-5-91

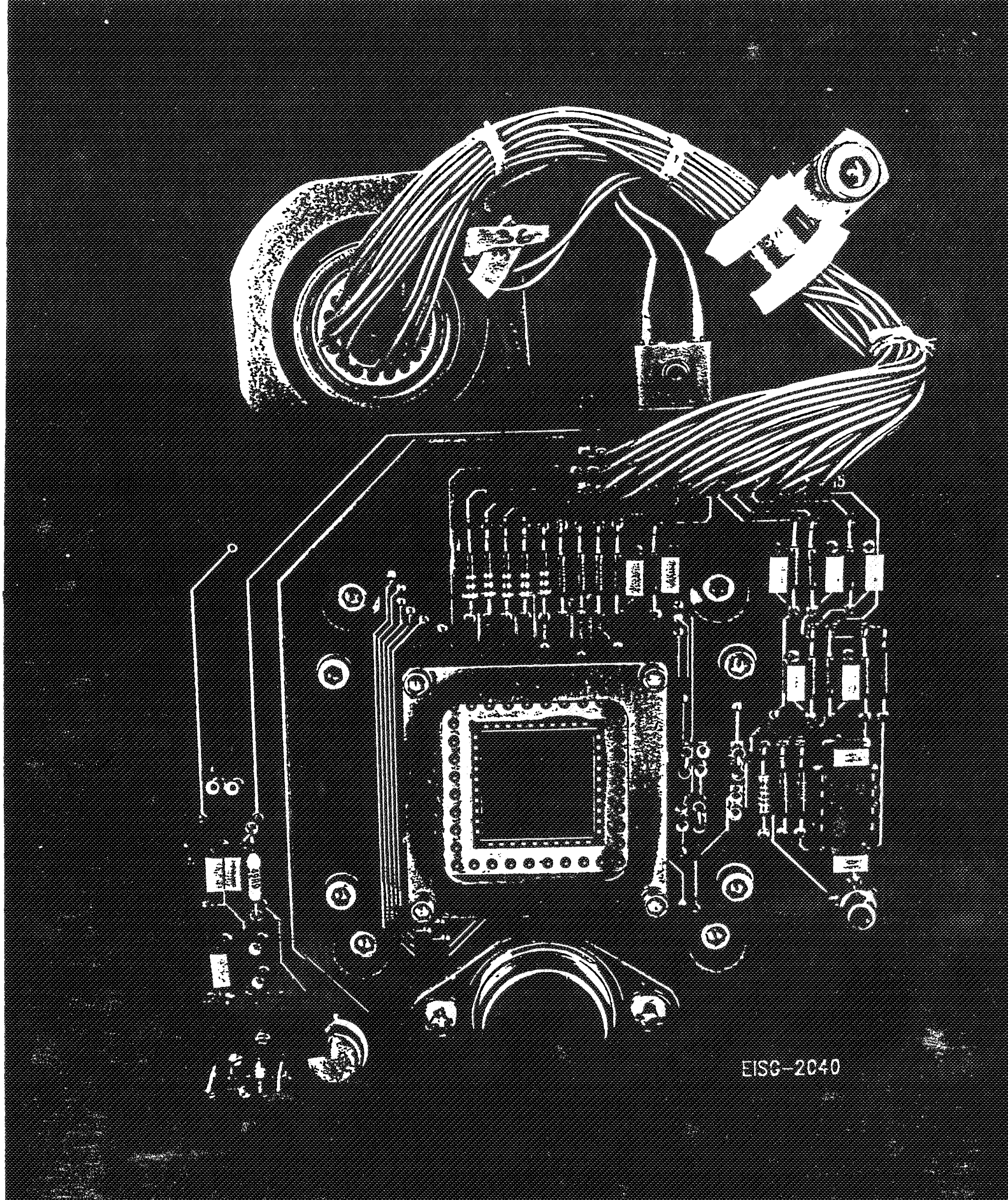
EISG IR / Chopper Drawing

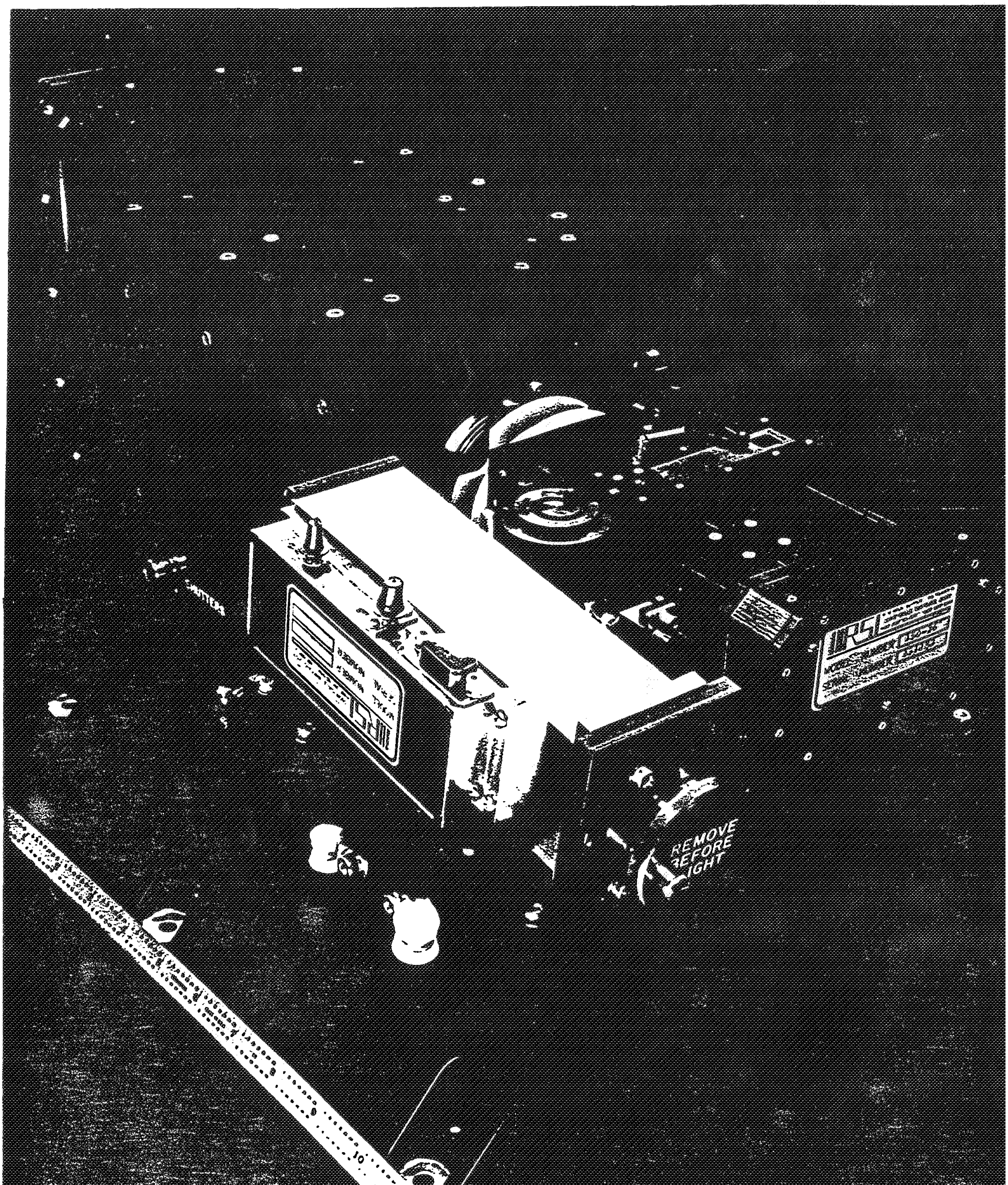




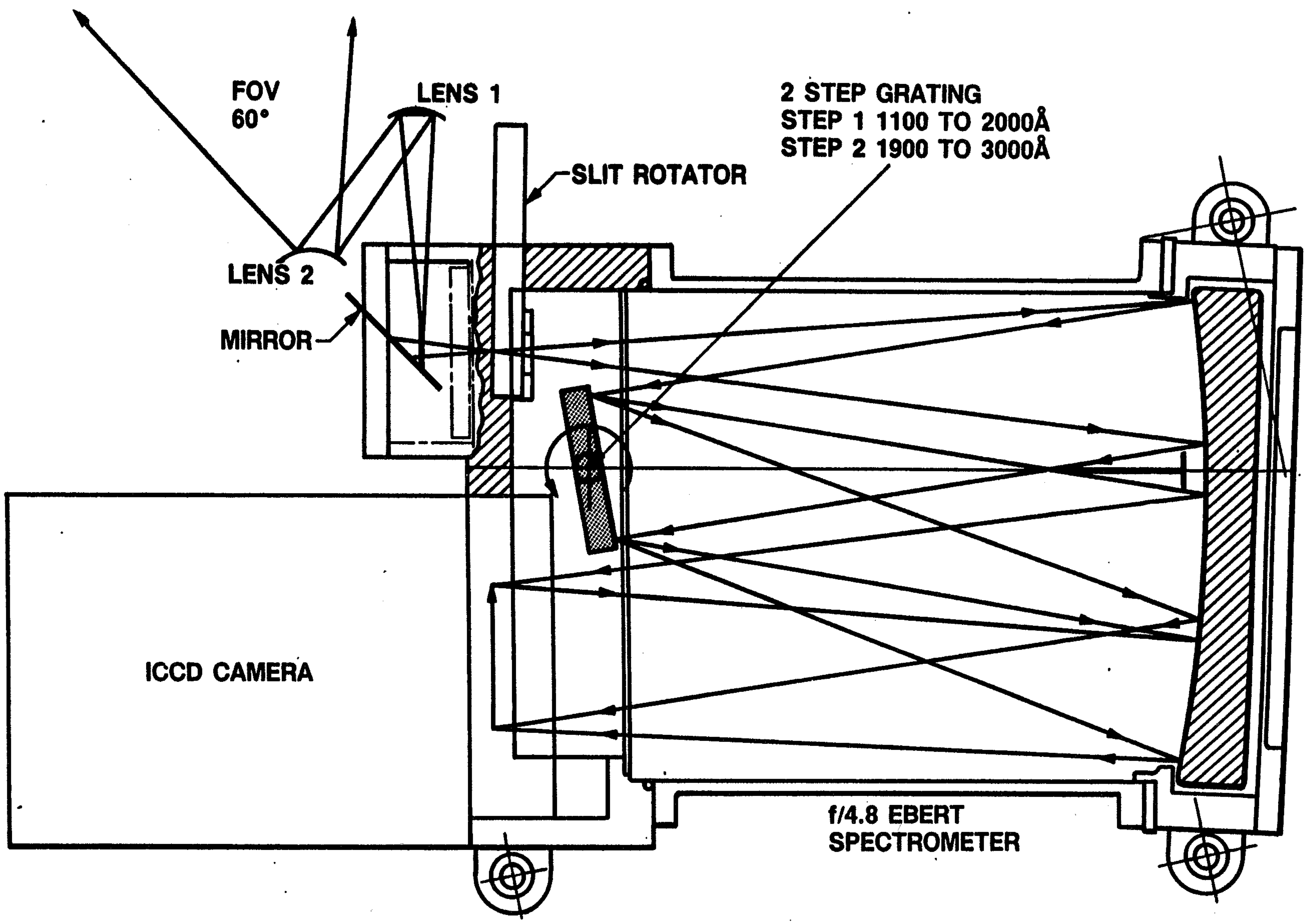
IR DETECTOR (2)

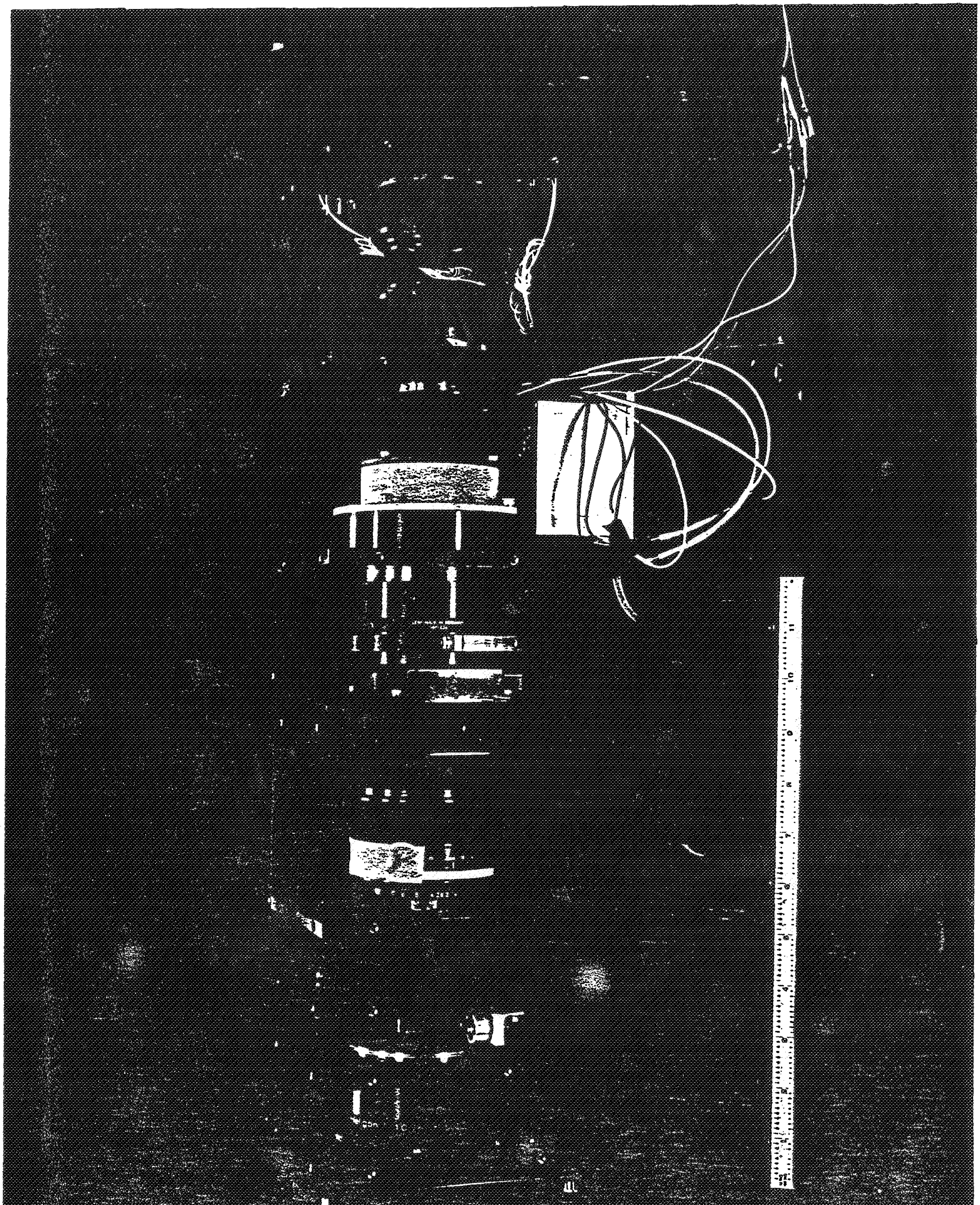


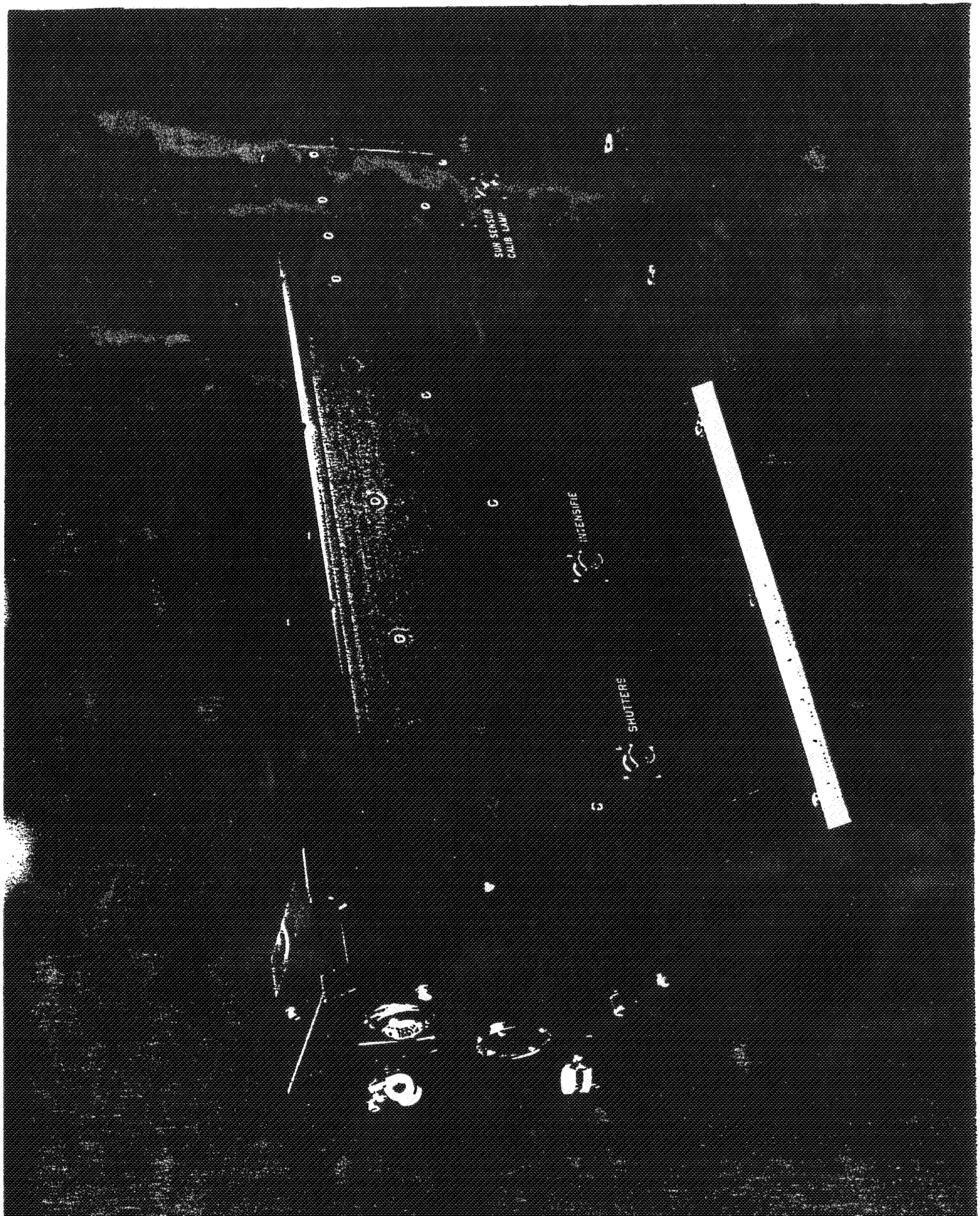

EISG-2040



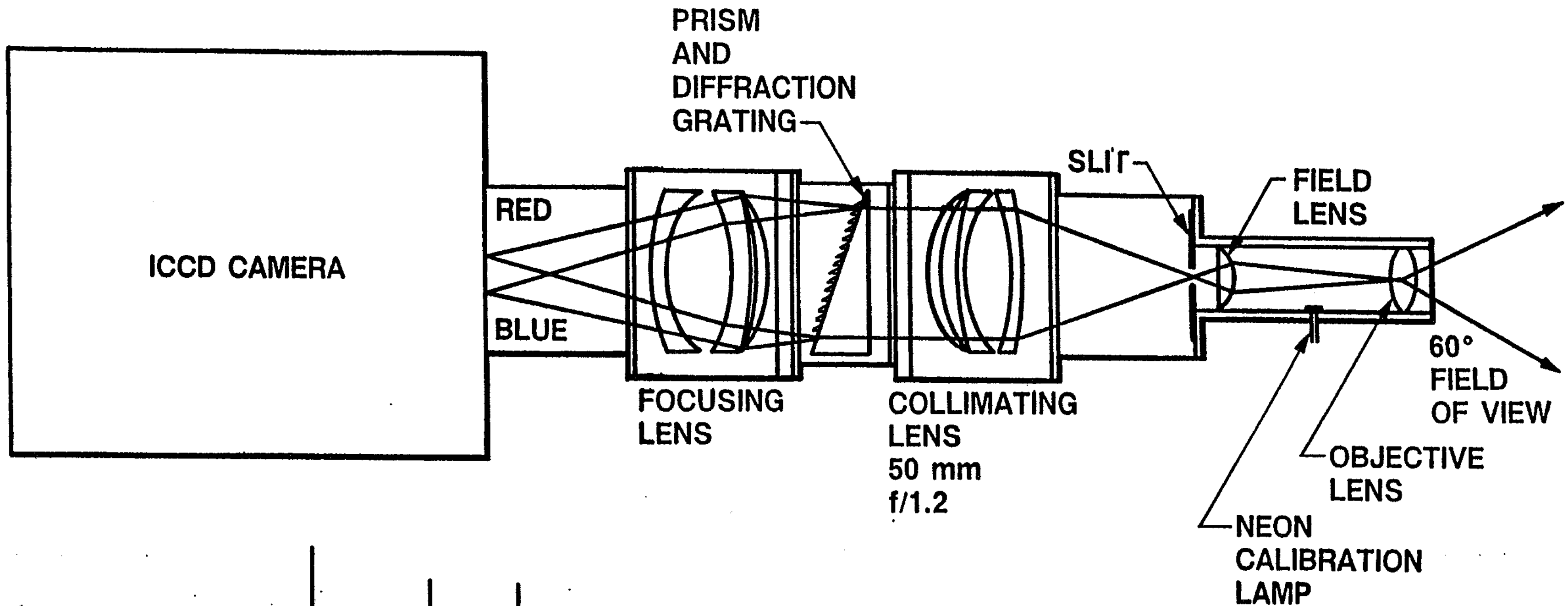
FUV IMAGING SPECTROMETER

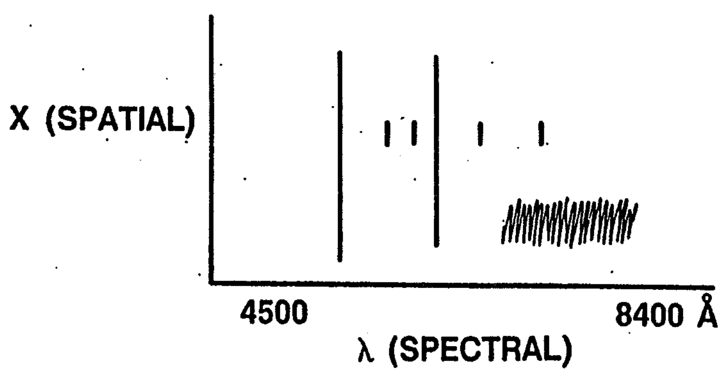




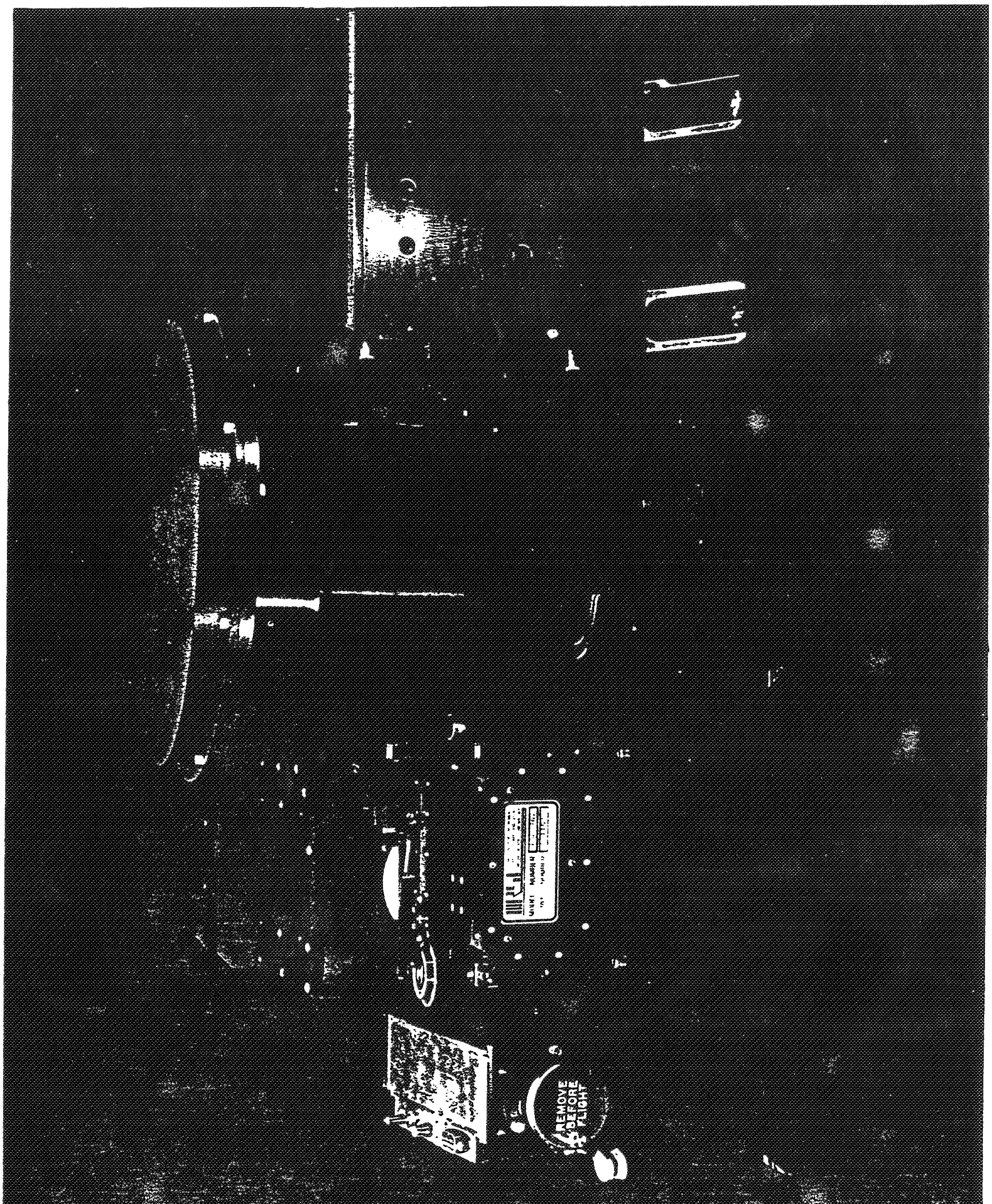

SUN SENSOR
CALIB LAMP
INTENSIFIE
SHUTTERS

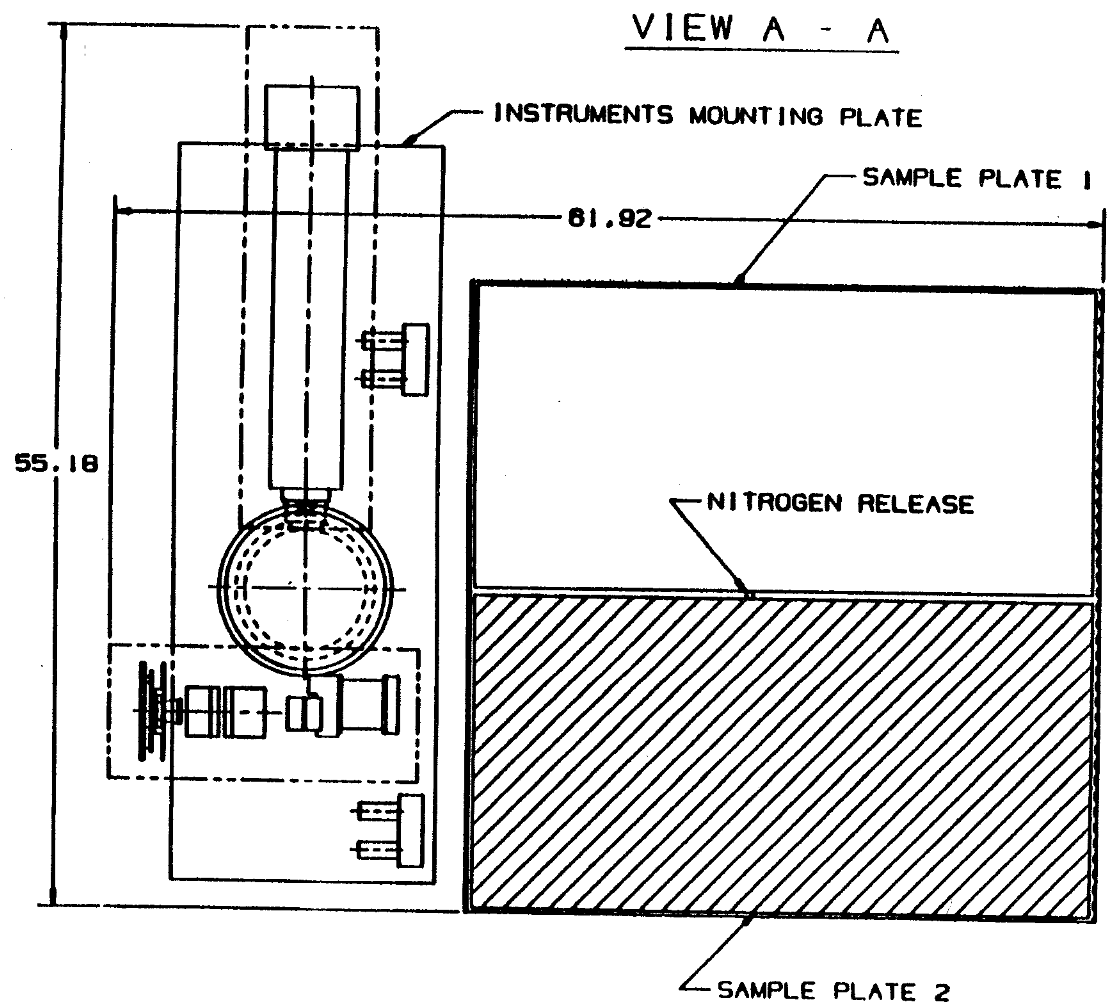
VISIBLE IMAGING SPECTROMETER

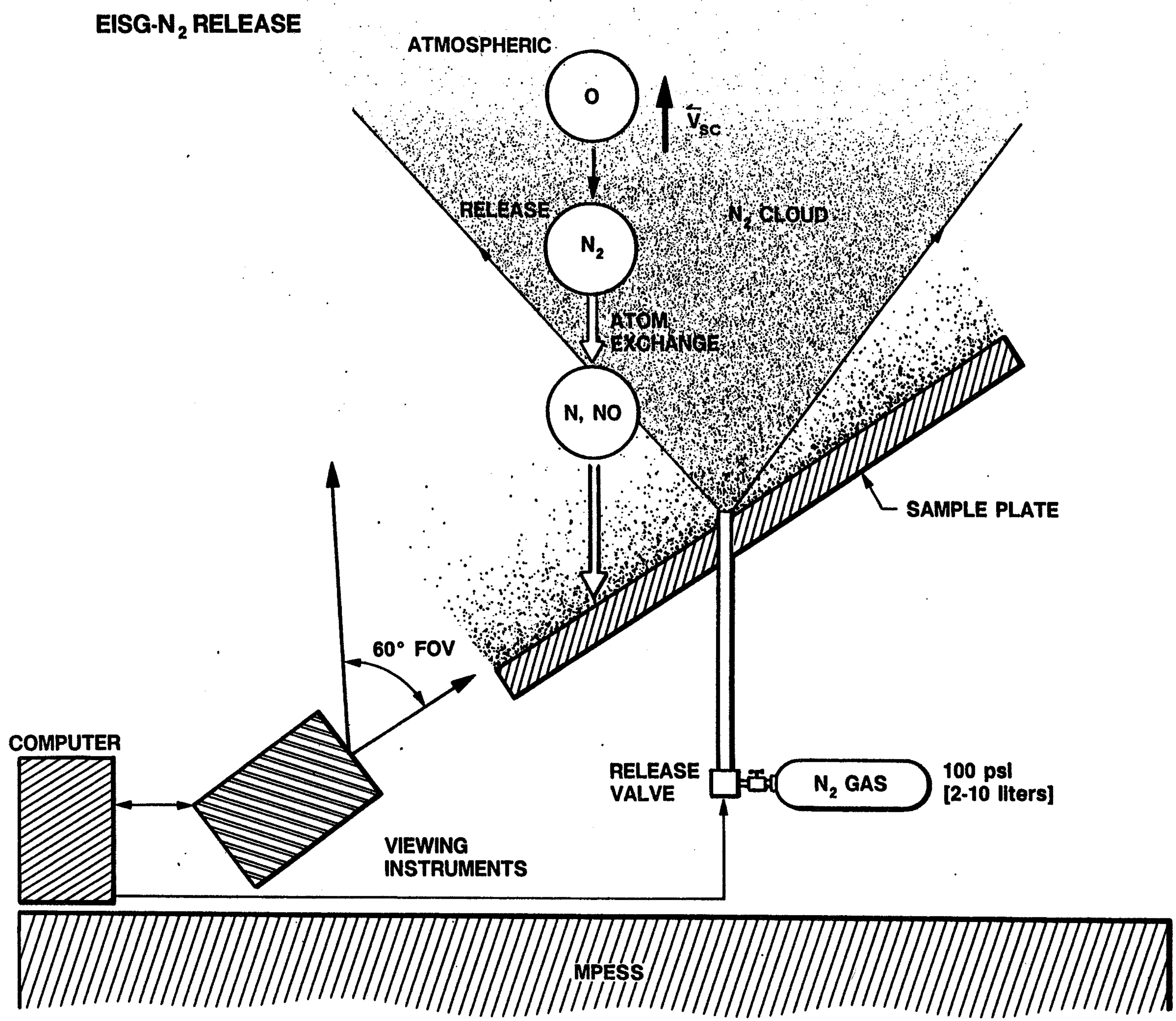




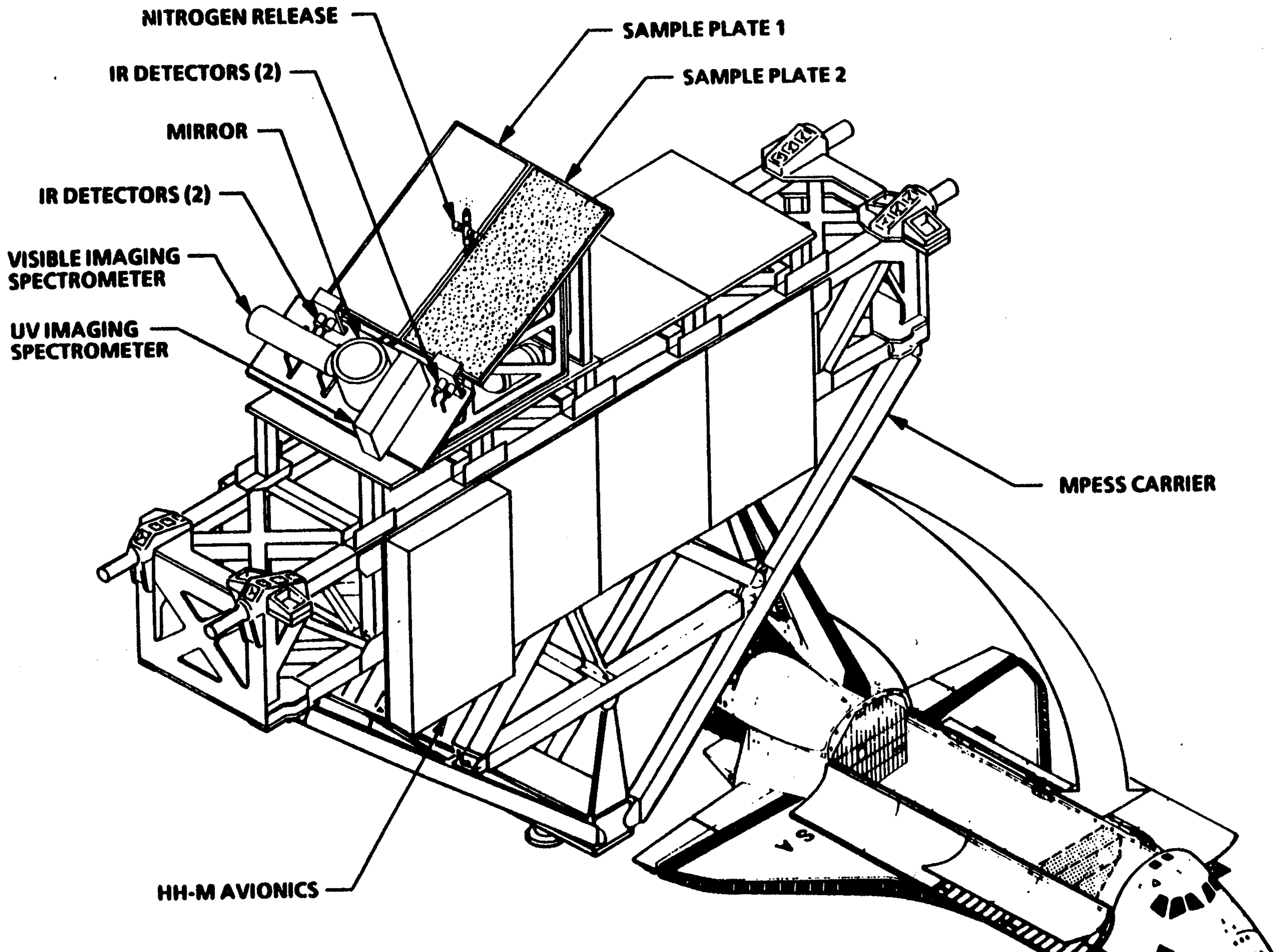
INTEGRATION
1/60 TO 180 s


REMOVE BEFORE FLIGHT

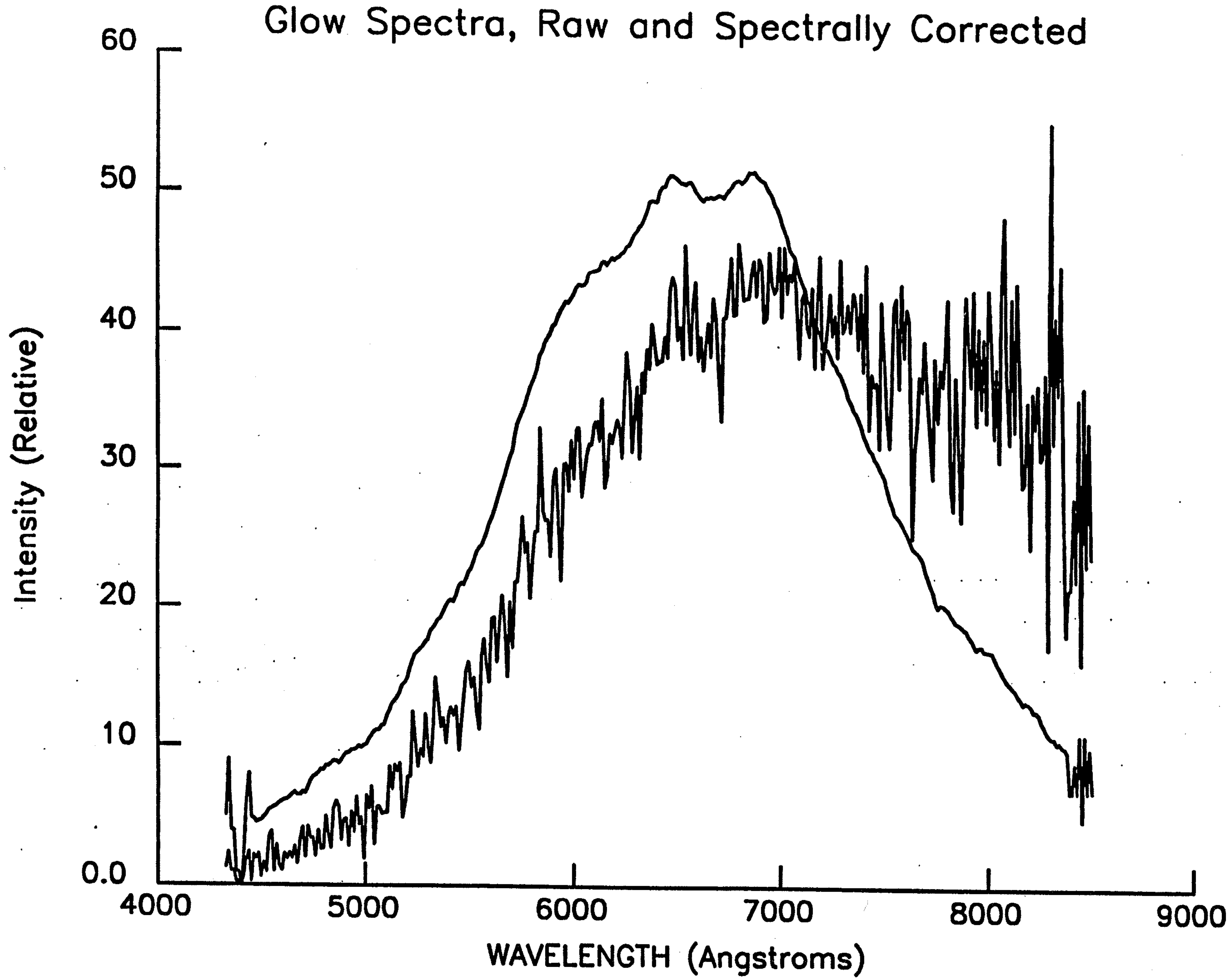

VIEW A - A
INSTRUMENTS MOUNTING PLATE
SAMPLE PLATE 1
81.92
55.18
NITROGEN RELEASE
SAMPLE PLATE 2

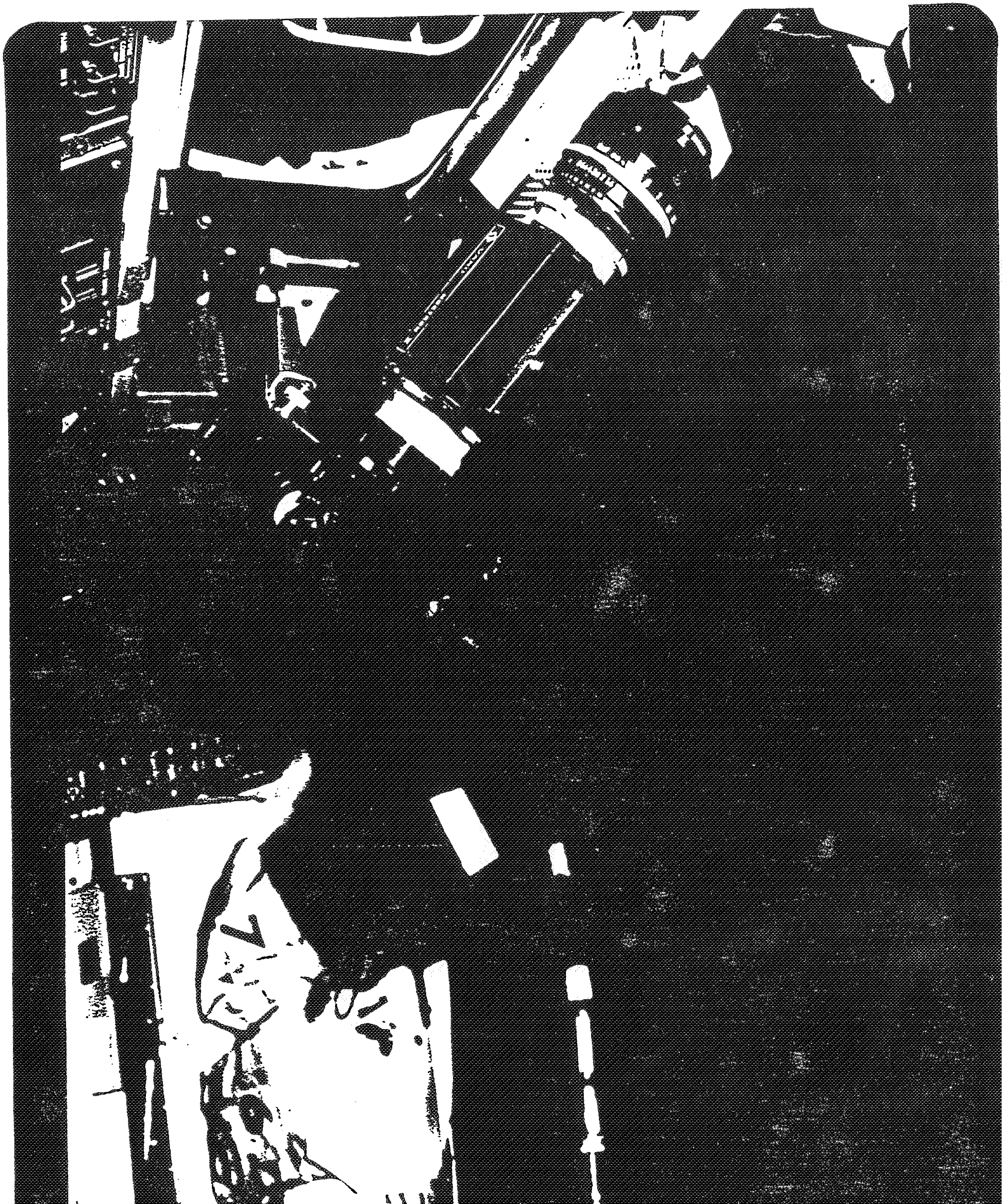

EISG-N_2 RELEASE
ATMOSPHERIC
O
$\vec{V}_{SC}$
RELEASE
N_2
N_2 CLOUD
ATOM
EXCHANGE
N, NO
SAMPLE PLATE
60° FOV
COMPUTER
RELEASE
VALVE
N_2 GAS
100 psi
[2-10 liters]
VIEWING
INSTRUMENTS
MPESS

Lockheed	Experimental Investigation of Spacecraft Glow	Preliminary Design Review	
		Sam Choudhary	3-5-91



NOTE: 2 TAPE RECORDERS AND ELECTRONICS BOX ARE MOUNTED ON SIDE PLATE IN AFT POSITION.


Glow Spectra, Raw and Spectrally Corrected
Intensity (Relative)
60
50
40
30
20
10
0.0
4000
5000
6000
7000
8000
9000
WAVELENGTH (Angstroms)



RECENT GLOW DEVELOPMENTS

(GRS, 4/23/92)

1. FLIGHT DATA-STS 39!

   -NO RELEASES CONFIRM GLOW BRIGHTNESS RELATED TO NO SURFACE DOPING.

   -NEW AND INTERESTING IR DATA (INCLUDING SKIRT EXPERIMENT).

2. LABORATORY EXPERIMENTS!

   -CONFIRM N_2 + O ATOM EXCHANGE CROSS SECTION IS LARGE!

3. STUDIES!

   -DAYTIME 1-3 MICRON INTENSITIES FROM SPACELAB 2, IRT EXPERIMENT SUGGEST DAYTIME ANAMOLIES.

Experiment Objective

Develop understanding of the physical processes leading to spacecraft glow phenomena, with emphasis on surface temperature and altitude effects. This development can be used to:

- Characterize optical instrument backgrounds
- Provide guidelines for thermal insulations
- Characterize material selection for flight optics and associated spacecraft
- Affect flight-operation altitude selection for relevant missions

CURRENT (MAJOR) GLOW MYSTERIES

SPECTRAL REGION	MYSTERY
VISIBLE (4000-8000 Å)	-WHAT IS THE SOURCE OF NO? ATMOSPHERIC? ATOM EXCHANGE? I.E. $O + N_2 \rightarrow NO + N$
UV - FUV (1100-4000 Å)	-WHAT IS THE SOURCE OF N? ATMOSPHERIC? ATOM EXCHANGE? I.E. $O + N_2 \rightarrow NO + N$
IR (.8-20 MICRONS)	- H_2O IS A BIG PLAYER, IS ODD N ALSO? -WHY IS DAYTIME SO BRIGHT AT 3.0 MICRONS?

GLOW TECHNOLOGY ISSUES

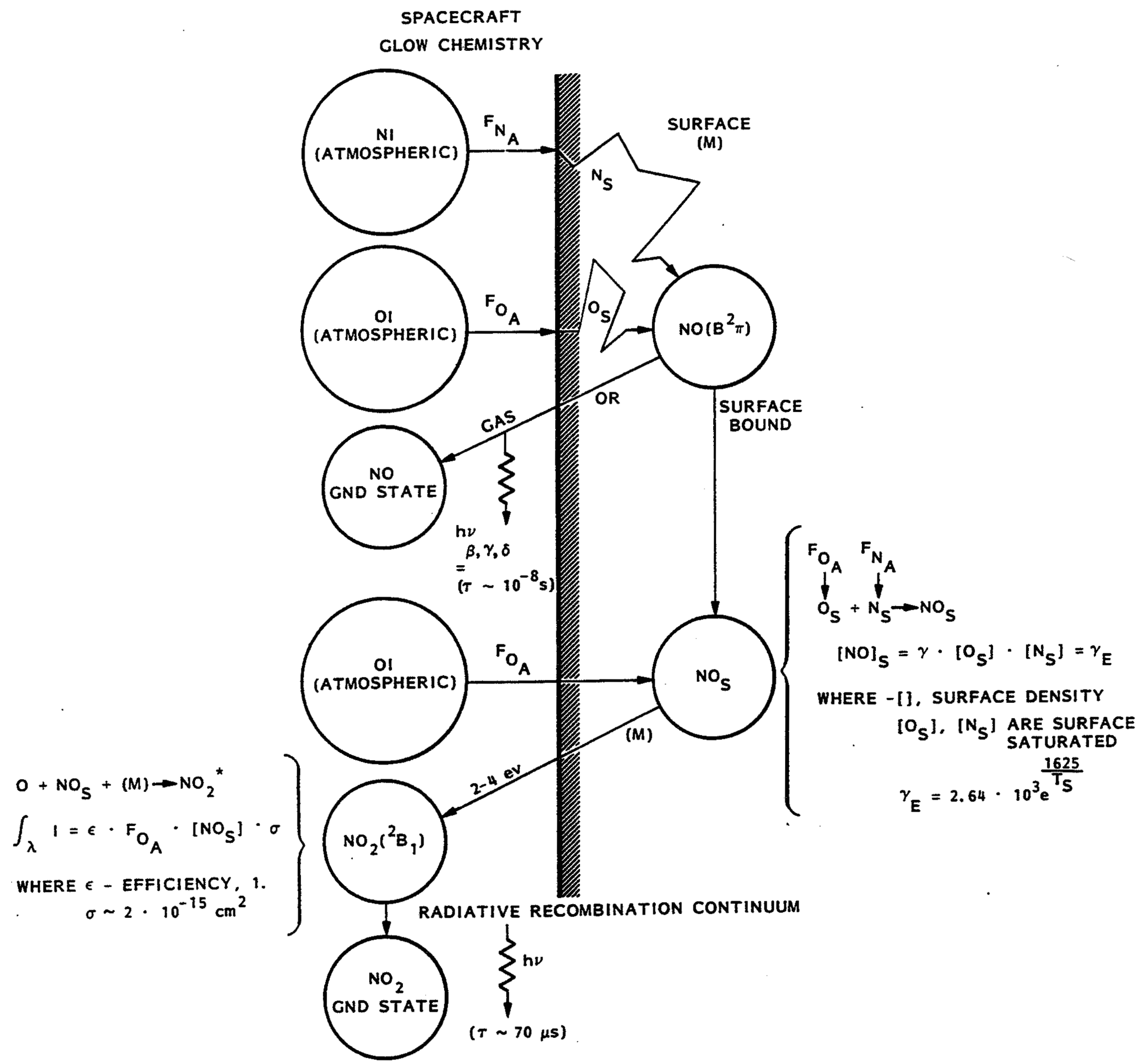
IDENTIFICATION OF THE PHYSICAL PROCESS

HETEROGENEOUS GLOWS (I.E. THOSE STIMULATED BY SURFACES)

- SURFACE MATERIAL
- SURFACE TEMPERATURE
- CONSTITUENTS INVOLVED
  (I.E. RAM ATMOSPHERE (ALTITUDE), THRUSTERS,
  PLOW CLOUD INTERACTIONS (VEHICLE SIZE), OUT/OFF GAS)
- CONSTITUENT ENERGY (VELOCITY)

HOMOGENEOUS GLOWS (I.E. THOSE INVOLVING GAS-GAS PROCESSES)

- CONSTITUENTS INVOLVED
  (I.E. RAM ATMOSPHERE (ALTITUDE), THRUSTERS,
  PLOW CLOUD INTERACTIONS (VEHICLE SIZE), OUT/OFF GAS)
- CONSTITUENT ENERGY (VELOCITY)

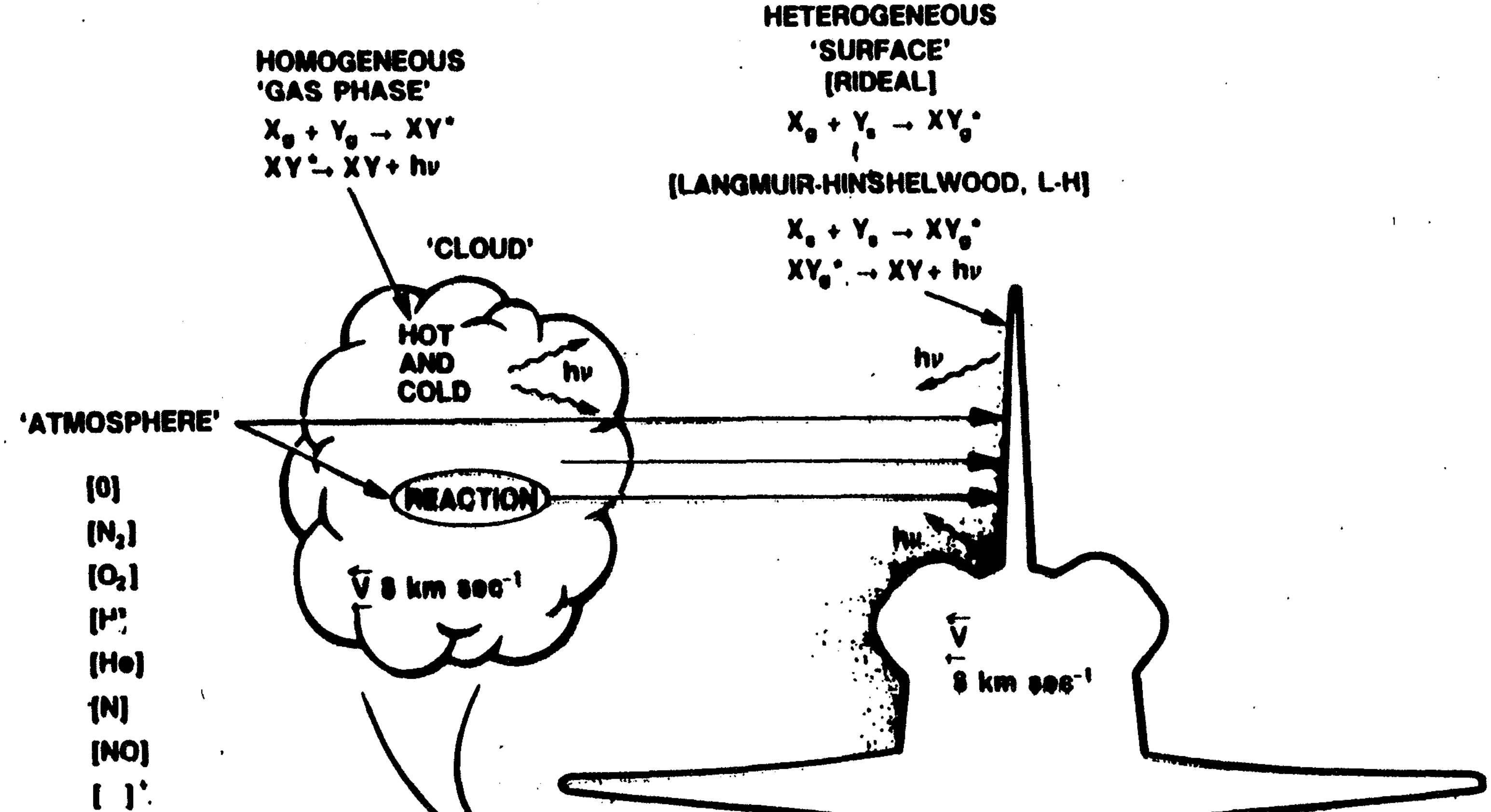


SPACECRAFT
GLOW CHEMISTRY
NI (ATMOSPHERIC)
F_{N_A}
SURFACE (M)
N_S
OI (ATMOSPHERIC)
F_{O_A}
O_S
$NO(B^2\pi)$
OR
GAS
SURFACE BOUND
NO GND STATE
$h\nu$
β, γ, δ
$(\tau \sim 10^{-8} s)$
F_{O_A} F_{N_A}
$O_S + N_S \rightarrow NO_S$
$[NO]_S = \gamma \cdot [O_S] \cdot [N_S] = \gamma_E$
WHERE -[ ], SURFACE DENSITY
$[O_S]$, $[N_S]$ ARE SURFACE SATURATED
$\gamma_E = 2.64 \cdot 10^3 e^{\frac{1625}{T_S}}$
OI (ATMOSPHERIC)
F_{O_A}
NO_S
(M)
2-4 ev
$O + NO_S + (M) \rightarrow NO_2^*$
$\int_\lambda I = \epsilon \cdot F_{O_A} \cdot [NO_S] \cdot \sigma$
WHERE ϵ - EFFICIENCY, 1.
$\sigma \sim 2 \cdot 10^{-15}$ cm^2
$NO_2(^2B_1)$
RADIATIVE RECOMBINATION CONTINUUM
$h\nu$
NO_2 GND STATE
$(\tau \sim 70\ \mu s)$

GLOW CONTAMINATIONS (MAJOR KNOWN)

SPECTRAL REGION	CONTAMINANT (MAJOR)
VISIBLE (4000-8000 Å)	-NO_2 FROM SURFACE RECOMBINATION I.E. $NO + O \rightarrow NO_2^*$ (4000-8000 Å)
UV - FUV (1100-4000 Å)	-N_2 FROM SURFACE RECOMBINATION I.E. $N + N \rightarrow N_2^*$ (1400-1800 Å)
	-NO (1900-2200 Å) ??
	-N_2 (2600-3400 Å) ?? 2P OR GH BANDS?
	-O_2 FROM SURFACE RECOMINATION I.E. $O + O \rightarrow O_2^*$ (2800-3800 Å)
IR (.8-20 MICRONS)	-H_2O FROM COLLISIONAL EXCITATION I.E. $H_2O + O \rightarrow H_2O^* + O$ (2.8 - 15 MICRONS)
	-NO, $NO^{\pm}$ EMISSIONS?? (4-5.4 MICRONS)
	-$H_2O^{\pm}$ EMISSIONS FROM CHARGE EXCHANGE?? I.E. $H_2O + O^+ \rightarrow H_2O^{+*} + O$ (2.6-3.3 MICRONS)
	-OH EMISSIONS?? (1.4-3 MICRONS)

CHEMICAL PROCESSES



SOURCE

PLOW CLOUD, $[\ ]_{CD} \sim 30 \cdot LT \cdot [\ ]$

SURFACE OUTGASSING

LIFE SUPPORT LEAKS; N_2, O_2

DUMPS; H_2O --

CRYOGENS; N_2, He ---

VENTING

THRUSTERS; HYDRAZINE ---

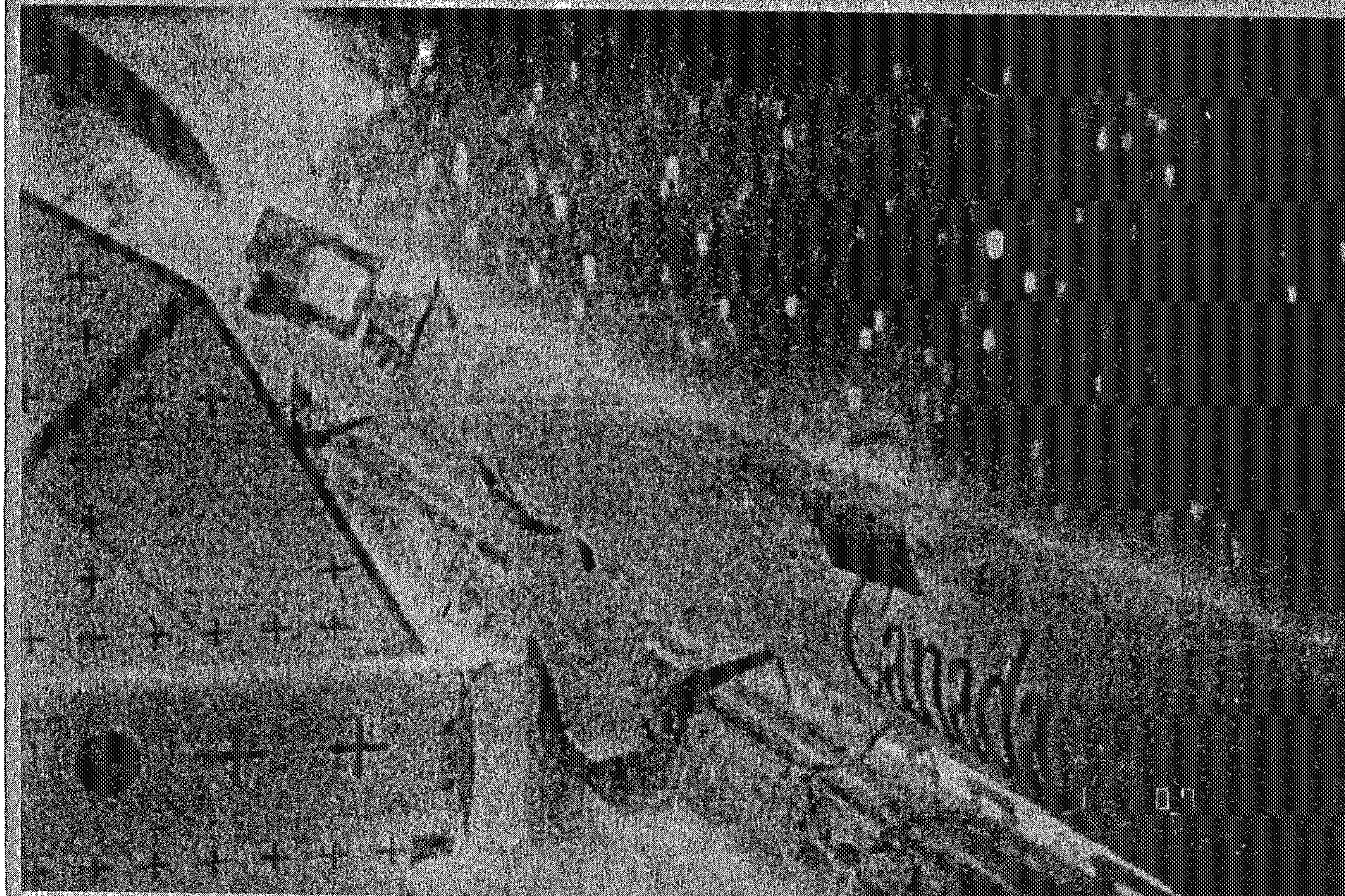
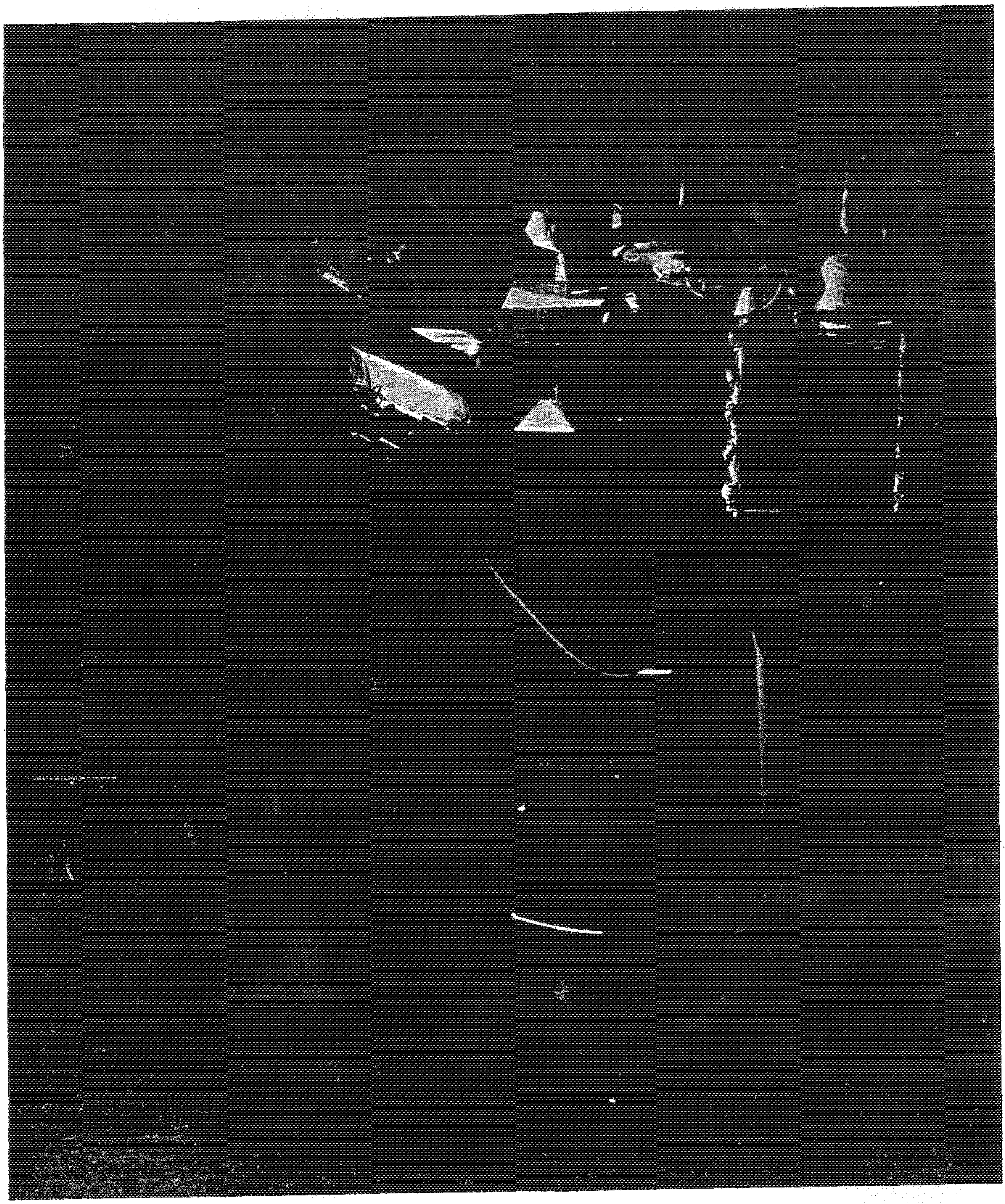


Figure 2. In order to investigate the dependence of the glow intensity on the properties of the spacecraft surface materials samples were mounted on the remote manipulator system (RMS) arm. The samples are in the following sequence: Kapton, Aluminum, Black chem-glaz, Aluminum, Kapton. The image shows a much brighter glow above the chem-glaz than the other two materials. (See Mende et al., 1985)



N93-28725

MEASUREMENT OF THE RELATIVE AND ABSOLUTE MICROGRAVITY ENVIRONMENT OF SPACE PLATFORMS, RECENT FLIGHT RESULTS AND DETERMINATION OF THE SPATIAL MICROGRAVITY FIELD

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159230
p. 19

ABSTRACT

Microgravity materials processes in space, such as crystal growth, require specific and accurate knowledge of the very low frequency, very low level microgravity. This means knowledge of absolute microgravity levels.

Most acceleration measurement systems for space platforms only measure "microvibrations", the measurement of the absolute microgravity level being severely corrupted by bias errors, temperature hysteresis, non-linearity by-products and bias changes due to launch vibrations.

An advanced system was developed incorporating a fixed 3-axes acceleration measurement system with improved signal conditioning and Invertible Accelerometers for absolute measurements. The higher sensitivity channels have lower frequency low pass filtering to meet crystal growers demands for recording of low level/low frequency disturbances. Also included is a pre-flight bias adjustment feature.

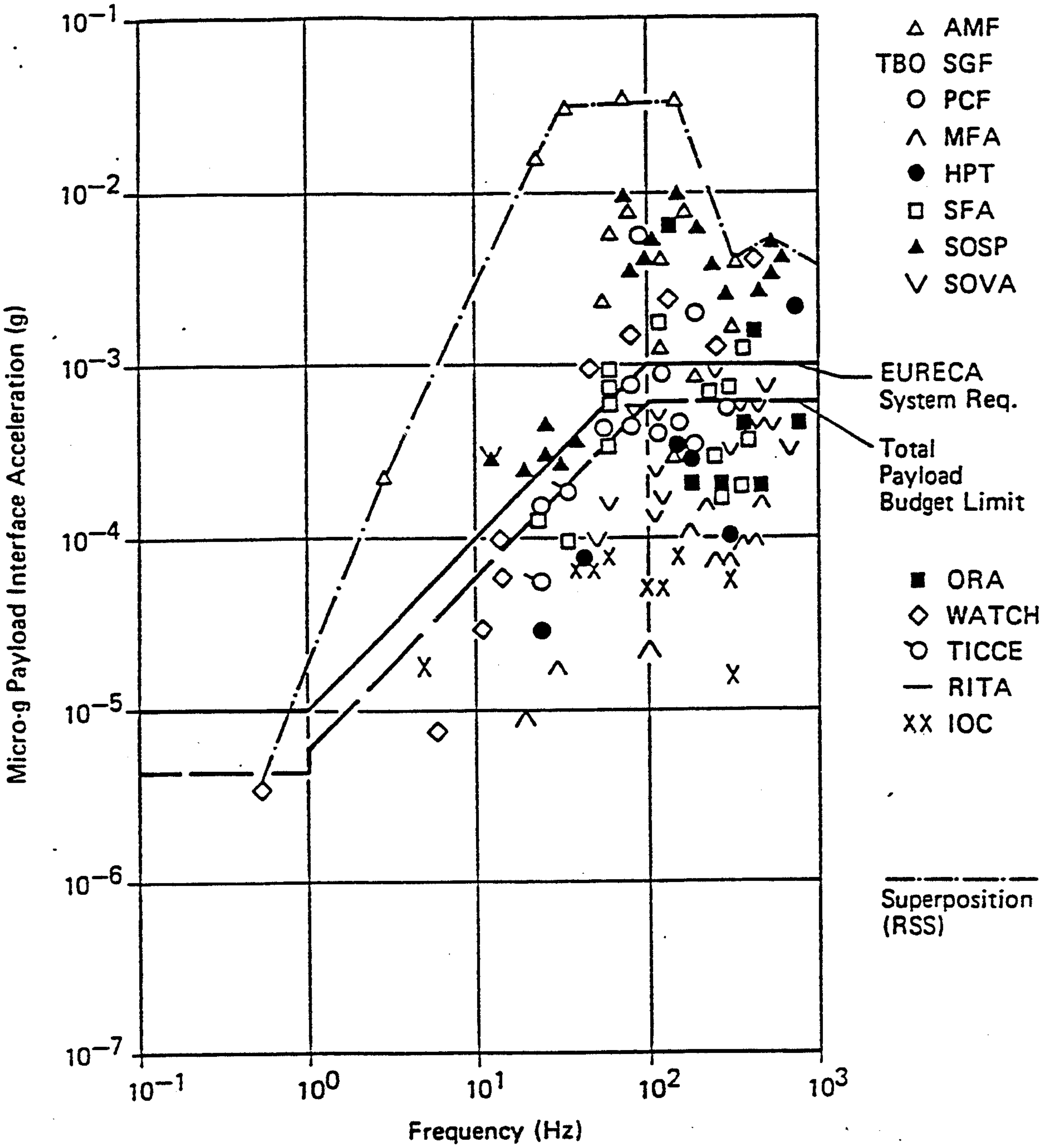
The Invertible Accelerometers invert the accelerometer sensitive axis over 180° at regular intervals permitting determination of both the in-space bias error and the residual absolute microgravity level.

Systems flew on the "CONSORT" Sounding Rockets in 1989, '90 and '91. A system has been assembled for flight on STS-46 for absolute measurement of the force on the orbiter from a tethered satellite and for the COMET satellite. A triple system is under construction for SPACEHAB-01 for flight in 1993.

An advanced display and analysis computer program was developed for unambiguous correlation with disturbing events from other experiments and

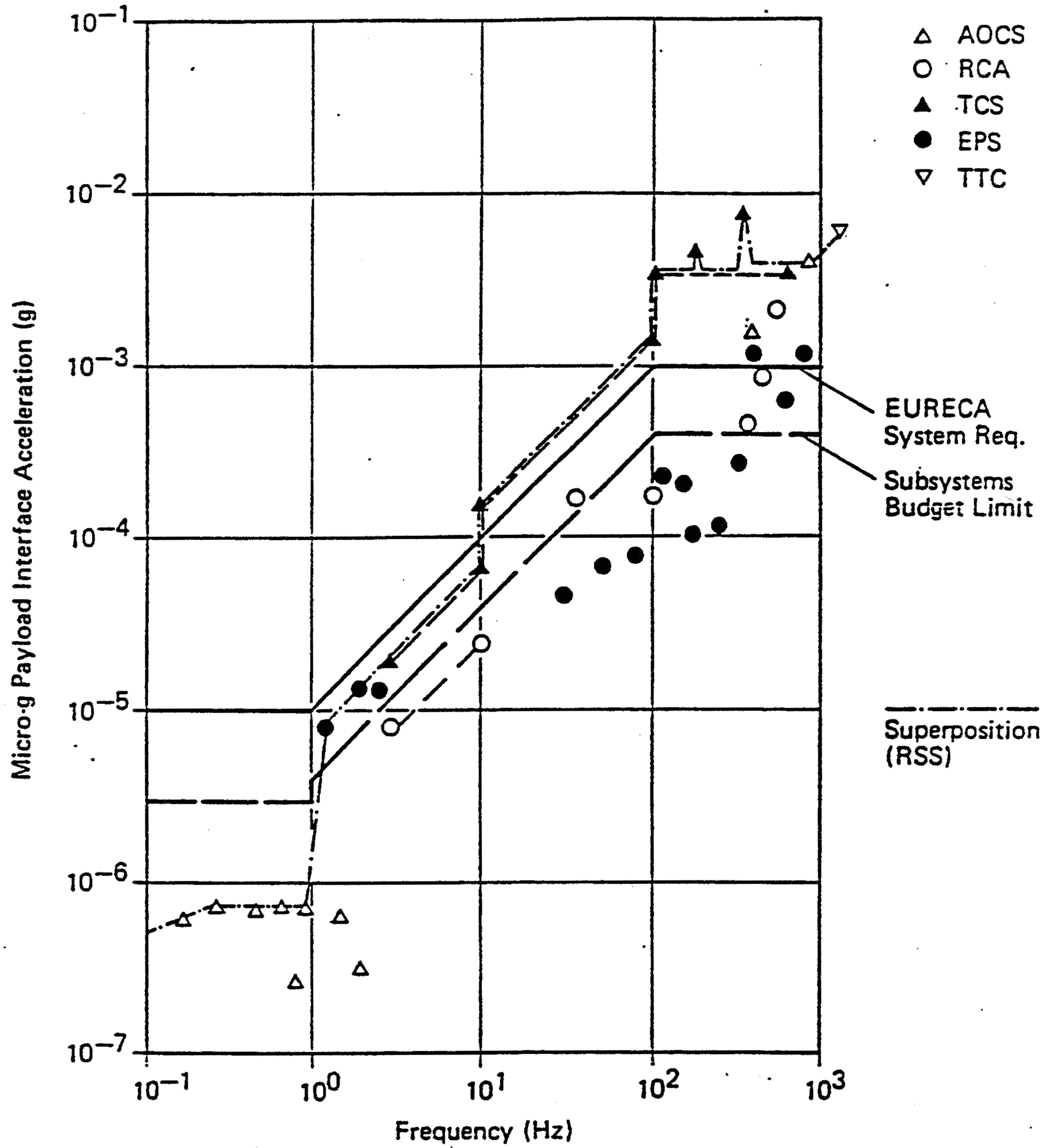
subsystems. Typical results on determining absolute microgravity levels and on the correlation between weak disturbances and events such as solenoid-and camera-operations will be shown. Full correlation between events and disturbances was obtained.

Above activities are also intermediate steps to a full system for determining the complete spatial microgravity field from data from a fixed number of accelerometers with different characteristics mounted on fixed locations on a space platform such as a space station.



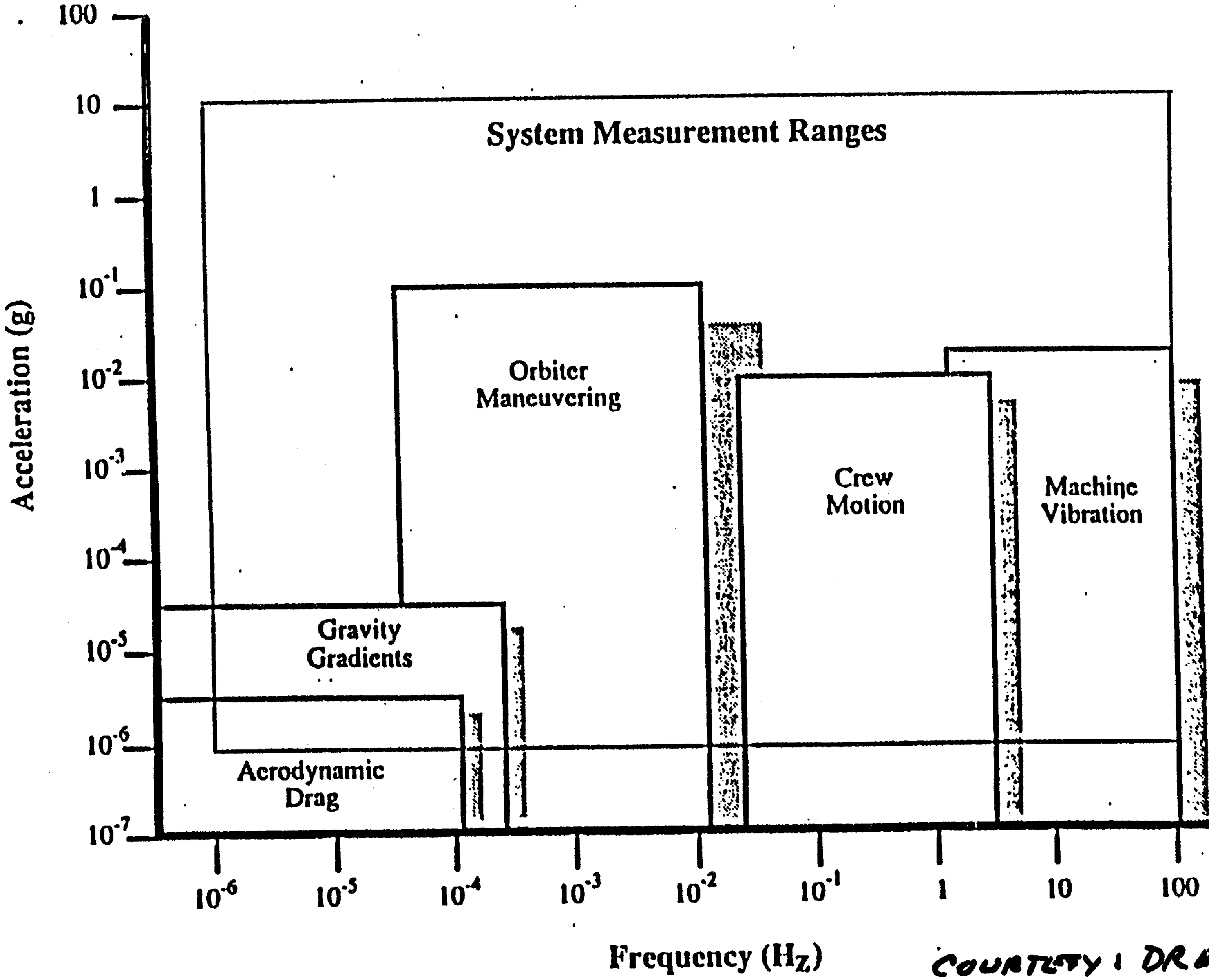
COURTESY DR ANDRESEN, ESA

Figure 6.2: Acceleration Response Spectrum due to Operating Facilities/Instruments of the EURECA Payload



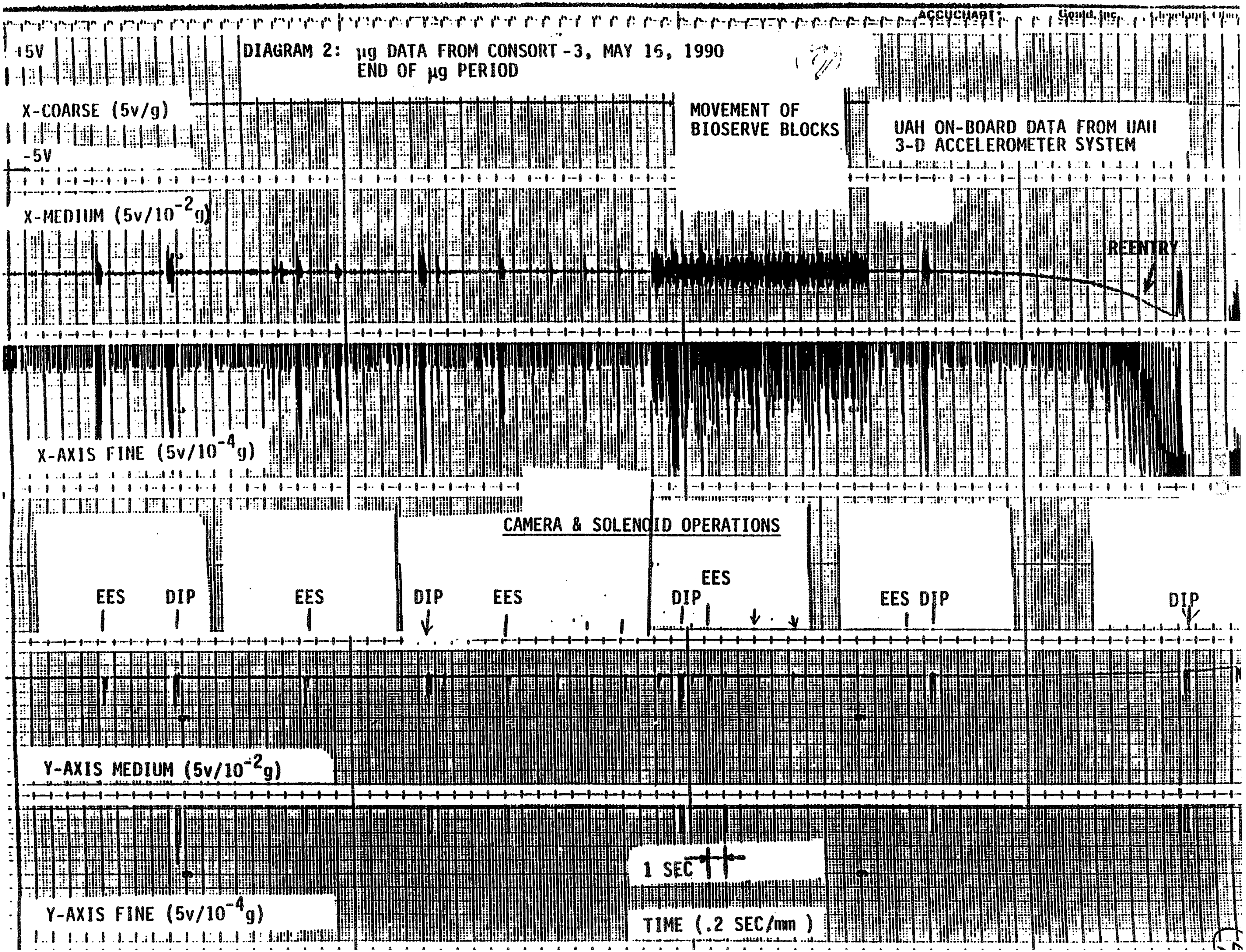
COURTESY : DR ANDRESEN, ESA

Figure 6.1: Acceleration Response Spectrum due to Subsystem Equipment

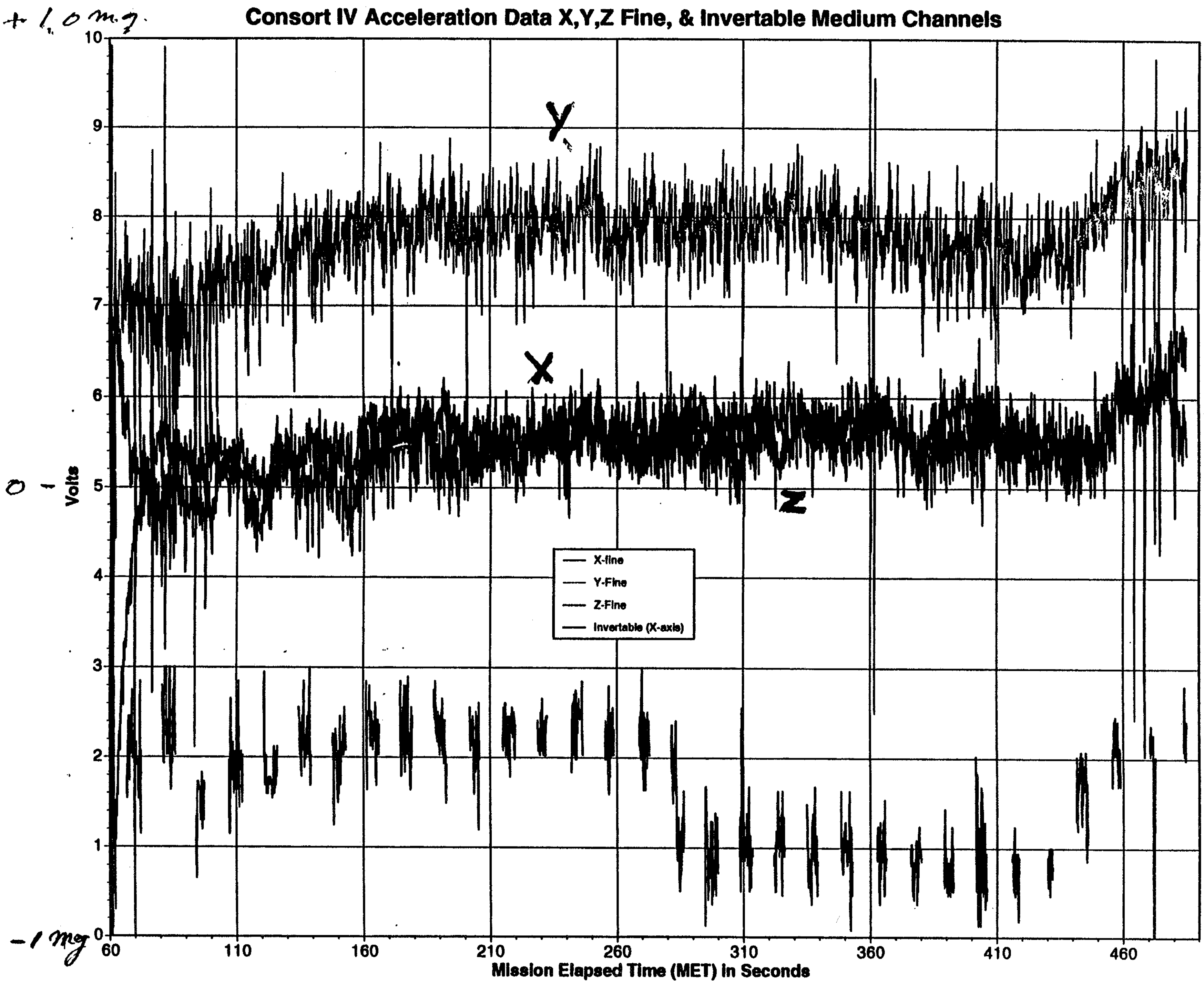


COURTESY: DR. E. STUHLINGER
HUNTSVILLE, AL

3



ACCUCHART
Gould Inc.
Cleveland, Ohio
+5V
DIAGRAM 2: μg DATA FROM CONSORT-3, MAY 16, 1990
END OF μg PERIOD
X-COARSE (5v/g)
-5V
MOVEMENT OF
BIOSERVE BLOCKS
UAH ON-BOARD DATA FROM UAH
3-D ACCELEROMETER SYSTEM
X-MEDIUM (5v/10^-2 g)
REENTRY
X-AXIS FINE (5v/10^-4 g)
CAMERA & SOLENOID OPERATIONS
EES
DIP
EES
DIP
EES
DIP
EES
EES
DIP
DIP
Y-AXIS MEDIUM (5v/10^-2 g)
1 SEC
Y-AXIS FINE (5v/10^-4 g)
TIME (.2 SEC/mm)



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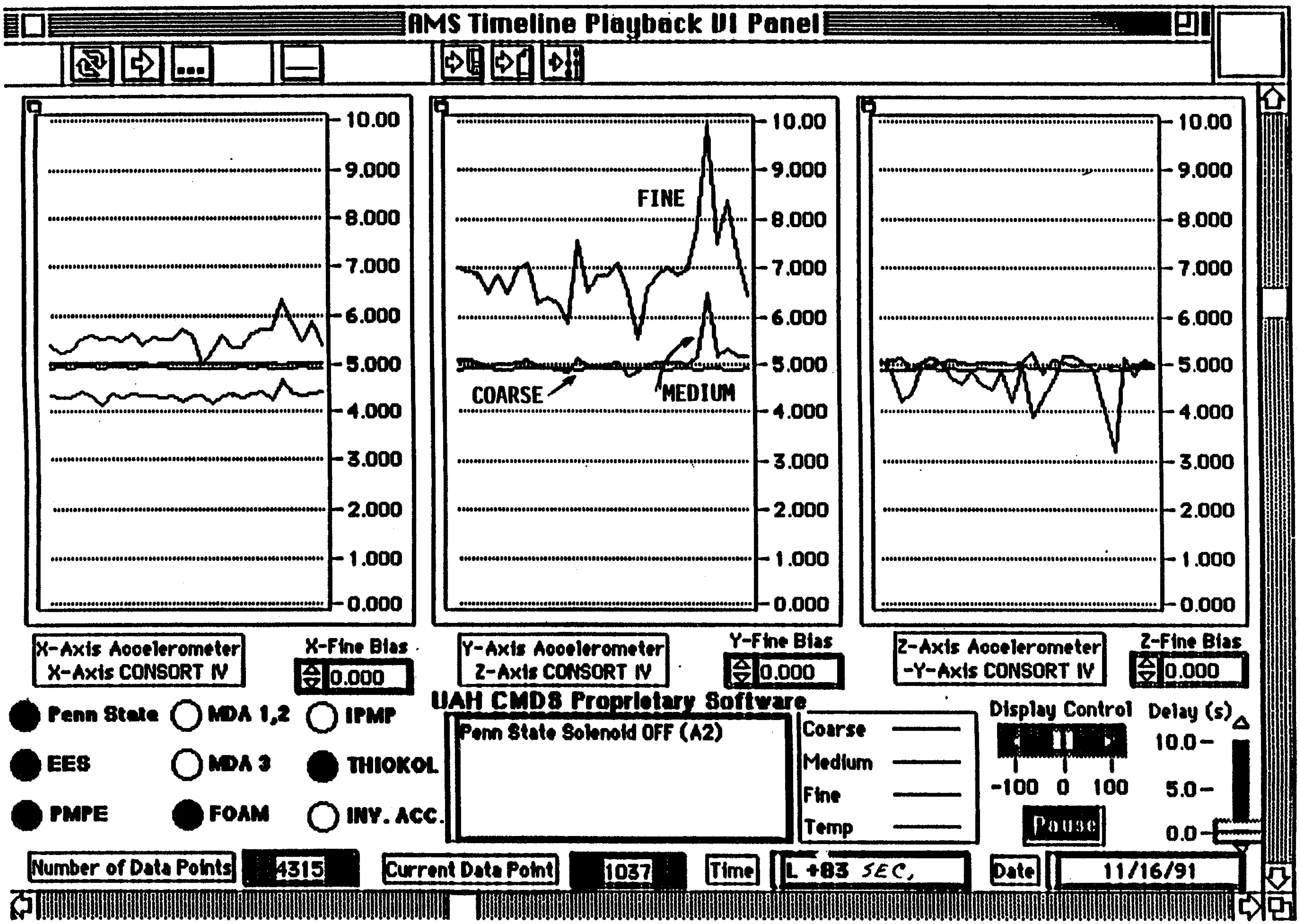
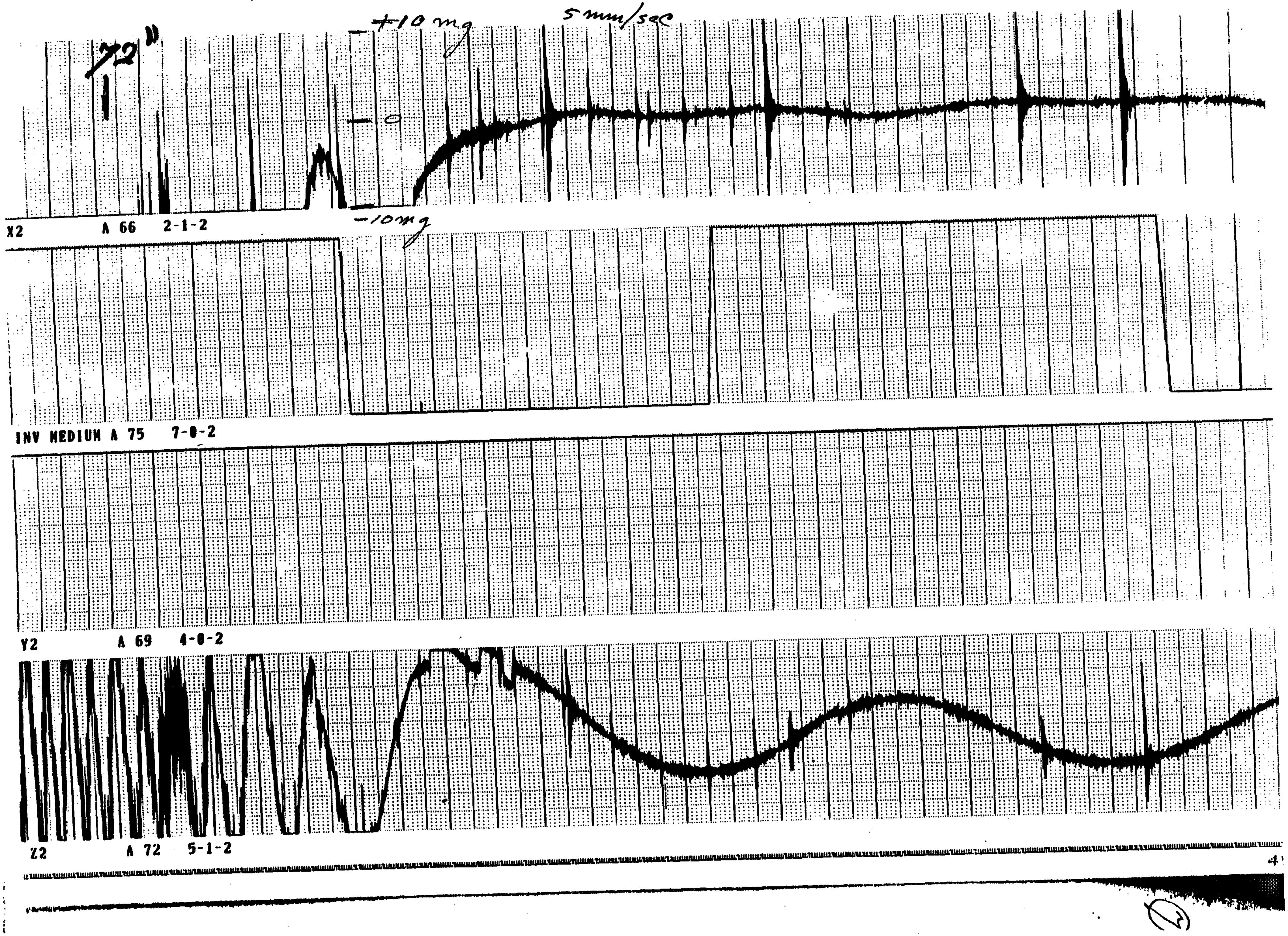
5

DIAGRAM 4: CONSORT -IV ACCELERATION DATA DISPLAY & CORRELATION



7



72"
+10 mg
5 mm/sec
0
-10 mg
X2 A 66 2-1-2
INV MEDIUM A 75 7-0-2
Y2 A 69 4-0-2
Z2 A 72 5-1-2

OBJECTIVES 3-D MICROGRAVITY ACCELEROMETER PROGRAM

- MEET SPECIFIC REQUIREMENTS MATERIALS PROCESSING IN SPACE
- ACCURATE MEASUREMENT OF ABSOLUTE MICROGRAVITY LEVELS IN SPACE
- ELIMINATION OF BIAS SHIFTS CAUSED BY LAUNCH VIBRATIONS
- EMPHASIS ON LOW FREQUENCY DISTURBANCES
- NO SUPPRESSION OF CONSTANT COMPONENTS SUCH AS DUE TO DRAG,GRAVITY GRADIENT, MOMENTS.
- ELIMINATION OF DC ERRORS CAUSED BY HIGH FREQUENCY HIGH LEVEL DISTURBANCES
- CLOSE CORRELATION OF MEASURED DISTURBANCES AND ON-BOARD EVENTS

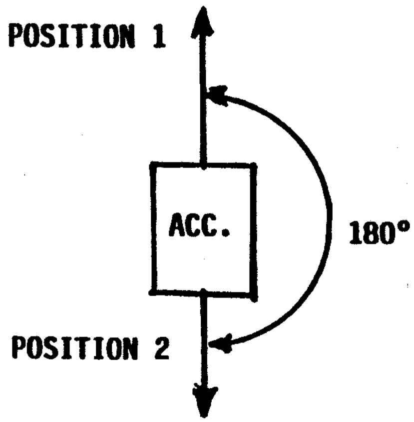
DIAGRAM 1

JAB
19 NOV 91

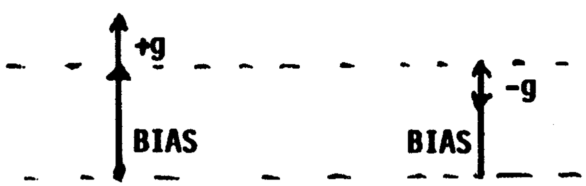
ABSOLUTE μG-MEASUREMENT

INVERTABLE ACCELEROMETER:

180° ROTATIONS BETWEEN MEASUREMENTS



- °BIAS ERROR RESULTS FROM
  - •TEMPERATURE HYSTERESIS
  - •LAUNCH VIBRATIONS
  - •NON-LINEAR RESPONSE TO HIGH LEVEL/HIGH FREQUENCY SIGNALS
- °BIAS ERROR CONSTANT WHEN ROTATED
- °G INDICATION INVERTING



$$\begin{cases} B + g \\ B - g \end{cases}$$

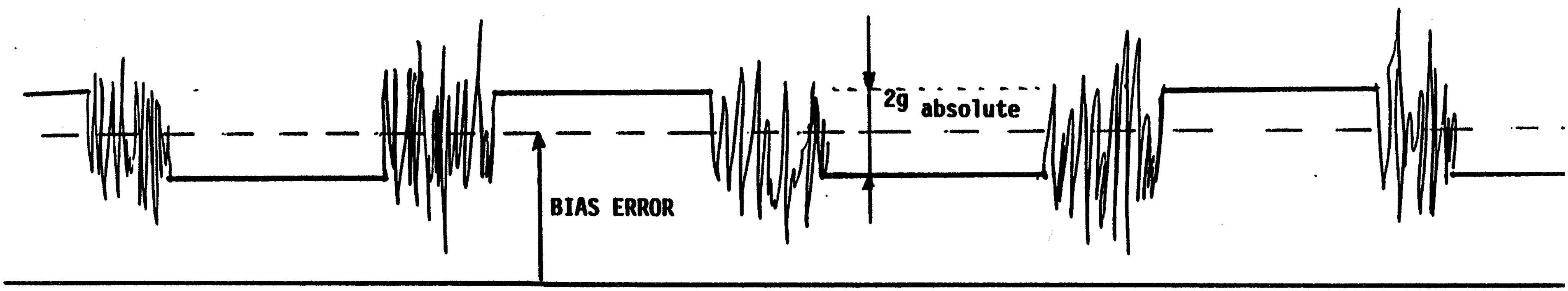
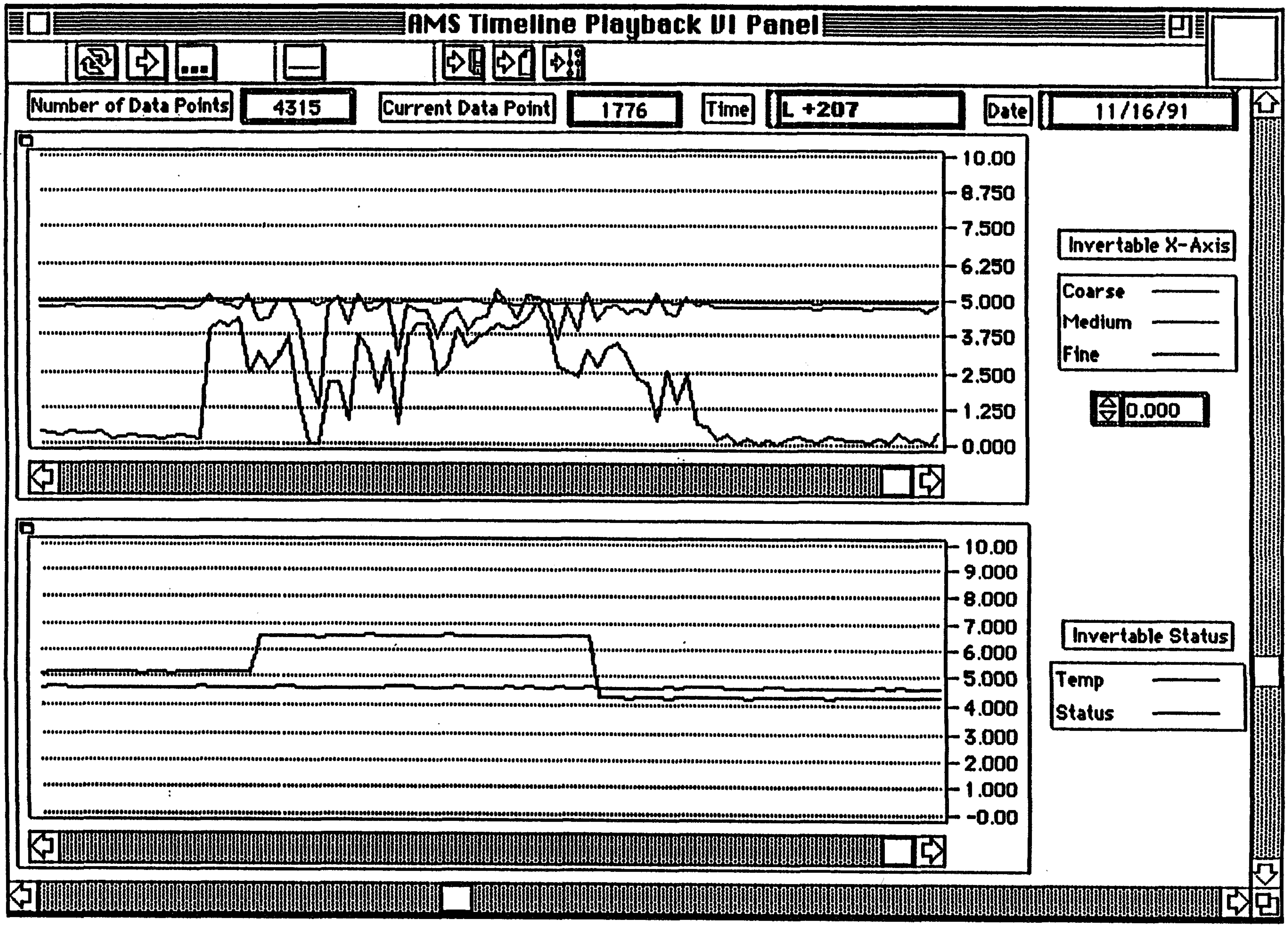
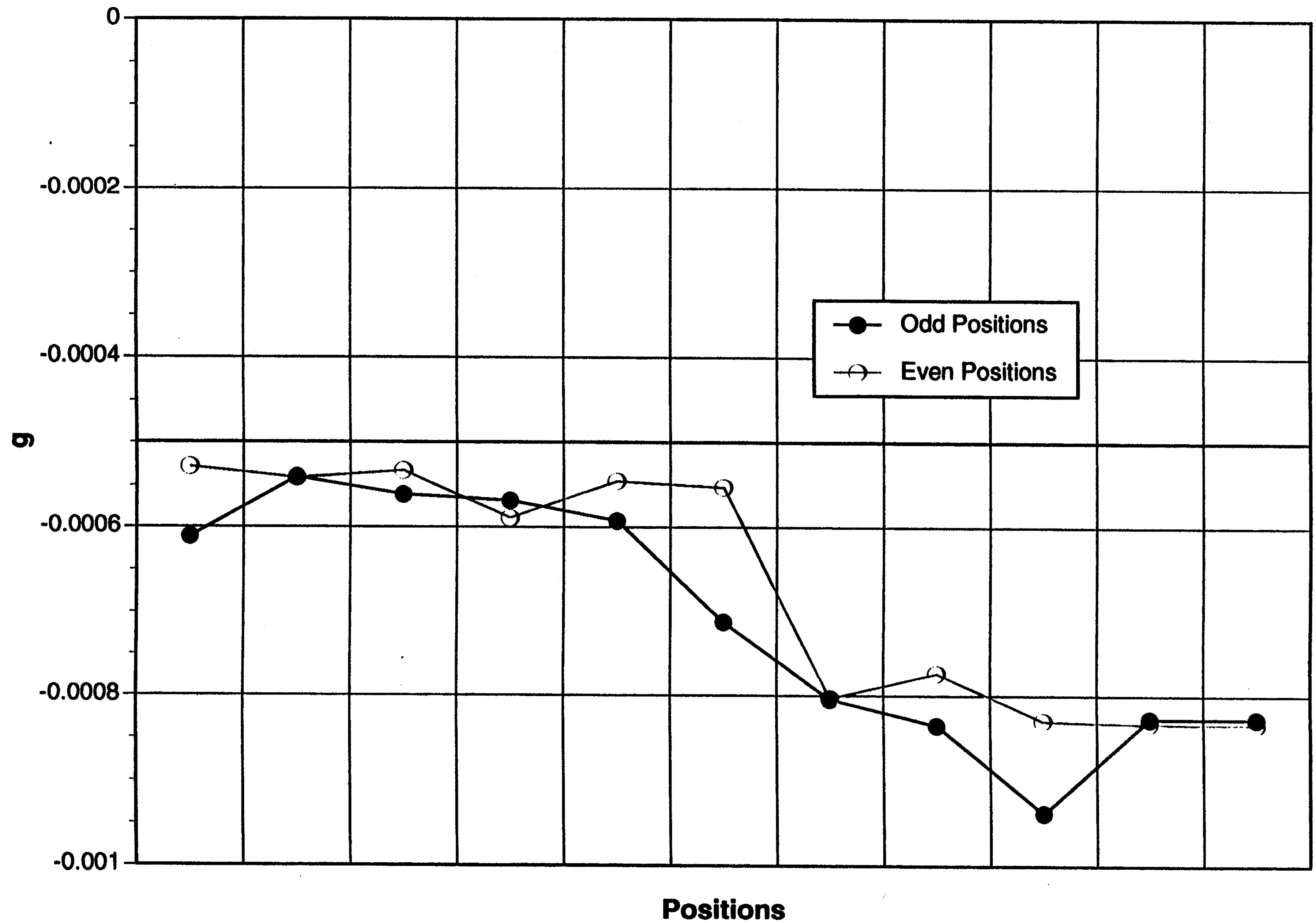


DIAGRAM 5: INVERTIBLE ACCELEROMETER DATA INTERVAL

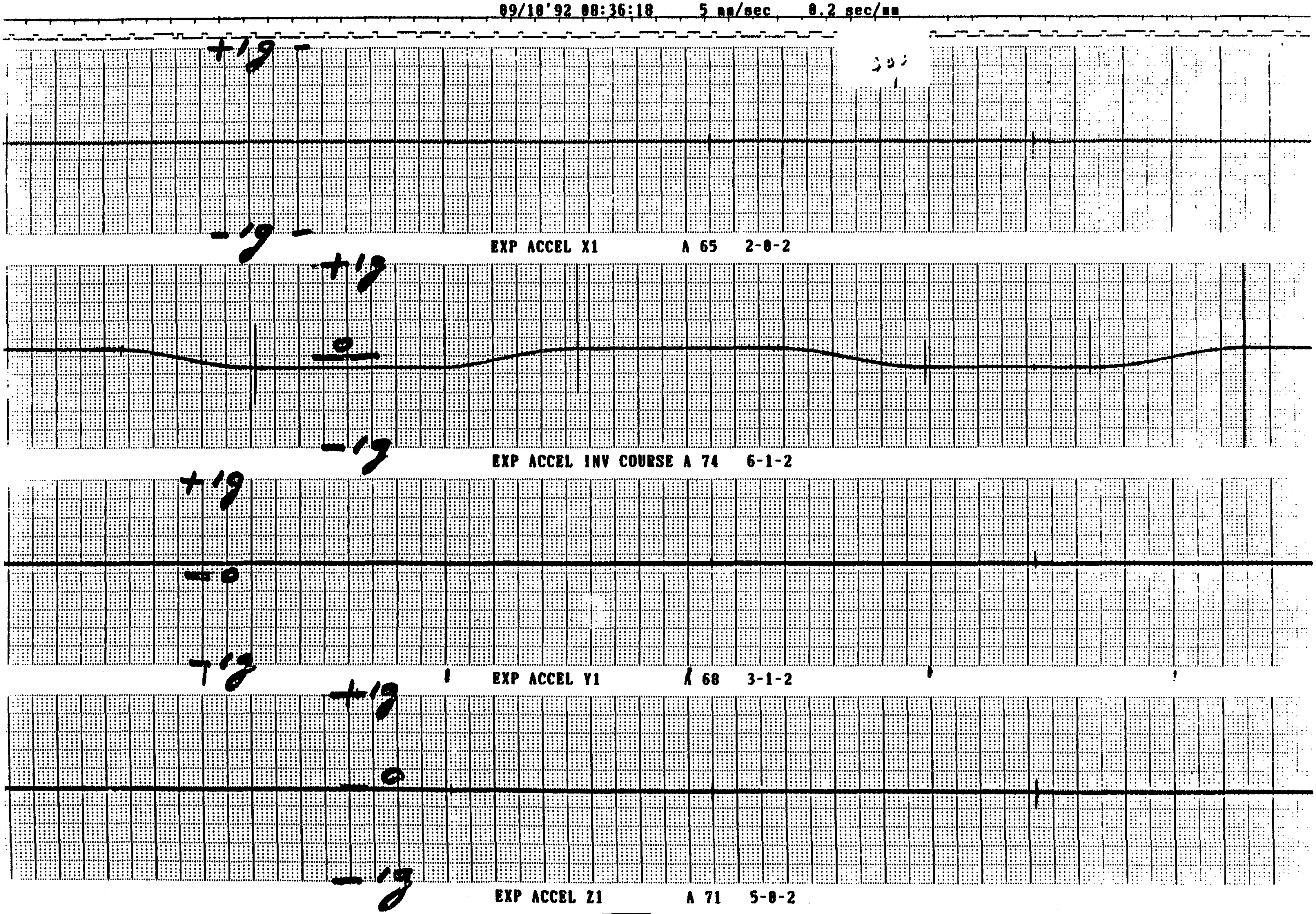


(10)

CONSORT IV Invertable Accelerometer: Alternate Position Averages



11


09/10'92 08:36:10 5 ms/sec 0.2 sec/mm
+1g
-1g
EXP ACCEL X1 A 65 2-0-2
+1g
0
-1g
EXP ACCEL INV COURSE A 74 6-1-2
+1g
0
-1g
EXP ACCEL Y1 A 68 3-1-2
+1g
0
-1g
EXP ACCEL Z1 A 71 5-0-2

[redacted]

1

CMDS/UAH ACCELERATION MEASUREMENT DEVELOPMENT PROGRAM

MISSIONS	DATES	INSTRUMENTATION	REMARKS
CONSORT 1,2,3	1989/90	UAH 3DMA + MSFC LGAS	SLOW RESPONSE LGAS CONSORT 2 MISHAP BIAS SHIFTS
JOUST 1	1991	NEW UAH - 3DMA + INVERTIBLE ACCELEROMETER (IA)	LAUNCH MISHAP-ALL H/W LOST ALL EVENTS RECORDED
CONSORT 4	NOV. '91	NEW 3DMA + I.A. (QA -700'S)	BIAS SHIFTS
STS 46, CONCAP-III	AUG. '92	3DMA + I.A. + DATA PROCESSING + RECORDING	TETHER MISSION FAILED SYSTEM NOT TURNED ON
CONSORT 5	SEP. '92	3DMA + I.A. (QA-3000'S)	FLIGHT MISHAP-NO μG ABSOLUTE G MEASURED ALL EVENTS RECORDED
COMET GU		3DMA-(QA 700) FOR EXPERIMENT TESTS ON THE GROUND	START TESTS SEP. 92
COMET FU	SPRING 93	3DMA (QA 3000) + TELEMETRY TO GROUND	DELIVERY SEP. '92
SPACEHAB-01, 02	SPRING 93 FALL 93	3DMA (3RU'S) + 3 I.A.'S + DATA PROCESSING + RECORDING	
USML-1	1995	3DMA (3RU'S) + I.A.'S + DATA PROCESSING + RECORDING	

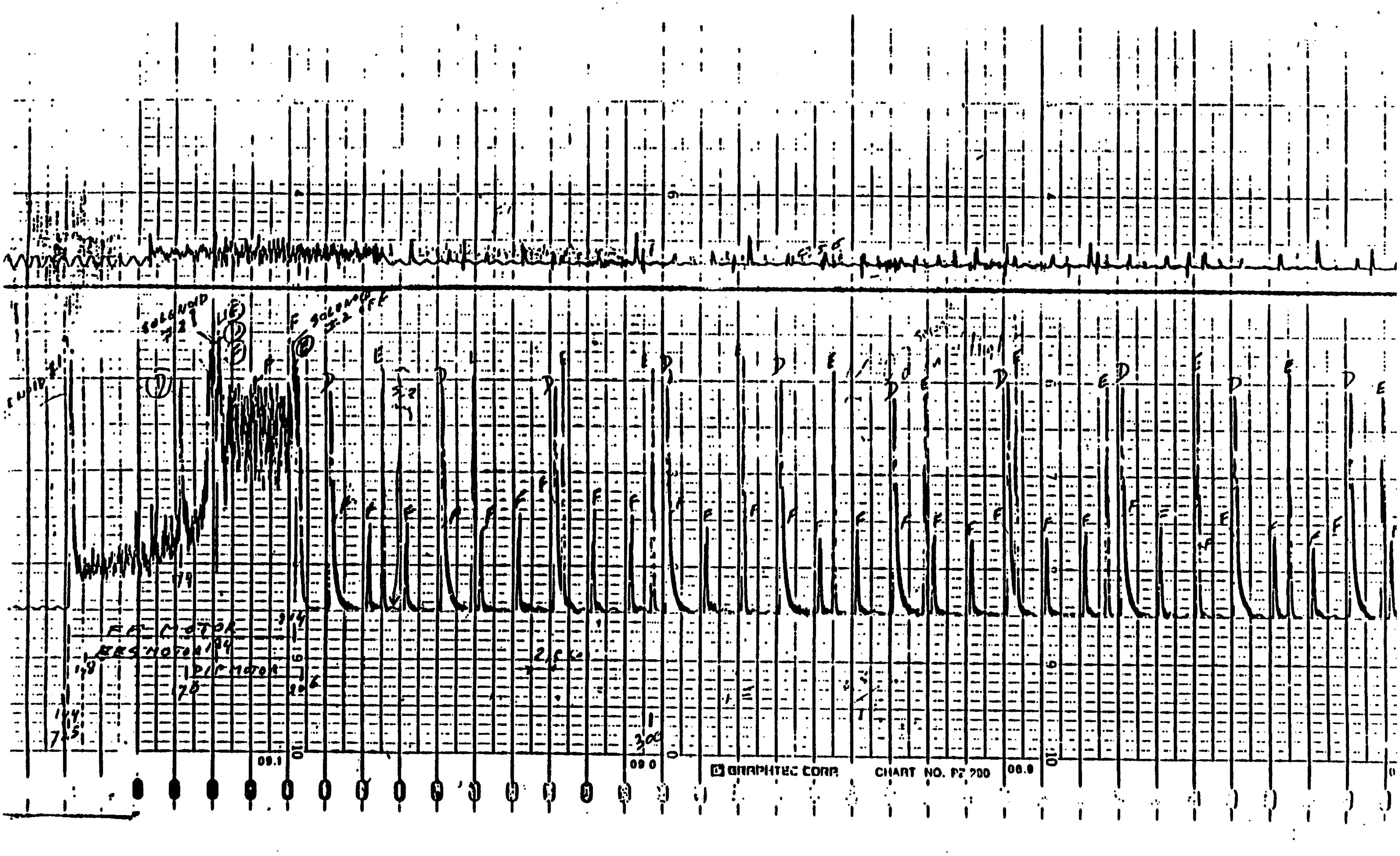


DIAGRAM 7: Camera Disturbances Measured at CMDS Sounding Rocket on-the-ground suspension test.

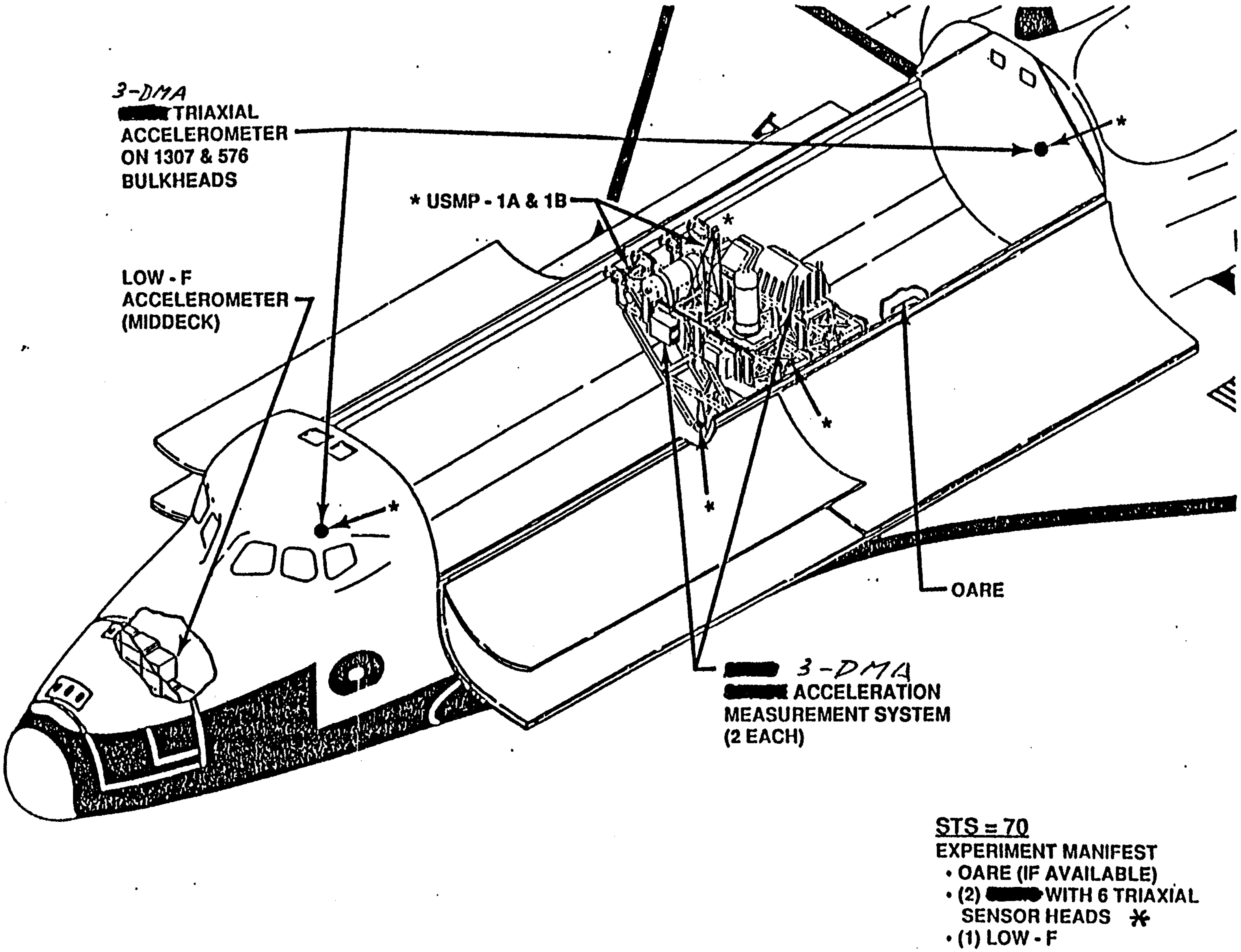


FIGURE 8 - POSSIBLE LOCATIONS OF ACCELEROMETERS ON SHUTTLE

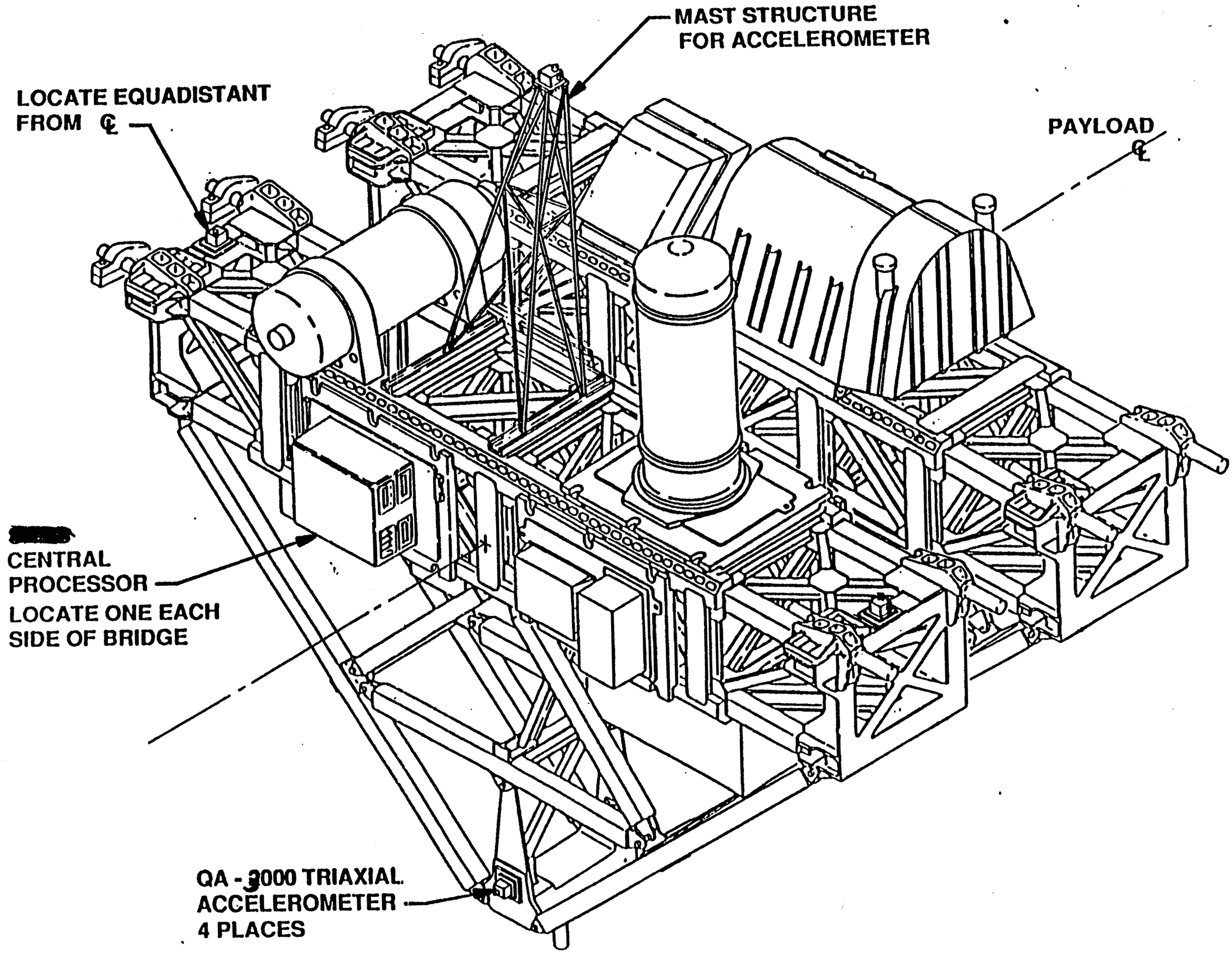
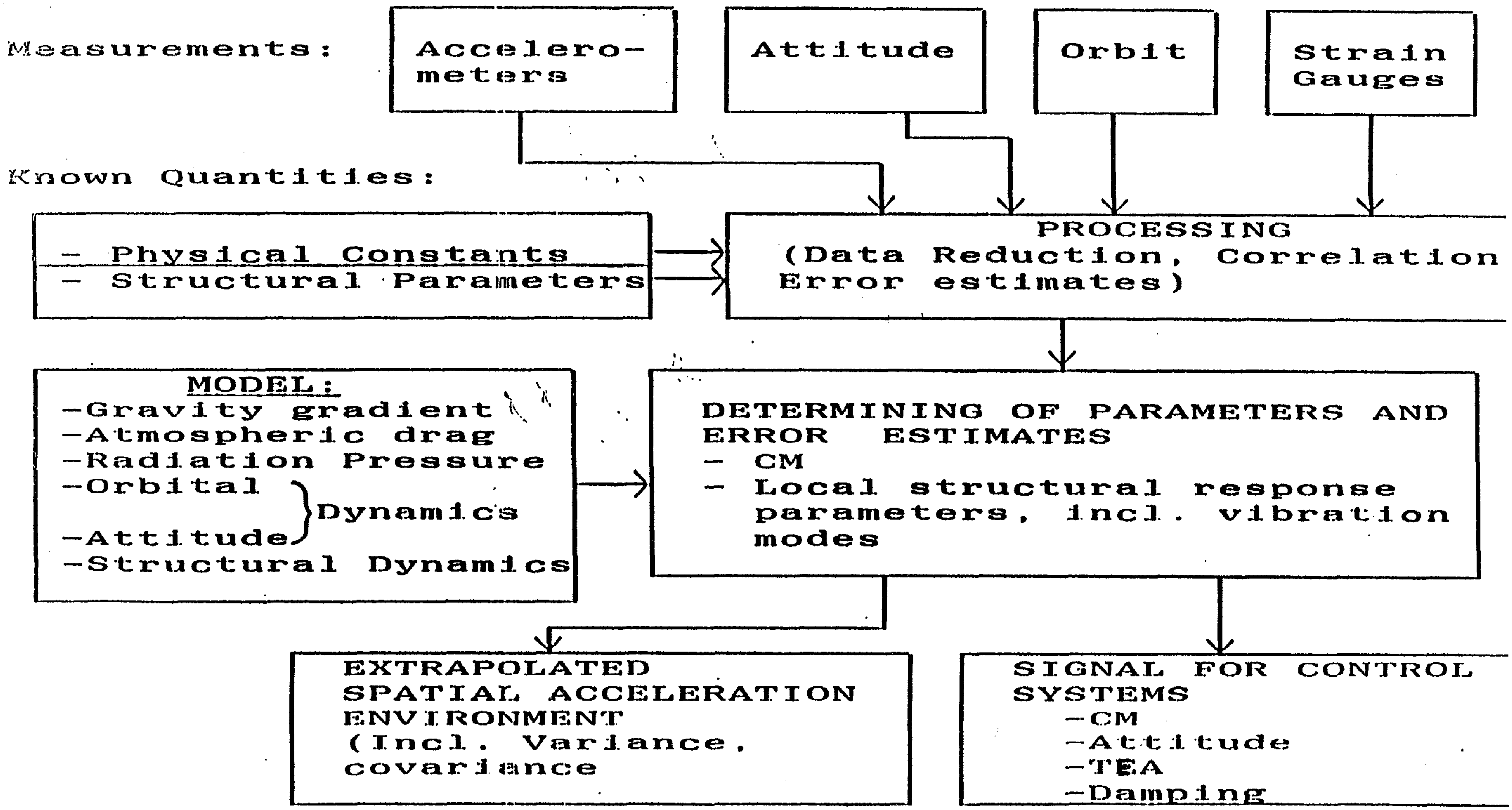


Fig. 9: Locations of Accelerometers and Processor on

FIG. 5.3

OUTREACH	ACCELERATION MEASUREMENT AND MANAGEMENT	– UAH – TBE

M & M PROCESS FLOW



NASA/DOD Flight Experiments
Technical Interchange Meeting
Monterey, California
October 5 - 9, 1992

ENVIRONMENTAL VERIFICATION EXPERIMENT FOR THE EXPLORER PLATFORM (EVEEP)

Laura Ottenstein
NASA/GSFC Code 724.2
Greenbelt, MD 20771

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- MECHANICAL DESIGN AND ANALYSIS - SWALES AND ASSOCIATES
- MECHANICAL FABRICATION - EER SYSTEMS, INC
- ELECTRICAL DESIGN AND FABRICATION - ITE
- SOFTWARE - STX

GSFC

ENVIRONMENTAL VERIFICATION EXPERIMENT FOR THE EXPLORER PLATFORM (EVEEP)

BACKGROUND

- SATELLITES AND LONG-LIFE SPACECRAFT REQUIRE EFFECTIVE CONTAMINATION CONTROL MEASURES TO:

  - ENSURE DATA ACCURACY

  - MAINTAIN OVERALL SYSTEM PERFORMANCE MARGINS

- CONTAMINATION CAN OCCUR FROM EITHER MOLECULAR OR PARTICULATE MATTER

  - SOURCES OF MOLECULAR SPECIES:

    * MASS LOSS FROM NONMETALLIC MATERIALS
    * VENTING OF CONFINED SPACECRAFT OR EXPERIMENT VOLUMES
    * EXHAUST EFFLUENTS FROM ATTITUDE CONTROL SYSTEMS
    * INTEGRATION AND TEST ACTIVITIES
    * IMPROPER CLEANING OF SURFACES

  - SOURCES OF PARTICULATES:

    * LEAKS OR PURGES WHICH CONDENSE UPON VACUUM EXPOSURE
    * ABRASION OF MOVABLE SURFACES
    * MICROMETERIOD IMPACTS

GSFC

ENVIRONMENTAL VERIFICATION EXPERIMENT FOR THE EXPLORER PLATFORM (EVEEP)

BACKGROUND (CONTINUED)

- SPACECRAFT CONTAMINATION CONTROL
  - MATERIALS SELECTION
  - CONTAMINATION MODELING OF EXISTING DESIGN
  - THERMAL VACUUM TEST OF SPACECRAFT WITH CONTAMINATION MONITORS

GSFC

ENVIRONMENTAL VERIFICATION EXPERIMENT FOR THE EXPLORER PLATFORM (EVEEP)

- Launched 7 June 1992 aboard a Delta Rocket
- EVEEP uses Temperature Controlled Quartz Crystal Microbalances (TQCM's) to measure the in-flight molecular contamination present in several locations on the Explorer Platform/Extreme Ultraviolet Explorer (EUVE) spacecraft
- Flight contamination data will be used to validate or update existing contamination modeling programs
- Contamination data was also used to evaluate whether the spacecraft outgassing rates were low enough to open the EUVE telescope doors
- EVEEP includes two Teflon-coated TQCM's which will be used to improve our understanding of the synergistic effects of Ultraviolet radiation and Atomic Oxygen on Teflon.
- Three year mission. During the first six months EVEEP does not have any significant sun time on any component.

GSFC

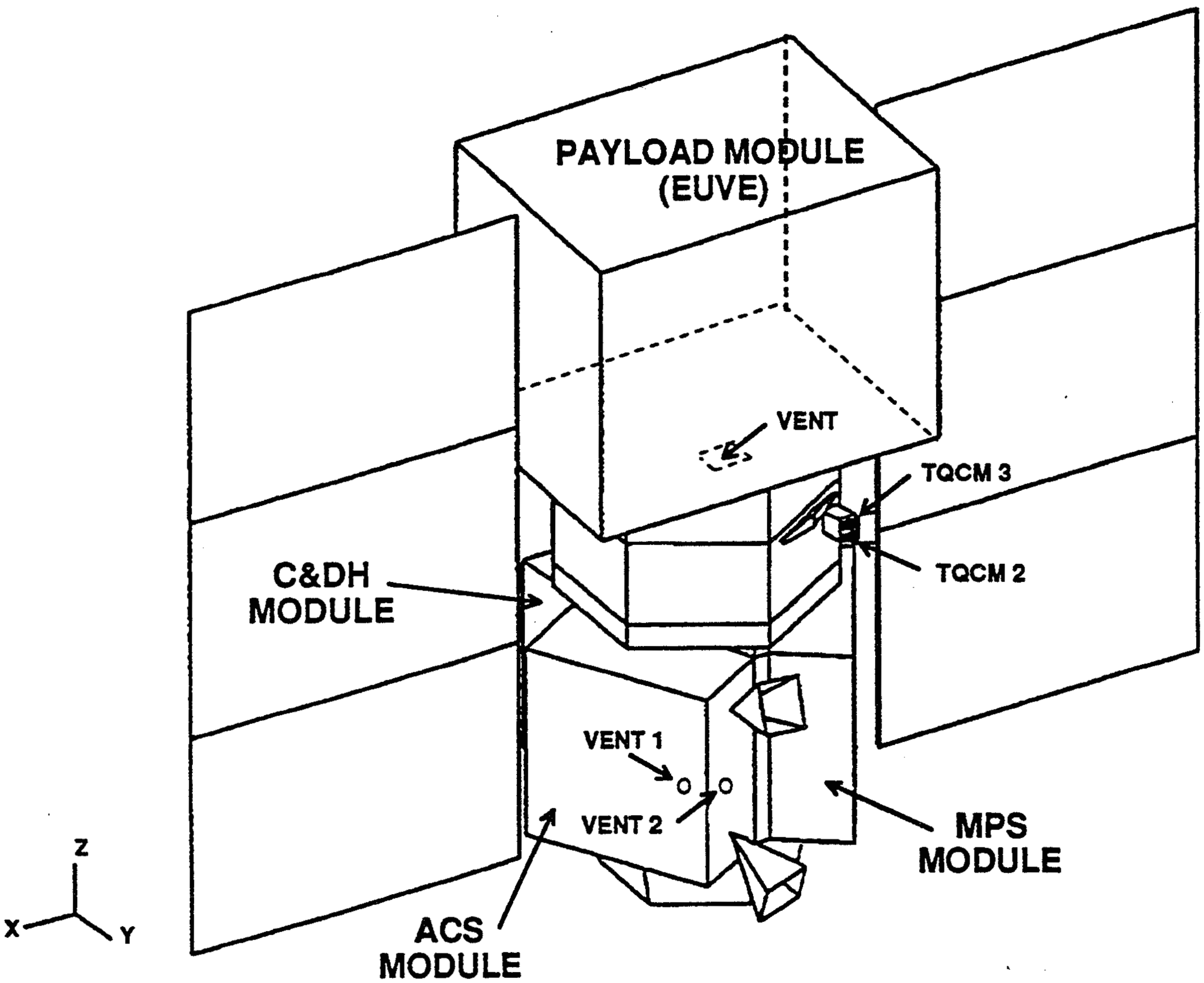
ENVIRONMENTAL VERIFICATION EXPERIMENT FOR THE EXPLORER PLATFORM (EVEEP)

EVEEP CONSISTS OF...

- A housing containing three TQCM's mounted on the shade side of the spacecraft
  - Two TQCM's face in the same direction as the EUVE telescopes; one of these TQCM's is Teflon coated
  - The third TQCM faces down toward the Explorer Platform
- A housing containing two TQCM's mounted on the sun side of the spacecraft
  - Both TQCM's face outward; one is Teflon coated
- An electronics box mounted inside the Explorer Platform
  - The electronics box provides frequency and temperature data to the spacecraft for downlinking
  - The electronics box also allows for ground control of the TQCM temperature setpoints

GSFC

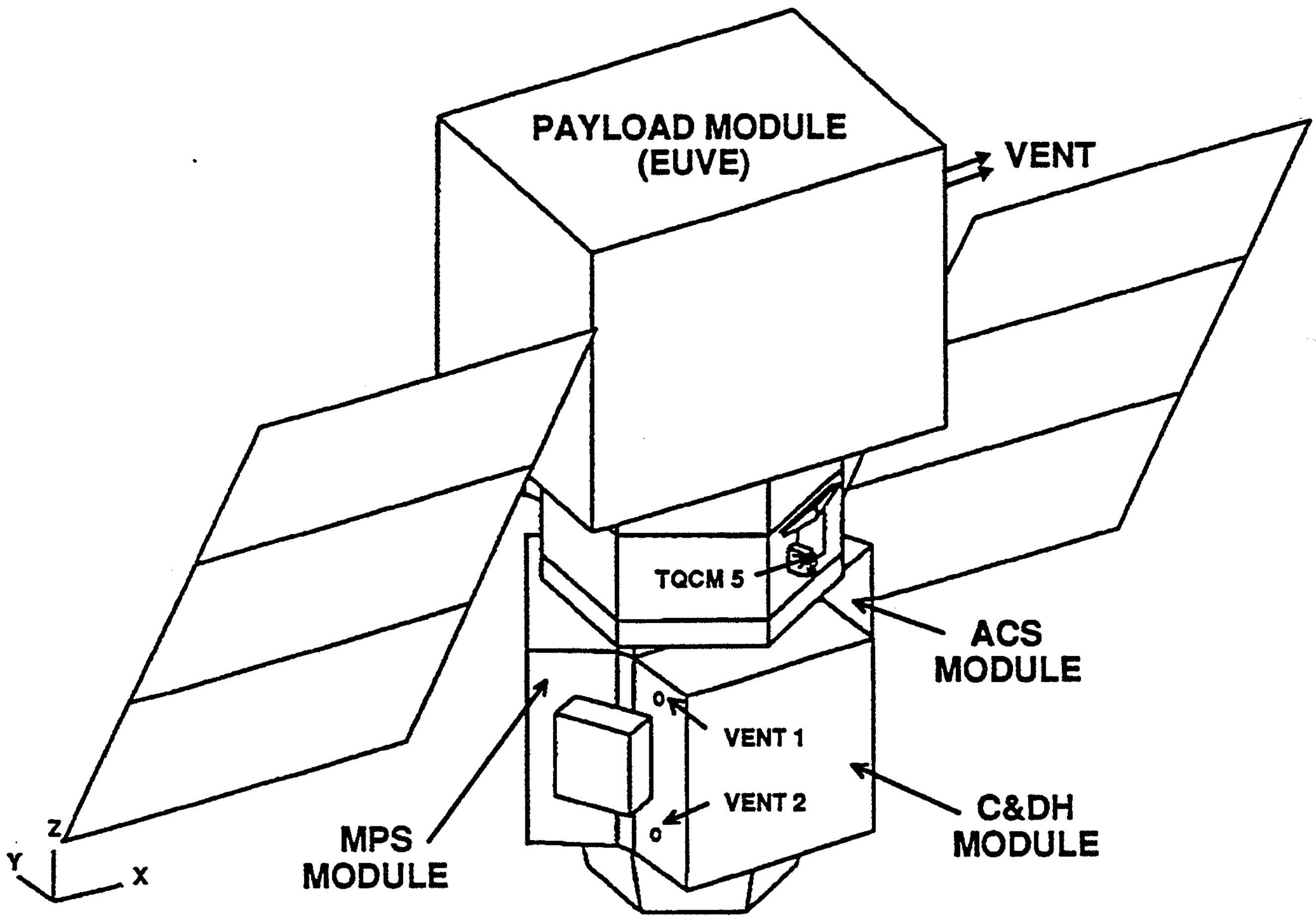
EXPLORER PLATFORM WITH EUVE INSTALLED



LOCATIONS OF EVEEP TQCM'S AND SPACECRAFT VENTS

GSFC

EXPLORER PLATFORM WITH EUVE INSTALLED



LOCATIONS OF EVEEP TQCM'S AND SPACECRAFT VENTS

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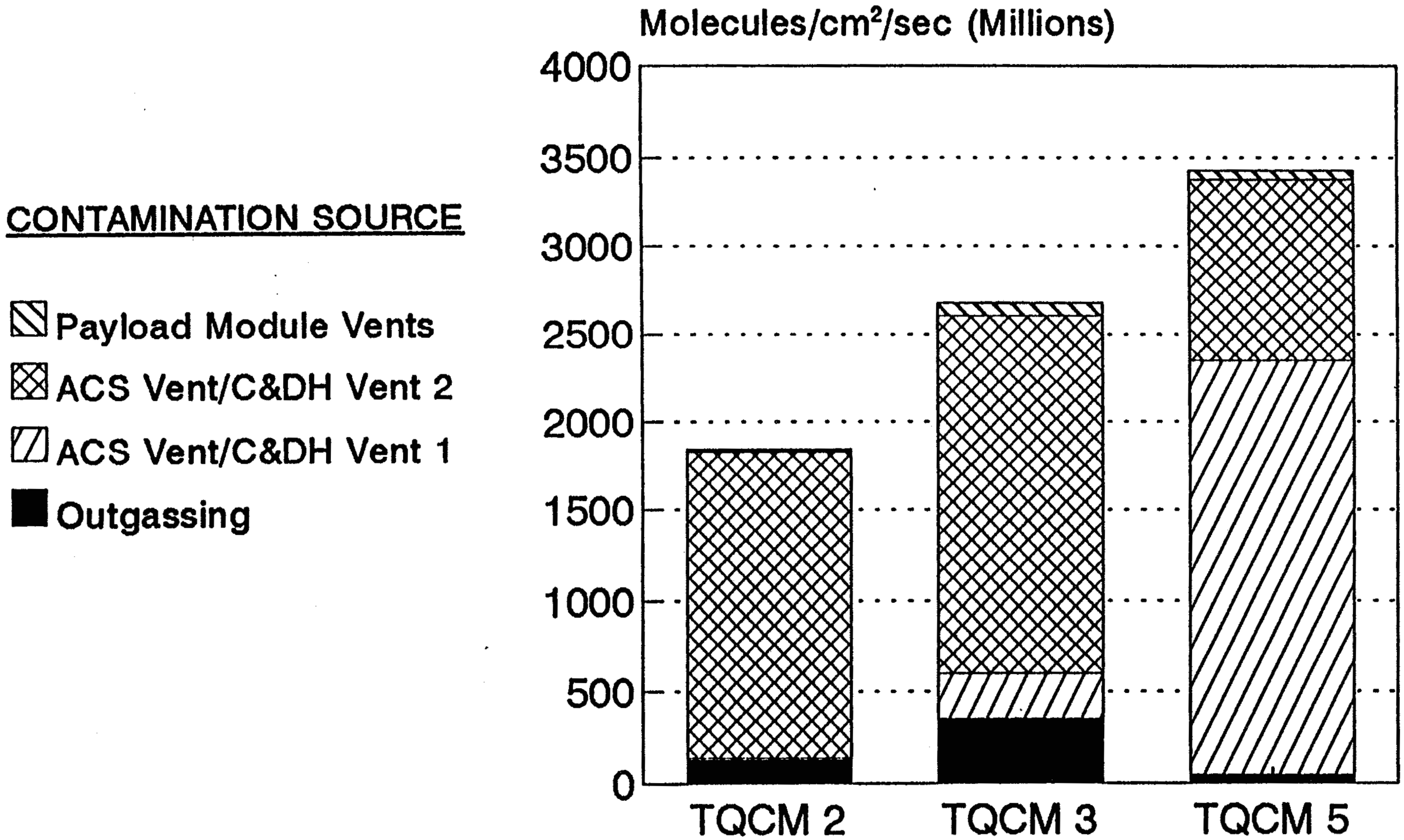
EVEEP PREDICTED OUTGASSING RATES

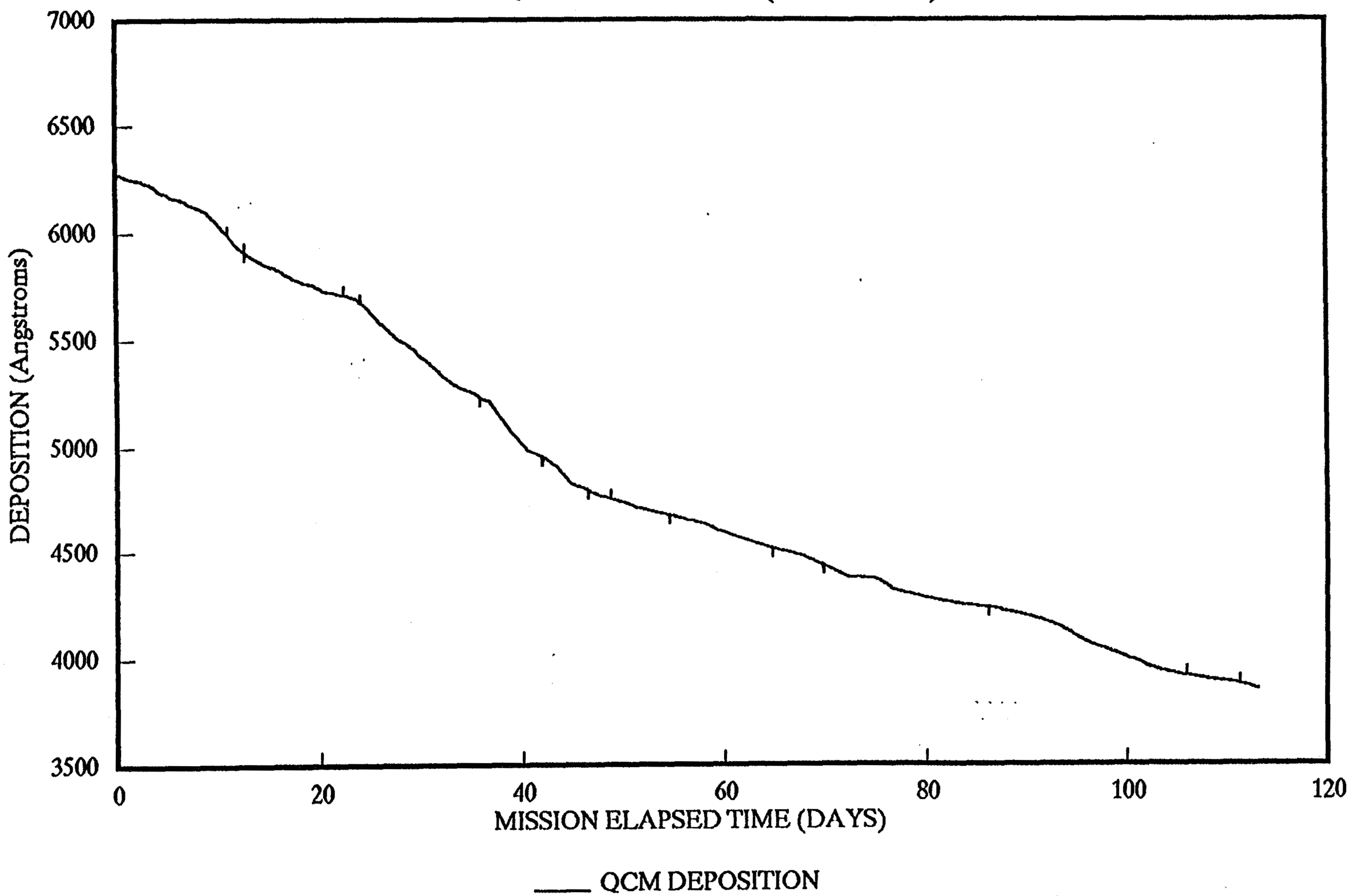
Normalized Flux of Organic Species



TQCM = Temperature Controlled Quartz Crystal Microbalance

EVEEP DATA CHARTS

QCM #1 DEPOSITION (DAY 1 – 113)



Teflon coated QCM

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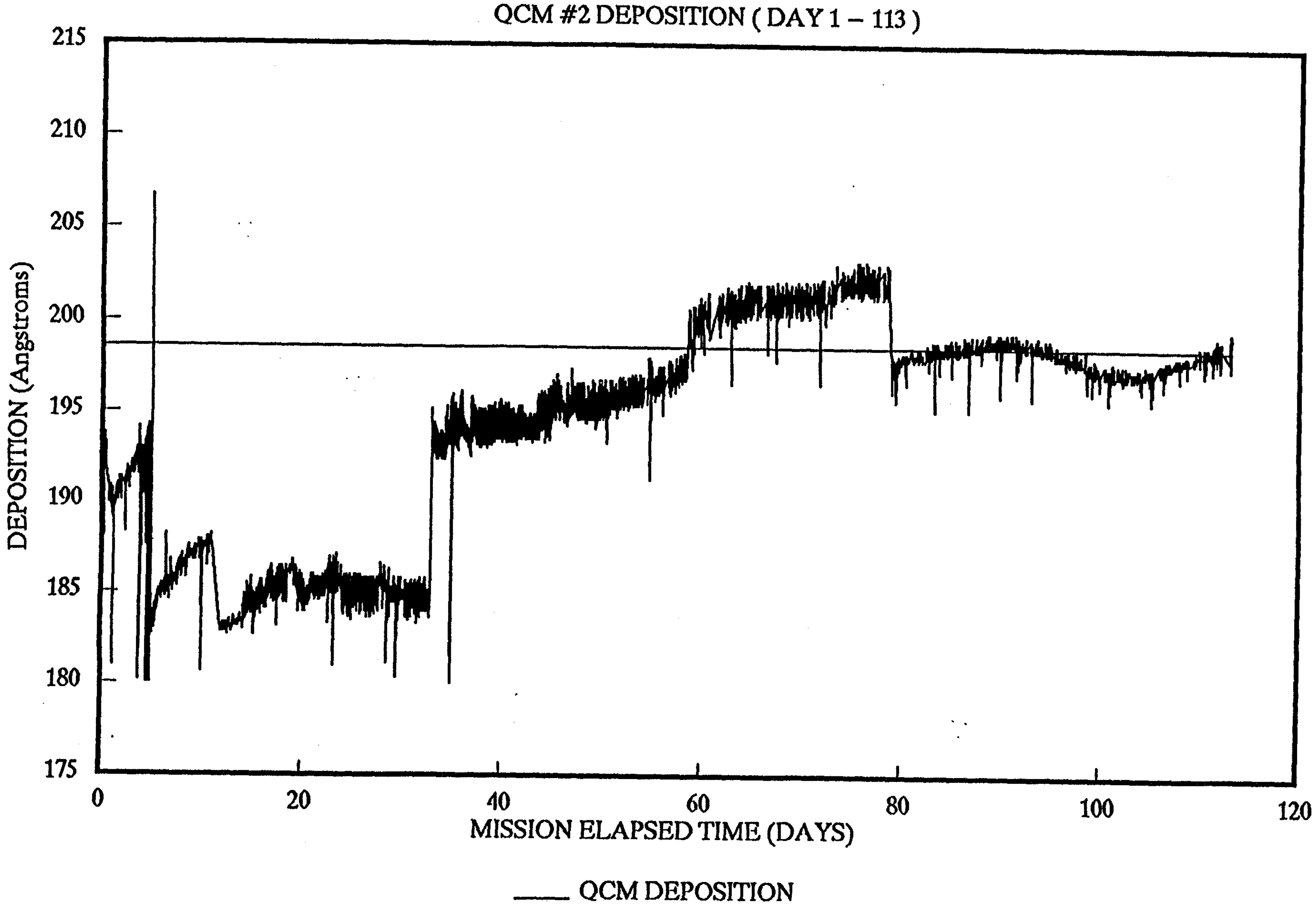
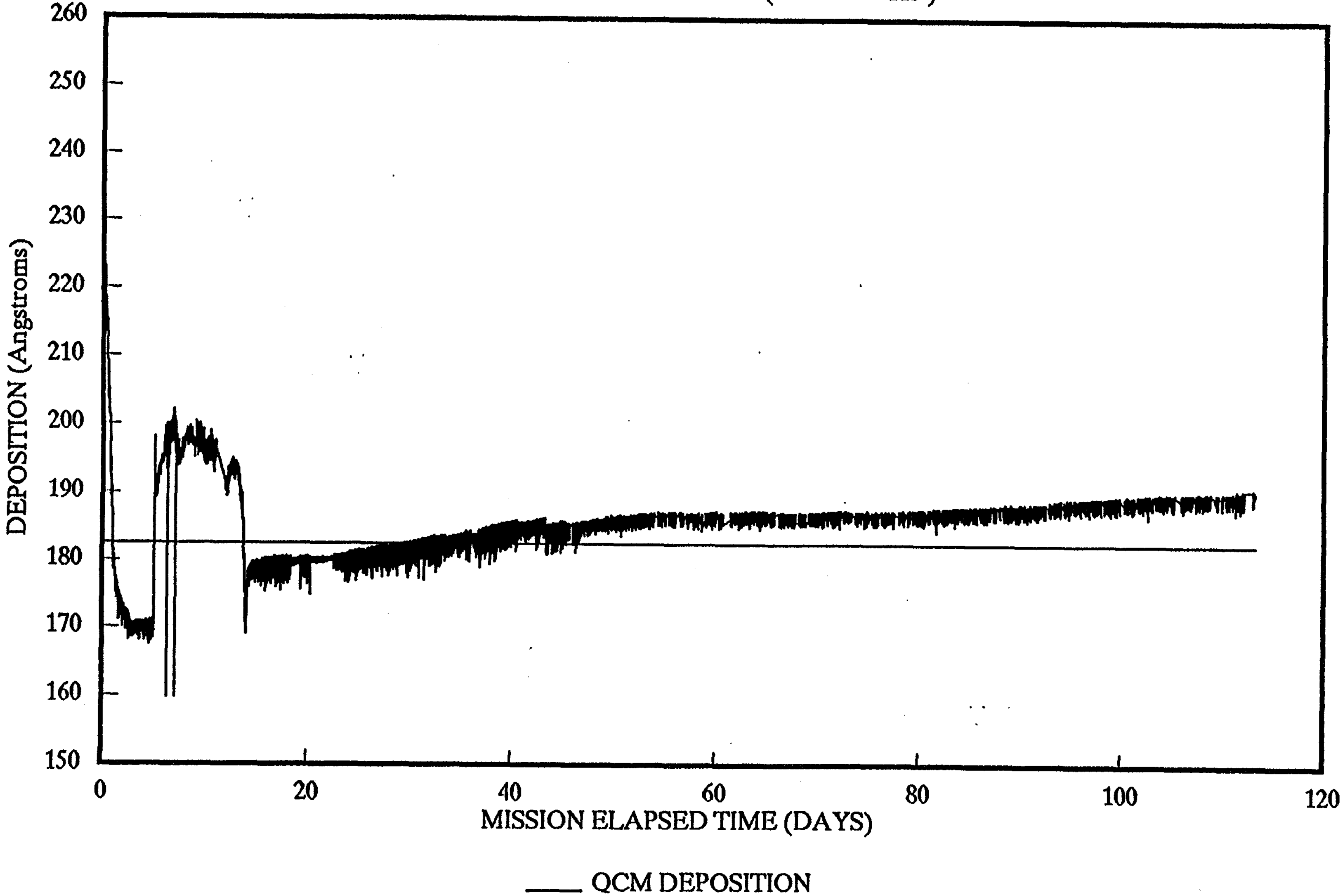
EVEEP DATA CHARTS



Non–Teflon coated QCM

EVEEP DATA CHARTS

QCM #3 DEPOSITION (DAY 1 – 113)

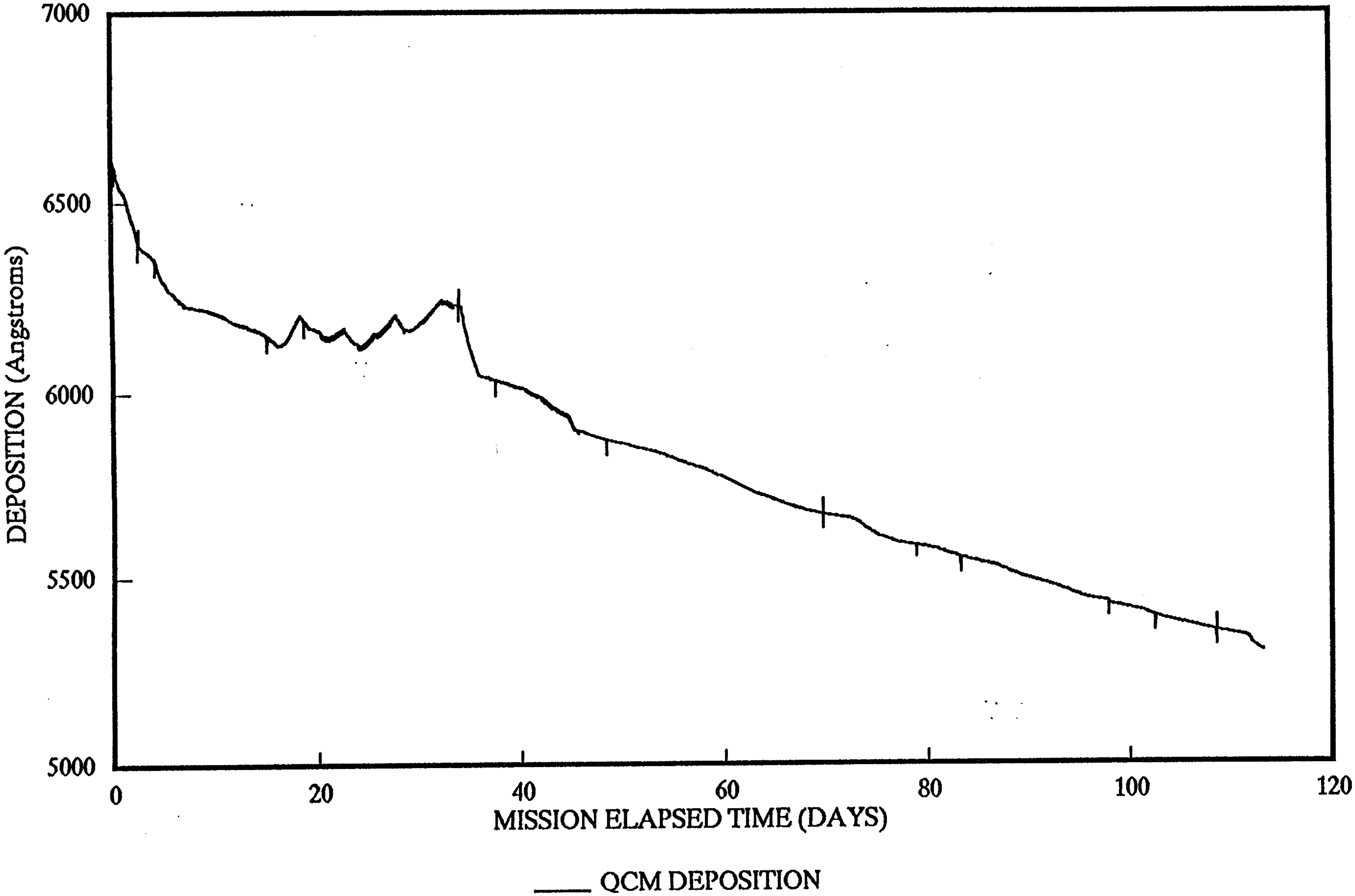


Non–Teflon coated QCM

GSFC

EVEEP DATA CHARTS

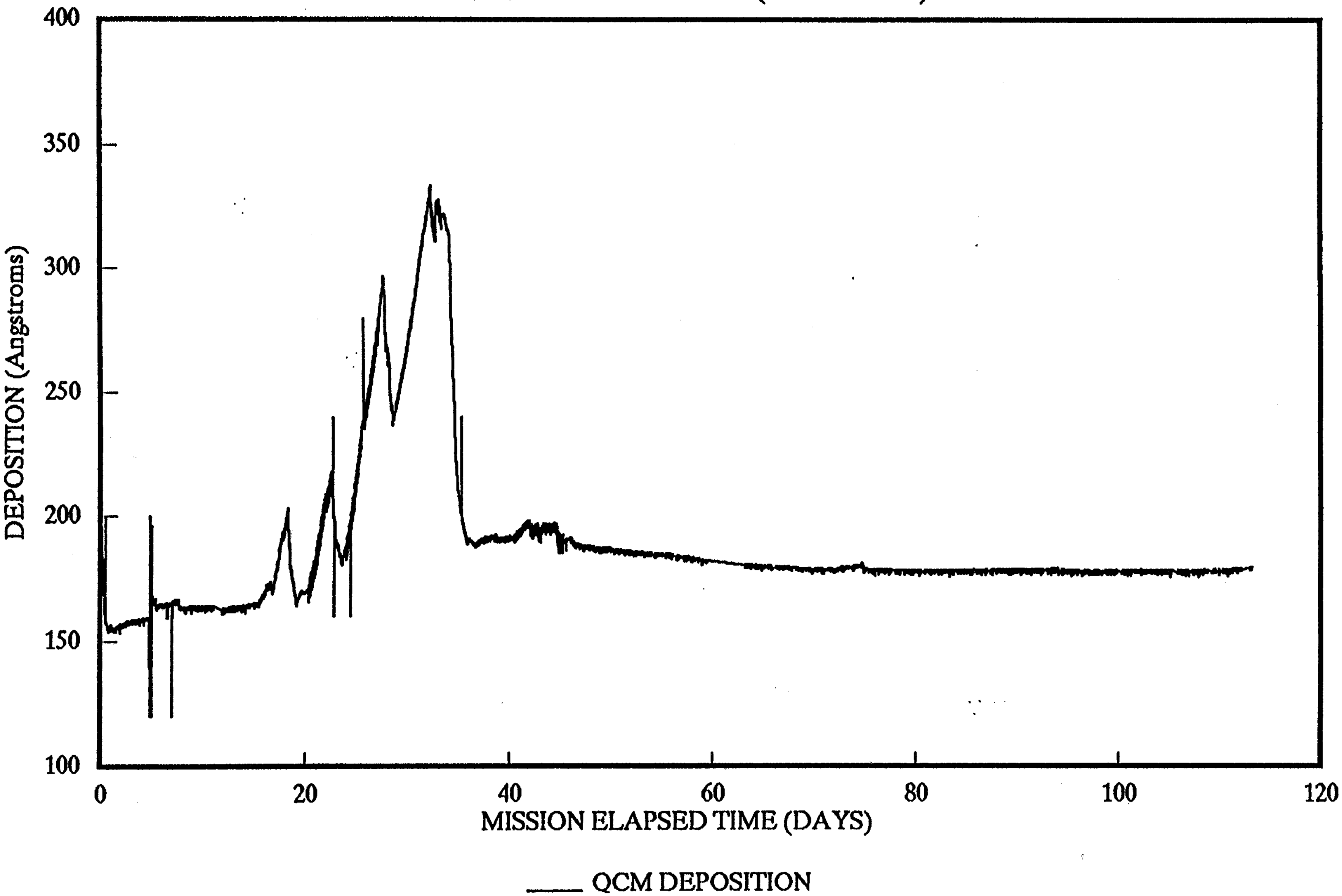
QCM #4 DEPOSITION (DAY 1 – 113)



Teflon coated QCM

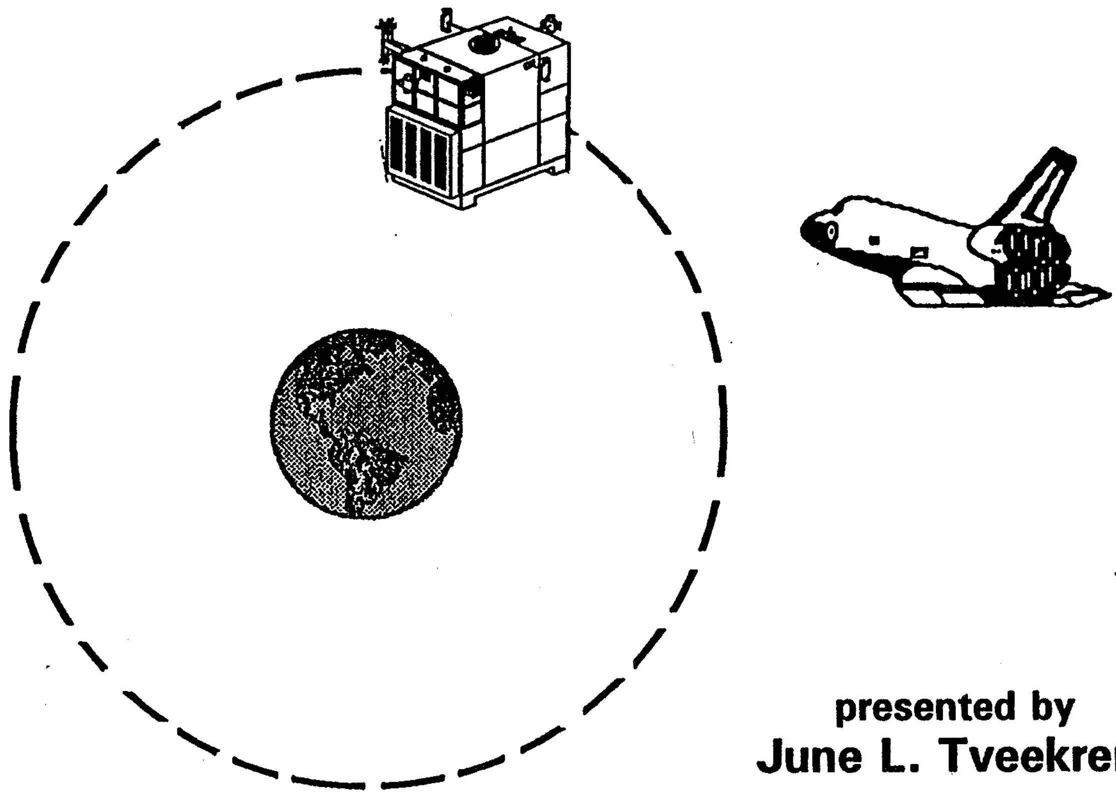
EVEEP DATA CHARTS

QCM #5 DEPOSITION (DAY 1 – 113)



Non–Teflon coated QCM

RETURN FLUX EXPERIMENT



presented by
June L. Tveekrem

Flight Experiments
Tech. Interchange Mtg.

October 7, 1992

BACKGROUND

- All spacecraft emit molecules via outgassing, thruster plumes, vents, etc. Return flux is portion of those molecules that scatter from ambient atmosphere and return to the spacecraft.
- Return flux allows critical spacecraft surfaces to become contaminated even when there is no direct line of sight between the contamination source and the critical surface. LDEF data shows that contamination on LDEF surfaces could not have come entirely from direct flux – suggests significant return flux.
- Several computer models simulate return flux, but predictions have never been verified in orbit. Large uncertainties in predictions lead to overly conservative spacecraft design.
- Purpose of REFLEX is to fly a controlled experiment that can be directly compared with predictions from several models.

REFLEX OBJECTIVES

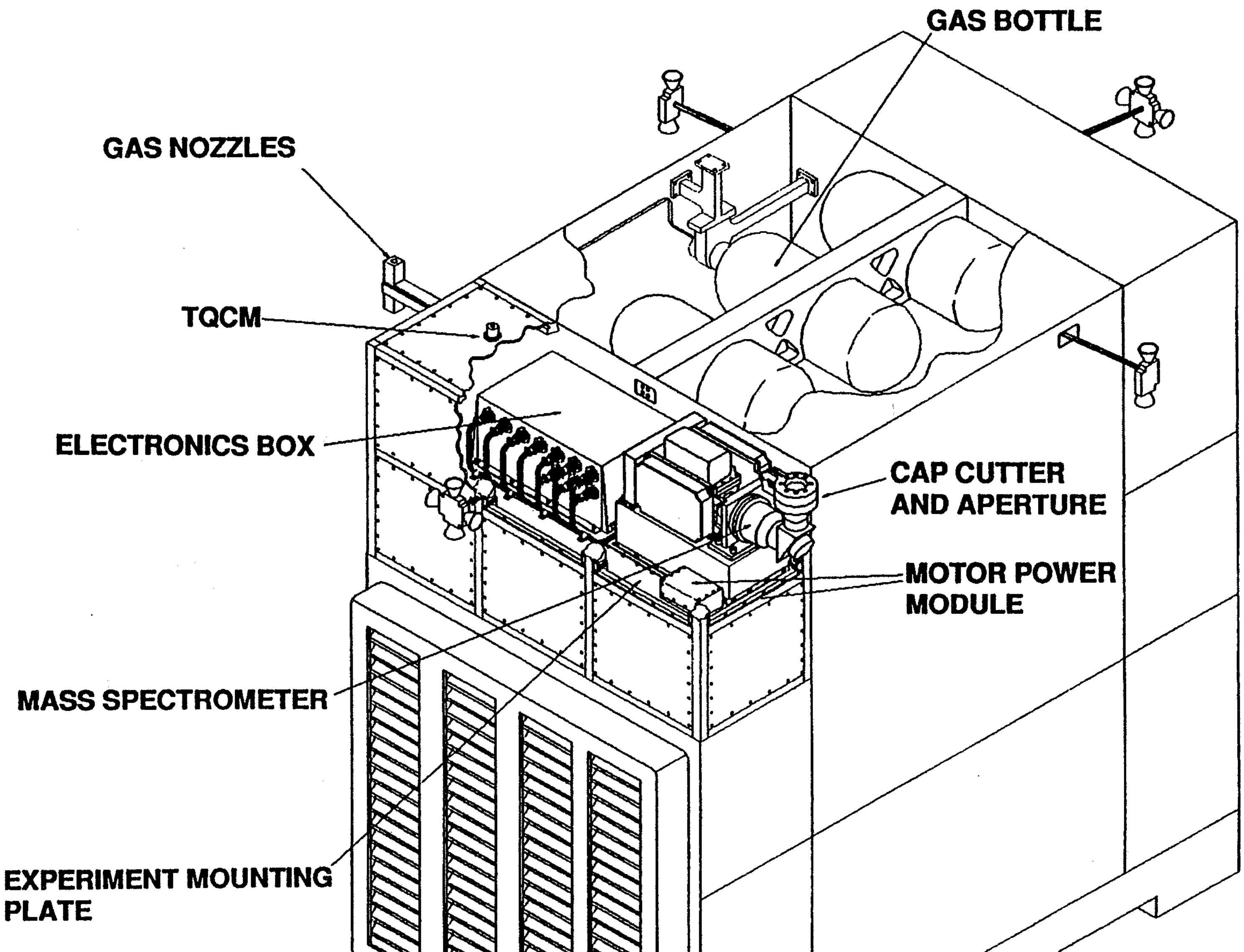
1) Measure the velocity, species, and density of the return flux molecules from an onboard noble gas source

2) Characterize the ambient environment by measuring the velocity, species, and density of the ambient molecules

3) Utilize REFLEX flight data for validating and updating available mass transport contamination models

4) Measure the erosion rate of sample coatings due to reaction with ambient atomic oxygen

REFLEX AND SPARTAN SPACECRAFT



REFLEX FEATURES

- Inert, monatomic gases: simplifies modeling, avoids chemical reactions
- Mixture of two gases: allows measurement of cross-section dependence
- Nozzle shape is sonic orifice: easily machined, minimizes backflow and creep
- Two spacecraft attitudes (into ram and 90° to ram): allows limited measurement of angular dependence
- Mass spectrometer with energy analyzer: gives positive identification of scattered species, measures energy distribution

REFLEX SYSTEM

SYSTEMS
Seabrook, Maryland

- Weight: 315 lb.
- Volume: 8.2 ft^3
- Power: 2000 Watt·hrs
- Duration: 21 hours
- Spacecraft: Spartan, shared with 1 or 2 other expts.
- Launch vcl: Shuttle
- Launch date: April 1995 (STS-72)

EXPERIMENTAL SCENARIO

NASA

ORBIT NUMBER	1 - 2	3 - 4	5 - 6	7 - 8	9 - 10	11 - 12	13 - 14
SPACECRAFT ATTITUDE*	RAM	RAM	RAM	RAM	90°	90°	90°
GAS NOZZLE SETTING	OFF	FLOW RATE OF 0.07 g/s	FLOW RATE OF 0.2 g/s	OFF	FLOW RATE OF 0.07 g/s	FLOW RATE OF 0.2 g/s	OFF

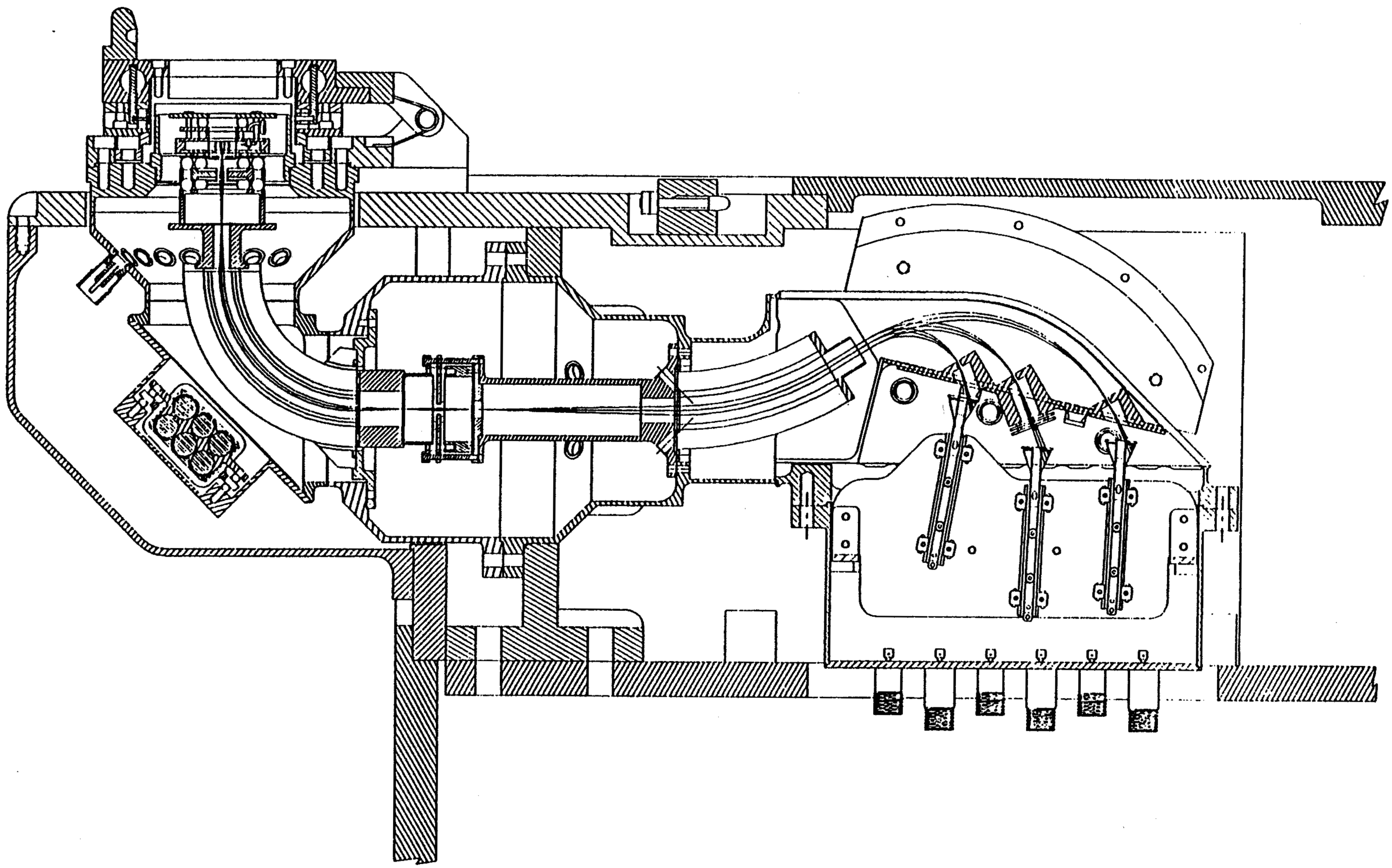
*"RAM"= Mass spectrometer aperture and gas nozzle pointing into velocity vector
"90°" = Mass spectrometer and gas nozzle pointed perpendicular to velocity vector, with nozzle "upstream" of mass spectrometer

MASS SPECTROMETER

EER SYSTEMS
Seabrook, Maryland

- Major components are energy analyzer, ion source, mass analyzer, detector.
- Energy analyzer distinguishes between ambient molecules (velocity 8 km/sec), return flux molecules (1 to 8 km/sec), and thermalized molecules (<1 km/sec).
- Ion source ionizes molecules by electron impact.
- Mass analyzer is Mattauch-Herzog geometry, double focusing, with permanent magnet.
- Two types of detector used: Counting multipliers for current $\approx 10^{-14}$ A or less, electrometer for current $> 10^{-14}$ A.
- Range of mass spectrometer is 4 to 180 amu in mass, nine orders of magnitude in counts.

REFLEX MASS SPECTROMETER SCHEMATIC



DATA ANALYSIS

Comparison With MOLFLUX Model

- MOLFLUX Does Not Calculate Molecular Velocity (Energy).
- Therefore, Must Integrate Out Energy Dependence of Mass Spectrometer Data.
- Compare Observed Total Fluxes of Neon and Krypton to Model Predictions.

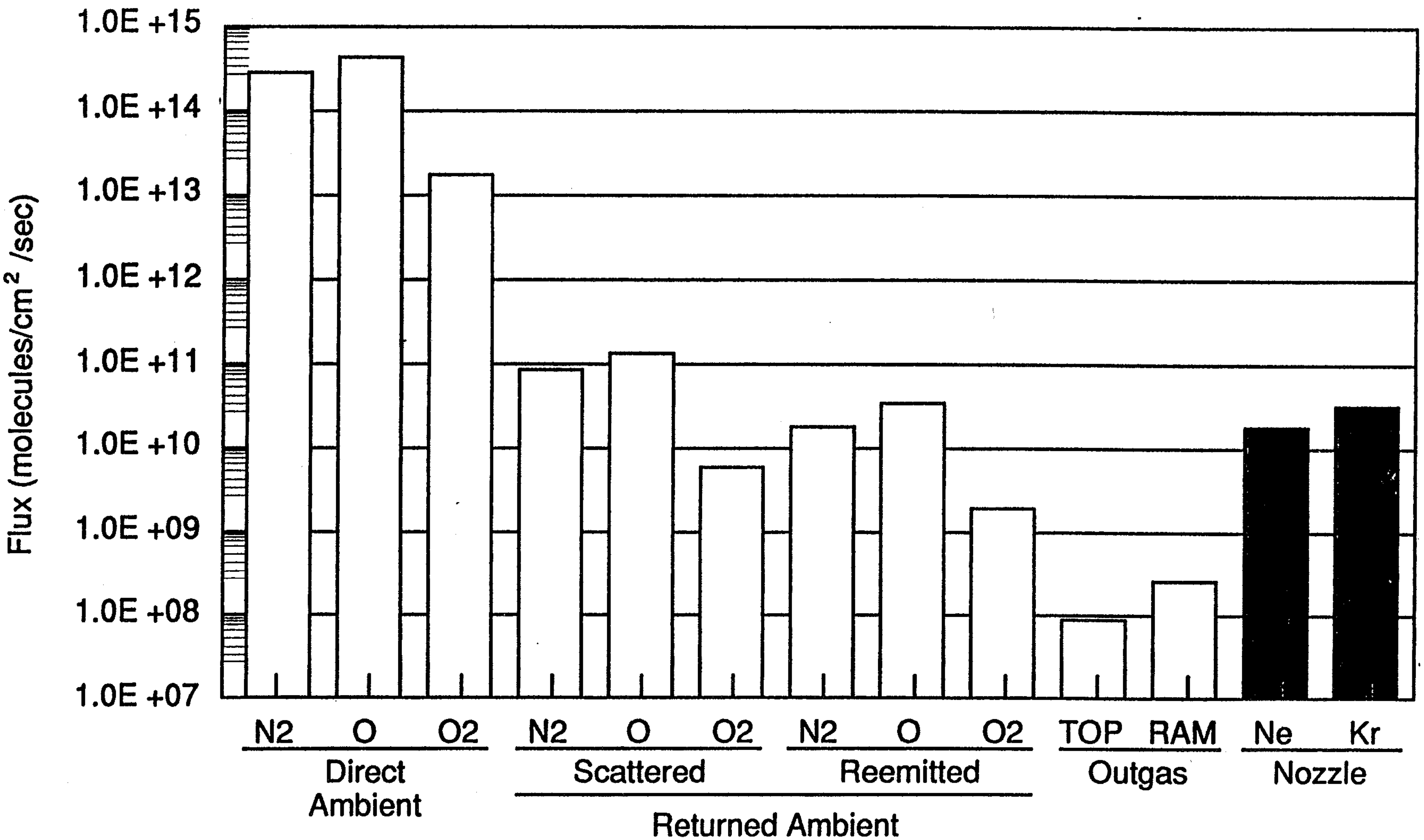
Comparison With ISEM Model

- ISEM Predicts Flux Versus Species, Like MOLFLUX, so Can Make the Same Comparison as for MOLFLUX.

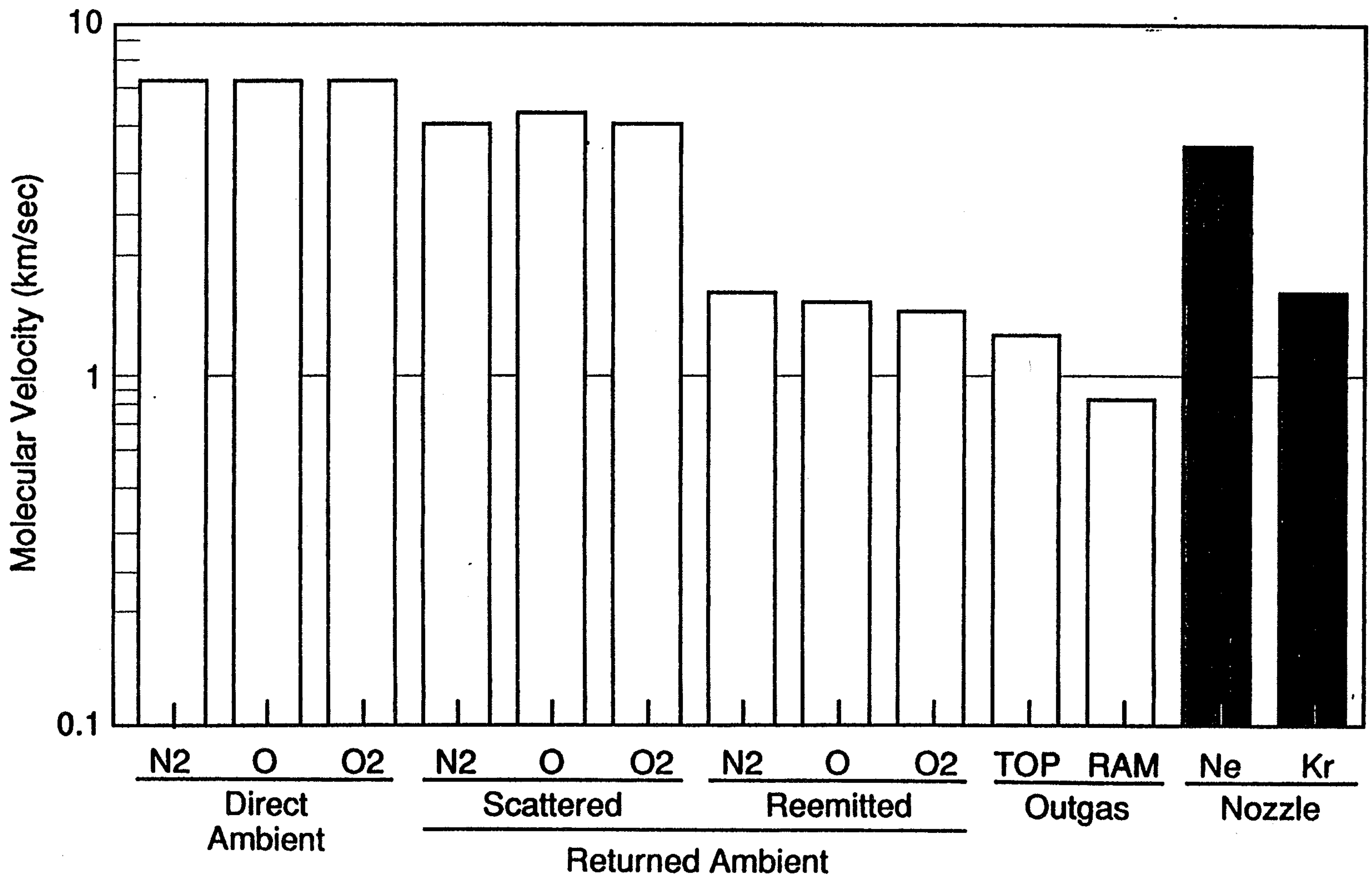
- Currently, ISEM Also Predicts *Average* Molecular Velocity for Each Species. Plan to Average the Mass Spectrometer Energy Data to Compare With This Prediction.

- In Process of Modifying ISEM to Give Velocity *Distribution*, i.e., Flux Versus Velocity, for Each Species. Then Can Compare Model Predictions to Mass Spectrometer Output.

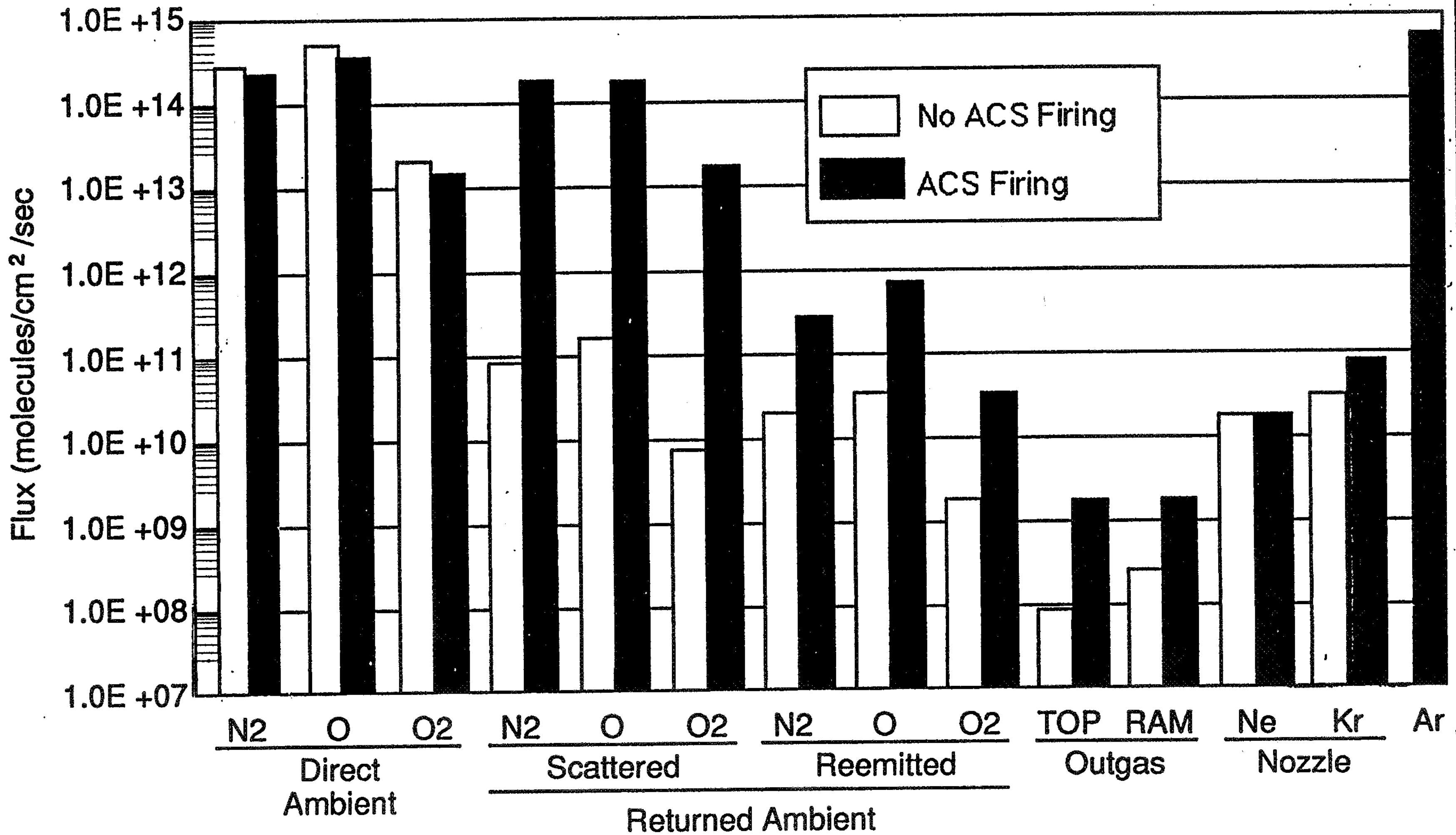
COMPOSITION OF MOLECULAR FLUX INCIDENT ON REFLEX SPECTROMETER



MOLECULAR VELOCITY DISTRIBUTION



IMPACT OF ACS FIRINGS ON INCIDENT FLUX OF REFLEX SPECTROMETER



SUMMARY

- REFLEX is a controlled experiment to quantify the return flux effect.
- Return flux is measured by releasing a Neon/Krypton gas mixture at a known rate and using a mass spectrometer to detect molecules which "return" to the spacecraft.
- Preliminary modeling has shown that there is enough return flux signal, that the Spartan ACS will not interfere with the measurements, and that it will be possible to directly compare model predictions to the flight data.
- REFLEX will fly on a Spartan with 1 or 2 other (TBD) experiments.
- Launch is scheduled for April 1995.

N93-28728

SPACECRAFT GLOW AND THE EISG/SKIRT EXPERIMENT

p- 41

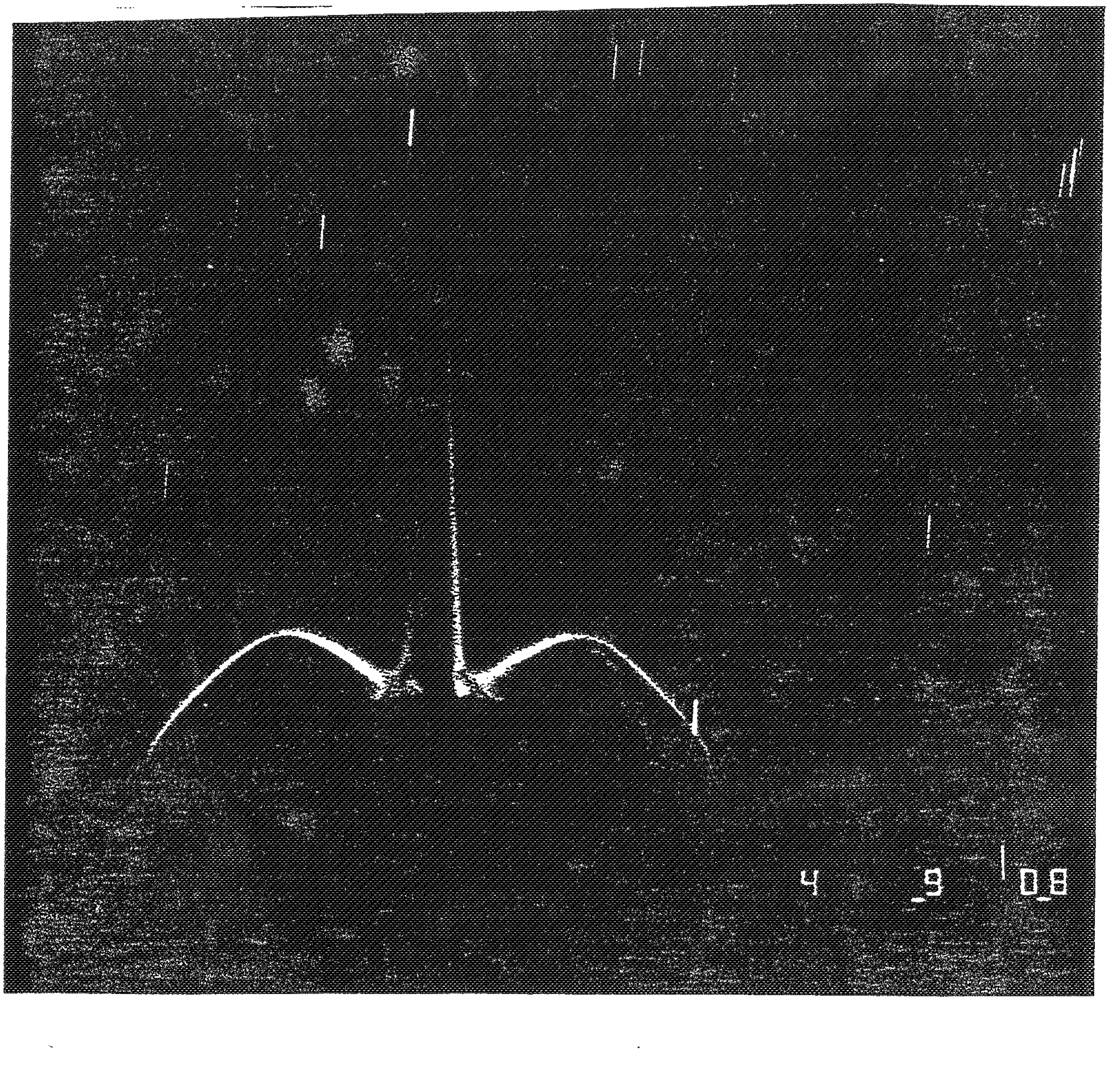
Gary R. Swenson
Lockheed R&D

Mark Ahmadjian
US Air Force Phillips Laboratory

Don Jennings
NASA/Goddard Space Flight Center

Jim Visentine
NASA/Johnson Space Center

STIG Meeting
October 7, 1992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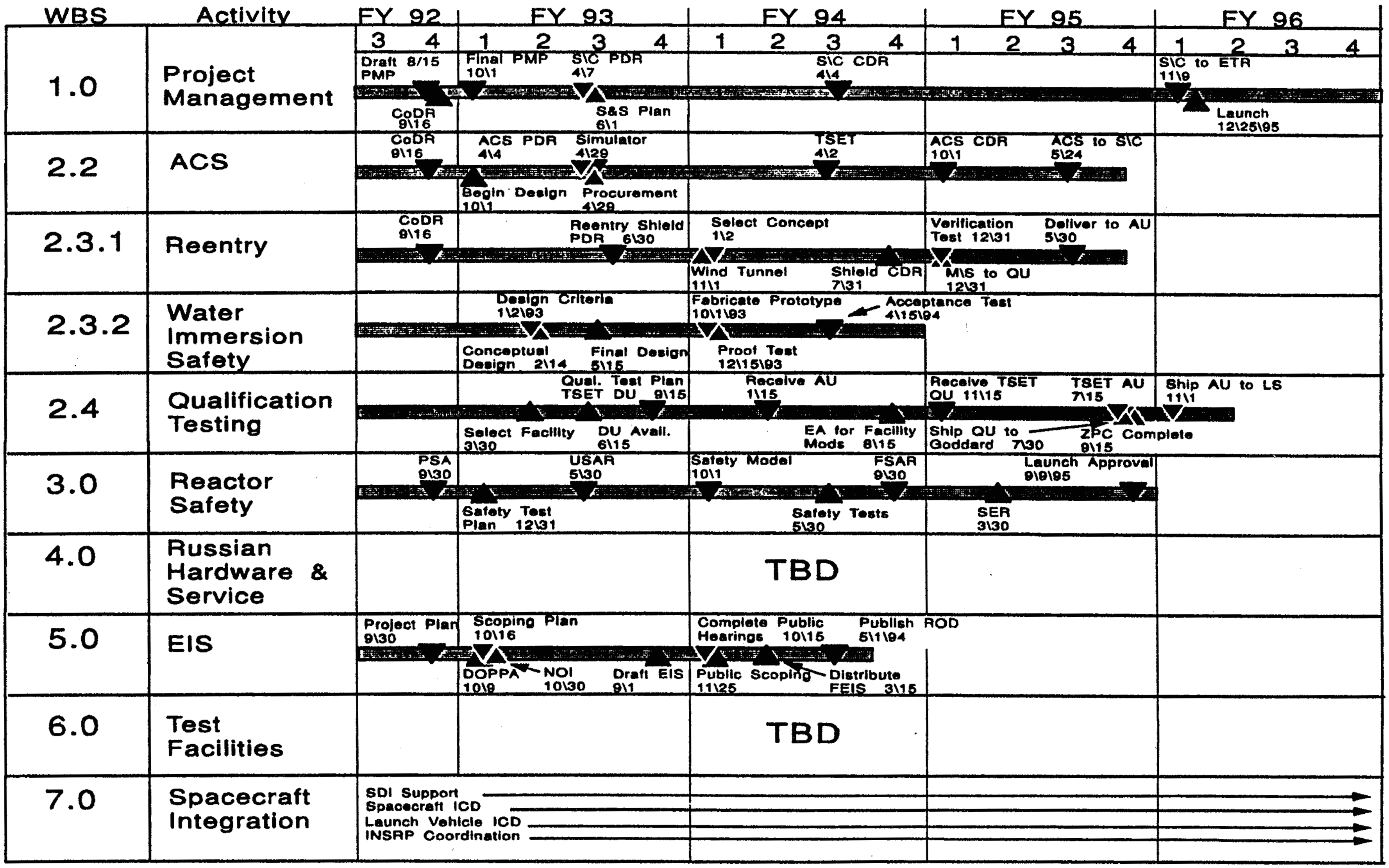
SESSION 7:

SPACE ENVIRONMENTS AND EFFECTS

Co-Chaired by:
Mr. Joseph C. Kolecki, NASA Lewis Research Center
Dr. Donald E. Hunton, Hanscom AFB

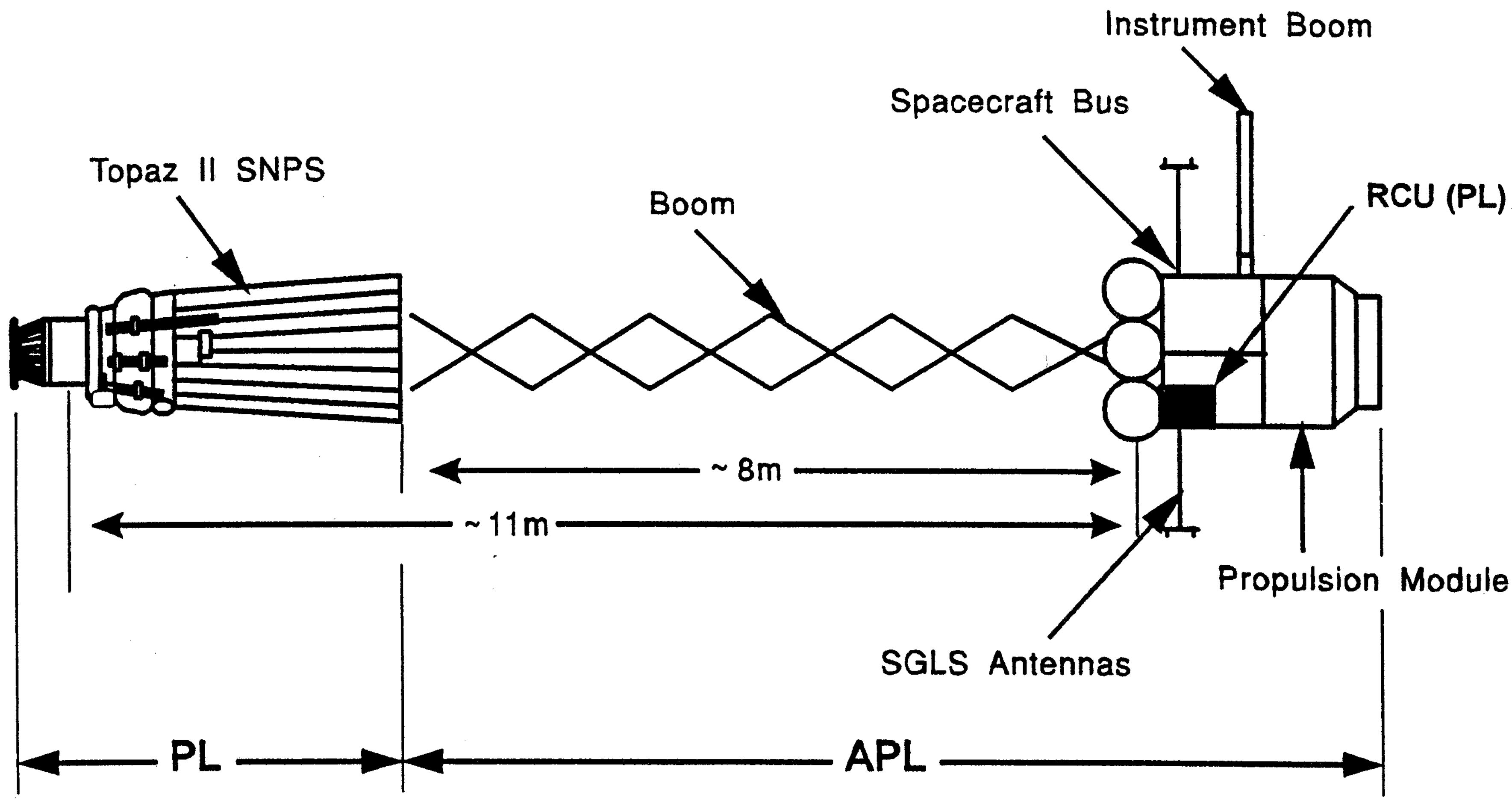
Preliminary Topaz II Reactor Program Schedule



Mission Profile

- Launch into Low Earth Orbit (LEO) (nominal orbit 1,600 km @ 28.5°)
- Confirm nuclear safe orbit (by ground skin track radar)
- Start up TOPAZ 2 reactor (by ground command)
- Evaluate performance of reactor and spacecraft bus
- Begin electric propulsion orbit raising; spiral out with thrust vector parallel to velocity vector
- Test each type of engine for 1,000 hours
- Duty-cycle thrusters to separate reactor measurements from thruster measurements; ~ 97% duty cycle
- Perform life testing on selected engine types; use high thrust engines first, low thrust engines later
- Above 6.6 R_E, begin twice per orbit yaw maneuver to change inclination, simultaneously raising orbit

NEP Satellite Configuration



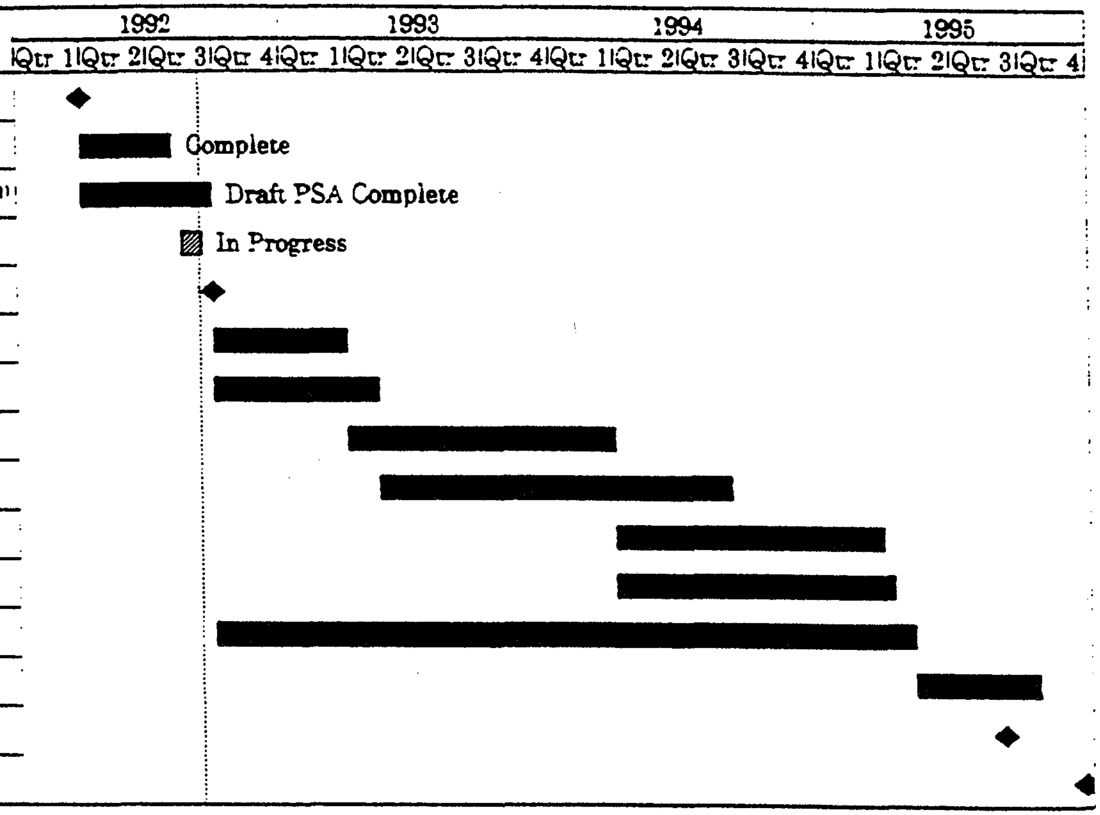
Major Milestones For The NEP Space Test Program

✓August 92	Satellite Conceptual Design Review
✓September 92	Reactor Conceptual Design Review
✓September 92	Topaz II Preliminary Safety Assessment Complete
October 92	Full Funded Start
April 93	Satellite Preliminary Design Review
May 93?	Updated Safety Analysis Report Complete
April 94	Satellite Critical Design Review
September 94	Final Safety Analysis Report Complete
April 95	Propulsion Module Complete
May 95	Instrument Module Complete
May 95	Reactor Modifications Complete
September 95	Topaz II Modifications Complete
September 95?	Launch Approval Received
December 95	Launch

NEP Space Test Program Schedule

TOПA3
TOPAZ

ID	Name
1	Program Start
2	Perform Satellite Conceptual Design
3	Perform Topaz Preliminary Safety Assessment
4	Perform Reactor Conceptual Design
5	Full Funded Program Start
6	Perform Satellite Preliminary Design
7	Develop Updated Safety Analysis Report
8	Perform Satellite Critical Design
9	Develop Final Safety Analysis Report
10	Construct Porpulsion Module
11	Construct Instrument Module
12	Modify Topaz Reactor
13	Integrate And Test Reactor & Satellite
14	Launch Approval Received
15	LAUNCH



NEP Space Test Program Fact Sheet

Summary

The NEP Space Test Program is sponsored by SDIO. The objective of the program is to launch an NEP satellite powered by a Russian Topaz II reactor by December 1995 for a cost of $150M, excluding the cost of the launch vehicle costs. The cost distribution is $80M for the satellite and $70M for the power system. The Applied Physics Lab (APL) is responsible for the satellite and the Phillips Lab (PL) is responsible for the power system.

Key Terms and Acronyms

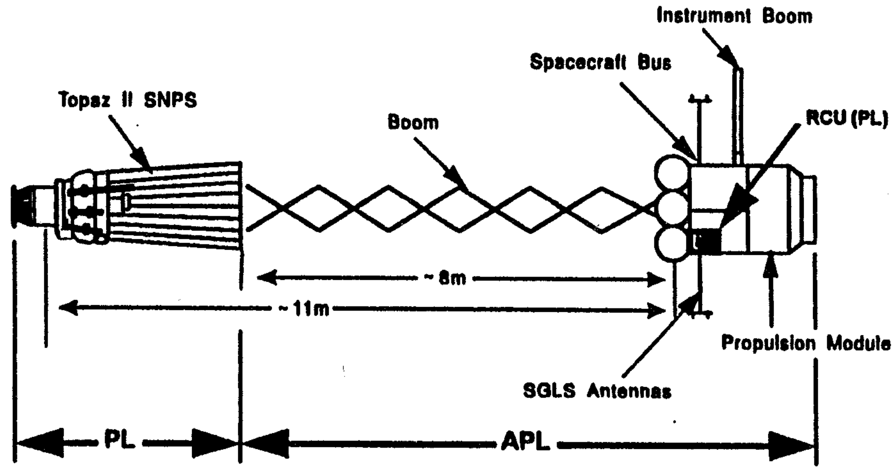
Topaz II: 6 kWe, Russian SNPS based on thermionic conversion
NEP: Nuclear Electric Propulsion
SNPS: Space Nuclear Power System
SDIO: Strategic Defense Initiative Organization
PL: Phillips Lab
APL: Applied Physics Lab
SNL: Sandia National Laboratories
LANL: Los Alamos National Laboratories
CDBMB: Central Design Bureau for Machine Building (St. Petersburg,)
KIAE: Kurchatov Institute of Atomic Energy (Moscow)
CoDR: Conceptual Design Review
PDR: Preliminary Design Review
CDR: Critical Design Review
PSA: Preliminary Safety Assessment (of the Topaz II)

Status

The Topaz Program was begun in April 1992. To date, the following milestones have been reached:

- Satellite CoDR (APL)
- Reactor CoDR (PL)
- Draft Topaz PSA (PL)

A program plan has been established for reaching the objective of launching the NEP satellite by December 1995.



Topaz Satellite Configuration

Topaz Modifications

Four modifications will have to be made to the Topaz II power system:

- A reentry shield will be added to ensure the system meets the safety requirement of an essentially intact reentry (SNL),
- A neutron poison may be added to the core to prevent criticality in accident scenarios involving water immersion and flooding (LANL),
- A new reactor control system will be added because the Russian system is not space qualified and does not meet US safety standards (PL, SNL), and
- US fuel may be used because of difficulties associated with obtaining special nuclear material from Russia (LANL).

NEP Space Test Program Goals

Primary Goals

- Demonstrate The Feasibility Of Launching A Space Nuclear Power System In The United States
- Demonstrate An Orbit Adjust Capability Using Nuclear Electric Propulsion
- Evaluate The In-Orbit Performance Of The Topaz II Reactor And Selected Electric Thrusters
- Measure, Analyze, And Model The NEP Self-Induced Environment

Secondary Goal

- Conduct A Space Science/Engineering Mission Compatible With The Primary Mission Requirements

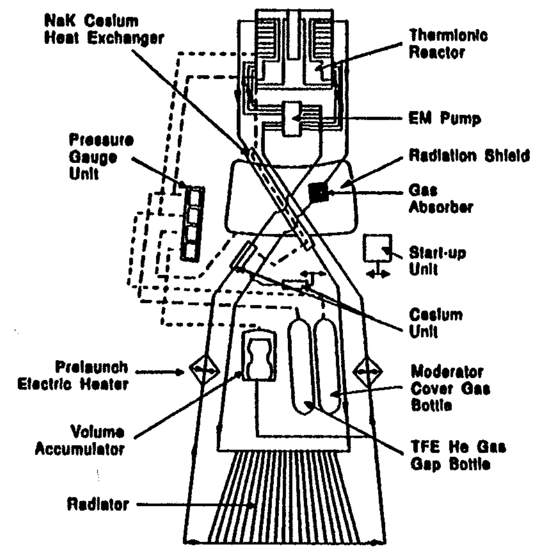
Topaz II Description Fact Sheet

Summary

The Topaz II is a Russian built thermionic space nuclear reactor. It is a single-cell design that has several advantages over multi-cell designs including: it permits full system qualification testing using electric heaters in place of nuclear fuel, it doesn't require early commitment of expensive enriched uranium (96% U-235), it allows removal of fuel for shipping and storage which improves the safety and safeguards environment, and the open cavity of the single-cell design facilitates fission gas escape (mitigating fuel swelling). The reactor is an epithermal system. The fuel loading is low (<27 kg). The output of the reactor is 28 VDC when operating at 135 kWth and 6 kWe. The reactor is cooled by flowing liquid NaK. The coolant loop is susceptible to single-point failures.

Key Terms and Acronyms

Thermionic: The physical process whereby heat energy is converted directly into electric energy via the mechanism of electron emission from an emitter to a collector.
TFE: Thermionic Fuel Element. The structure which contains the components required to produce and utilize the thermionic conversion process.
TSET: Thermionic System Evaluation Test. The facility used to perform full scale Topaz II system tests using electric heaters in place of nuclear fuel in the TFEs.
Single-cell/Multi-cell: Differing designs of the thermionic fuel elements where the TFE consists of a single energy converter versus several energy converters connected in series much as batteries stacked in a flashlight.
NaK: Eutectic composition of sodium and potassium metals which is used as the primary coolant in the Topaz II reactor.
Moderator: ZrH blocks that surround the TFEs and reduce neutron energies down to thermal equilibrium levels.
Reflector: Beryllium blocks above and below the moderator, and segments and drums which surround the reactor vessel that "reflect" neutrons back into the core region.
Core: The central region of the reactor consisting of the fuel, TFEs and moderator where the peak neutron flux exists.
Radiator: A skirt consisting of small pipes, welded to copper plates, carrying the NaK coolant which radiates the excess heat from the coolant to space.
Radiation Shield: Lithium hydride filled stainless steel casing that is used to reduce the level of neutron and gamma ray radiation intensity in the direction along the boom.



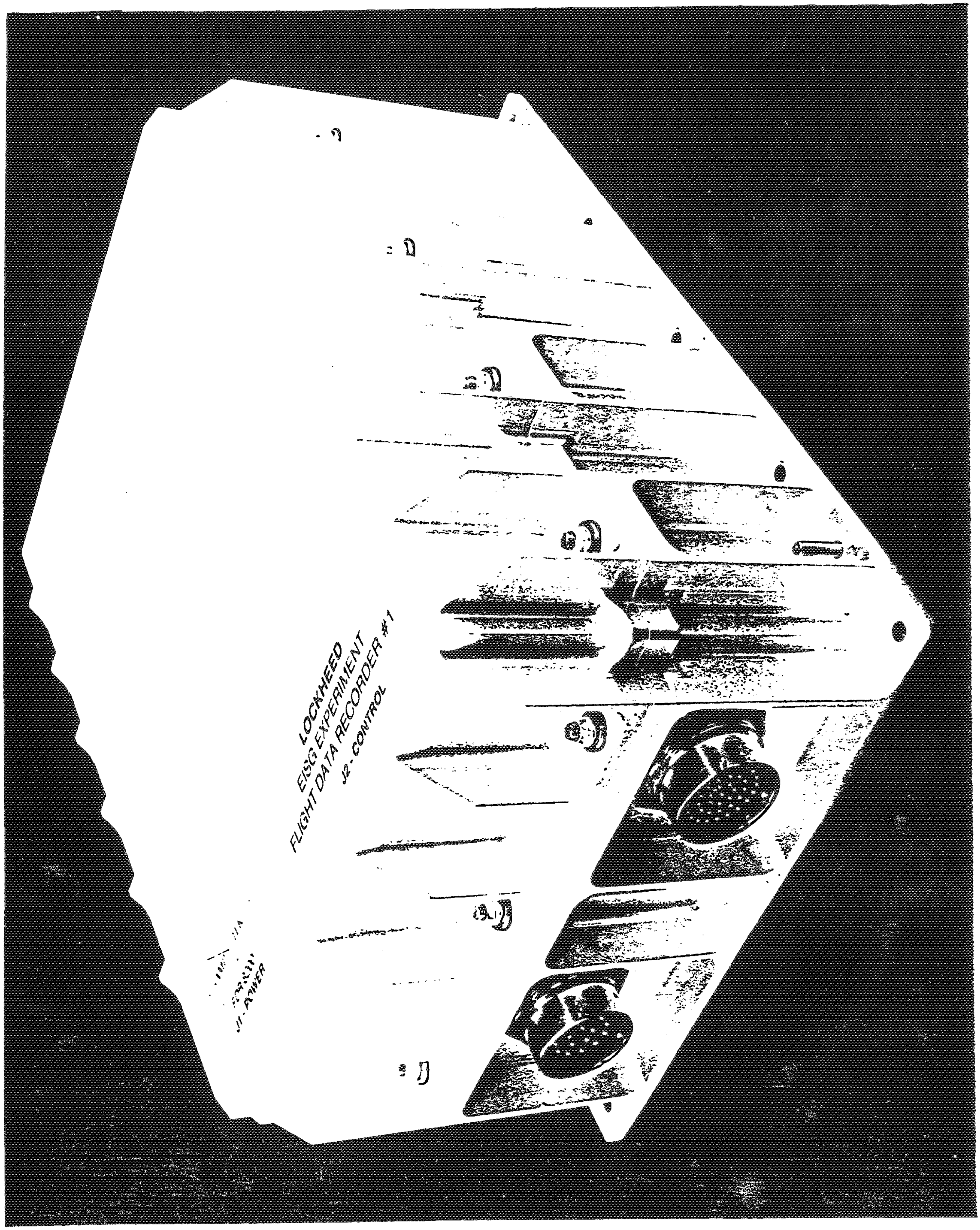
Topaz II System Diagram.

Key Facts

The Topaz II system consists of three principal components: the reactor, the radiation shield and the radiator/frame structure. Additionally, there are four fluidic systems used to support the Topaz operation: NaK coolant, cesium in the TFE interelectrode gap, and two bottled gas systems.

TFE emitter temperature is ~1600 C during operation and the collector temperature is ~900 C. There are ~11 liters of NaK coolant in the Topaz system at temperatures ranging between 500 C at the reactor inlet and 600 C at the outlet. The cesium system is designed to provide 2 torr of pressure within the TFE interelectrode gap, thus enhancing the thermionic conversion process.

The reactor mass is 290 kg, that of the radiation shield is 390 kg, and the radiator mass is 50 kg (filled with coolant). The frame structure has a mass of 45 kg.



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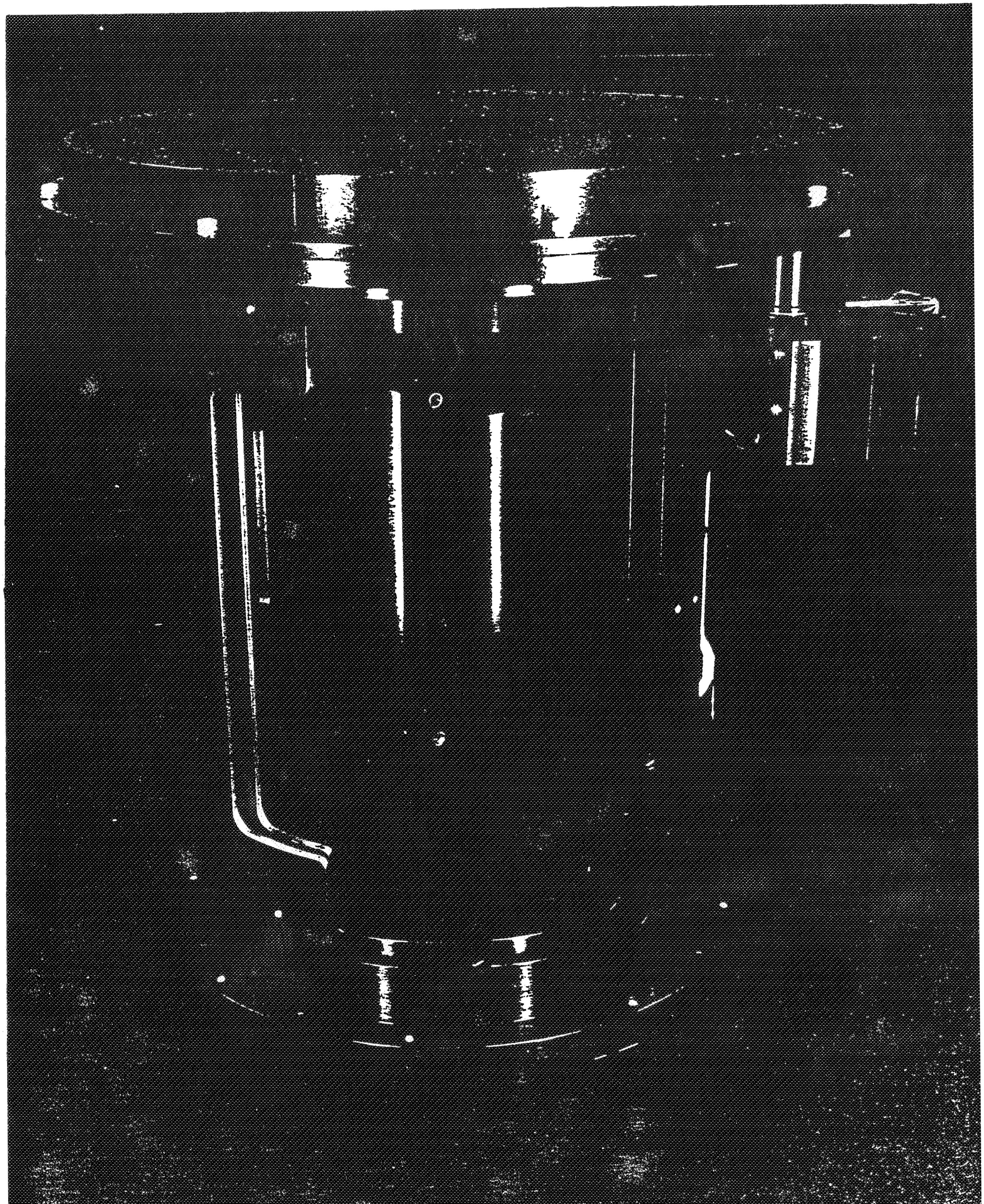
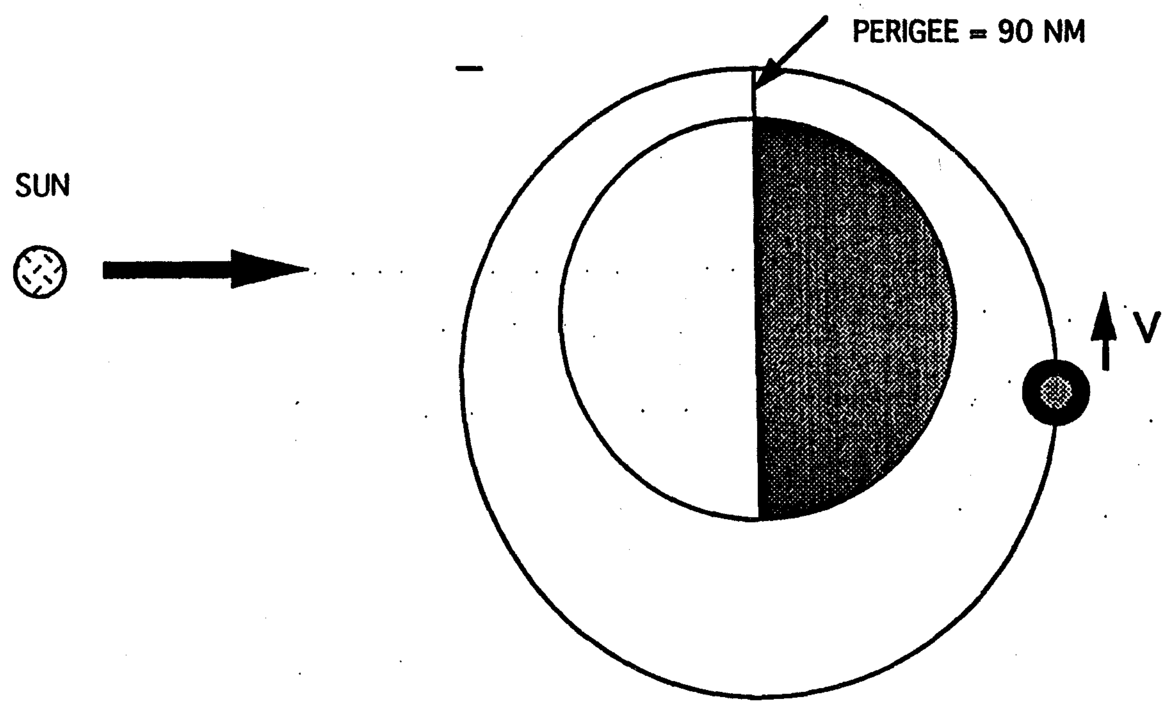


EXPERIMENTAL INVESTIGATION OF SPACECRAFT GLOW

OPERATIONS DURING FOUR ORBITER DARKNESS PERIODS IS DESIRED, AT LOW EARTH ORBIT (LEO). TWO CIRCULAR ORBITS (<140 NM) AND TWO ELLIPTICAL ORBITS (WITH PERIGEE ~90 NM) ARE DESIRED. IT IS REQUESTED THAT FOR THE TWO ELLIPTICAL ORBITS, PERIGEE BE TARGETED FOR THE SUNRISE TERMINATOR. THE INTENT IS TO MAKE MEASUREMENTS ON THE SHADOW PORTION OF THE ORBIT, TO OBSERVE THE EFFECT OF THE ATMOSPHERIC DENSITY CHANGE (WITH ALTITUDE) ON SPACECRAFT GLOW.

ELLIPTICAL ORBIT GEOMETRY



MISSION OPERATIONS (PRIMARY SCIENCE)

ORBITS

-4 SHADOW PERIODS IN LOW EARTH ORBIT (LEO)

-2 ORBITS CIRCULAR
-ALTITUDE <140 NM

-2 ORBITS ELLIPTICAL, PERIGEE 90 NM (OR AS LOW AS POSSIBLE)
-PERIGEE AT ORBITER SUNRISE

-ATTITUDES PRIMARILY WITH BAY TO RAM WITH THERMAL CONDITIONING

OPERATIONS

-GROUND ACTIVATION OF POWER/THERMAL

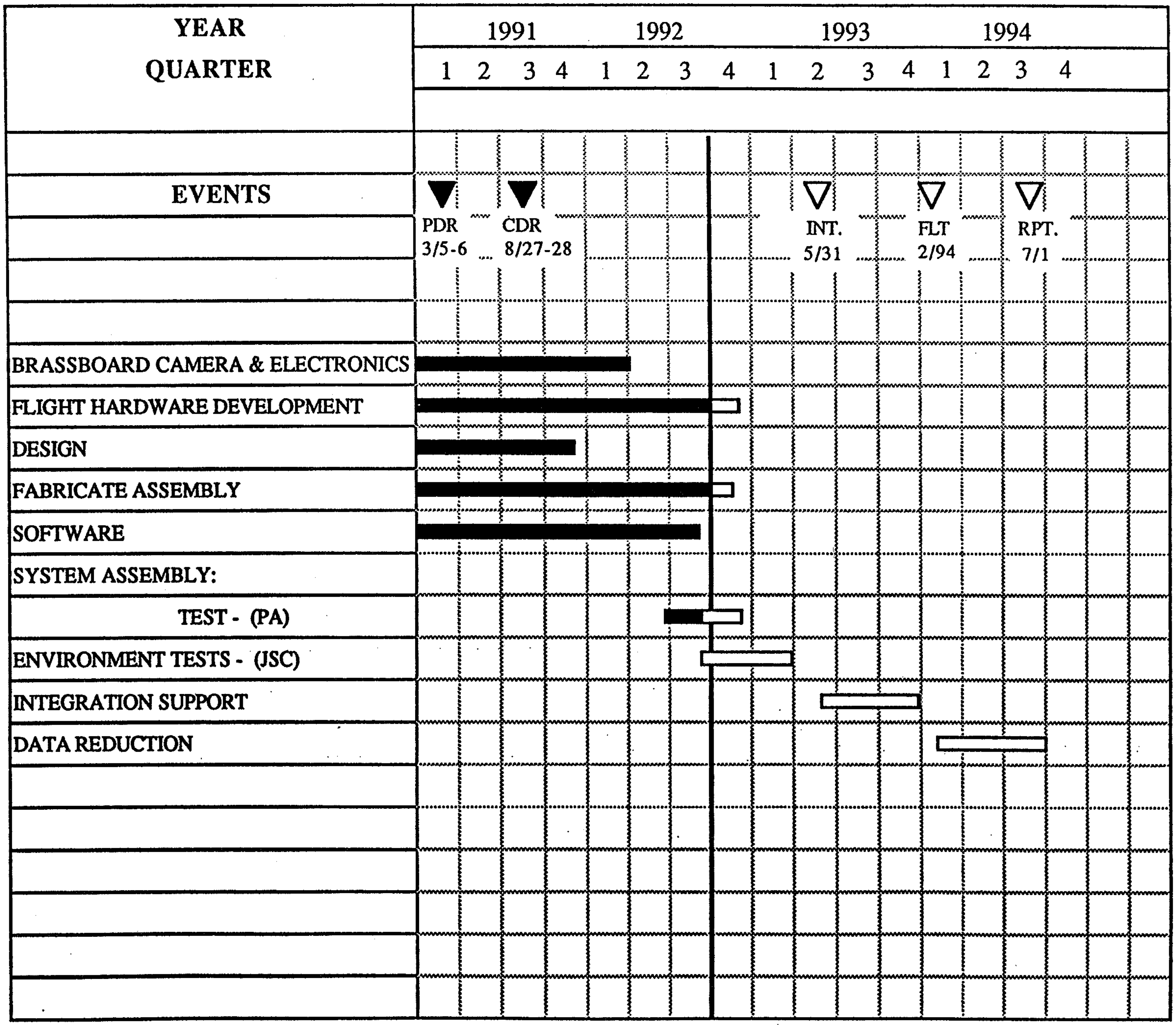
-UPLINK COMMANDS TO PREP EXPERIMENT

-DOWNLINK DATA S-BAND (SAMPLE) AND KU BAND (ALL DATA) WITH TAPE RECORDER BACKUP

-TAPE RECORDER PLAYBACKS ON ORBIT PLANNED

-HANDHELD CAMERA OPERATED BY CREW

EISG Program Milestone Schedule for 4th Quarter



Tuesday, October 6, 1992

AZ Tek
AZ TECHNOLOGY

STIG Flight Experiments
Technical Interchange Meeting

OPM

The Optical Properties Monitor (OPM)
A Multipurpose Optical Laboratory In Space

Donald R. Wilkes

AZ Technology, Inc.
3322 Memorial Parkway SW, Suite 93
Huntsville, AL 35801
(205) 880-7481

October 5-9, 1992
Monterey, California

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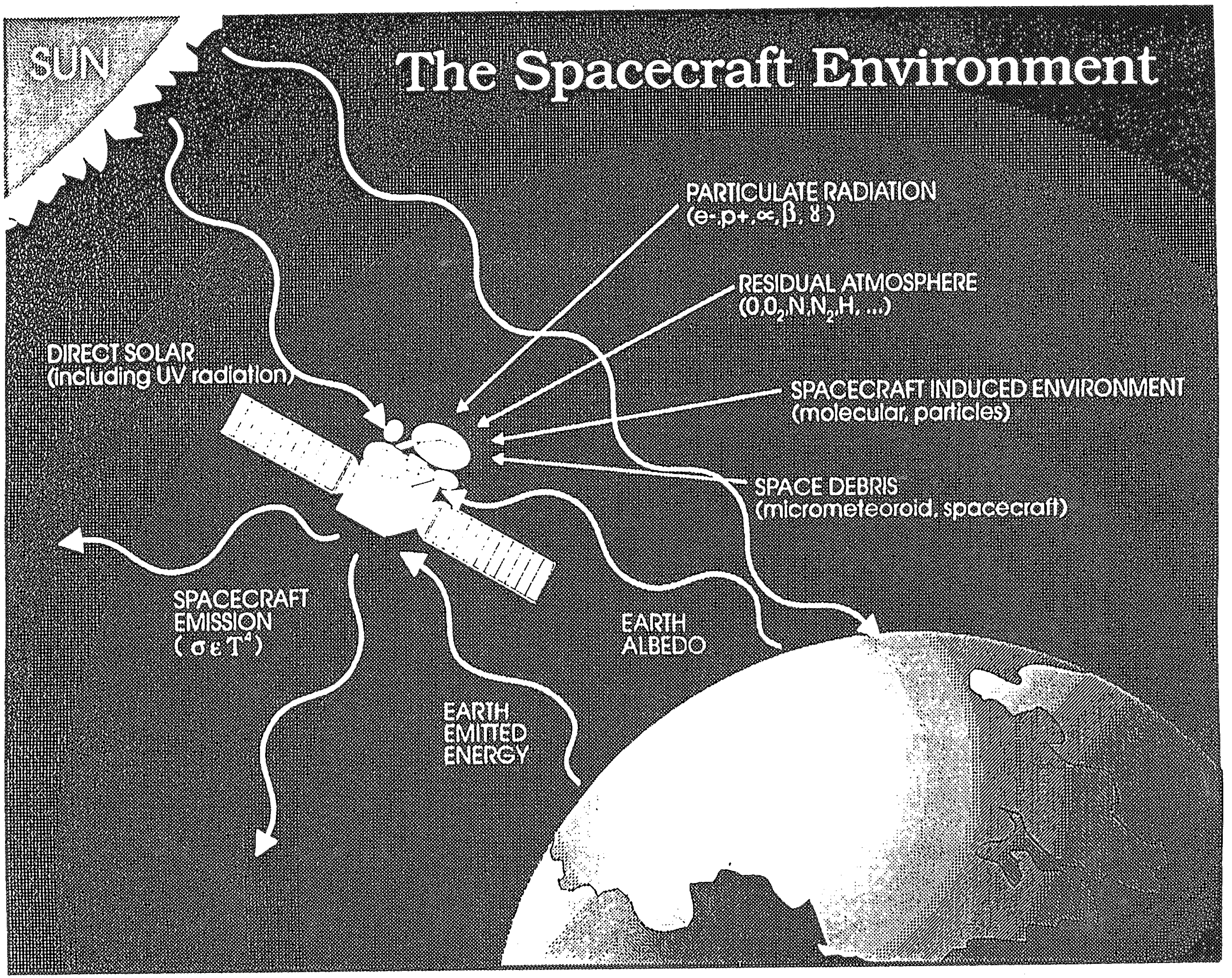

SUN
The Spacecraft Environment
PARTICULATE RADIATION
(e-, p+, ∝, β, γ)
RESIDUAL ATMOSPHERE
(O, O₂, N, N₂, H, ...)
DIRECT SOLAR
(including UV radiation)
SPACECRAFT INDUCED ENVIRONMENT
(molecular, particles)
SPACE DEBRIS
(micrometeoroid, spacecraft)
SPACECRAFT
EMISSION
(σεT⁴)
EARTH
ALBEDO
EARTH
EMITTED
ENERGY

Technical Background

- The natural and induced space environment can damage spacecraft and instrument materials
- Space environmental effects and damage mechanisms are not well understood
- The space environment cannot be fully simulated in the laboratory
- There have been only limited in-space optical measurements of material properties
- Analytical lifetime prediction models are limited due to lack of time vs. effects flight data

Technology Need

- Longer duration, and more complex missions, such as Space Station Freedom, require better materials and improved materials performance characterization

- A better understanding of space environmental damage mechanisms will lead to:
    - More stable materials and coatings
    - More accurate ground simulation testing
    - Lifetime prediction models for materials in the space environment

- Improved materials and better material performance characterization will lead to more cost effective, lower weight, higher performance, maintainable space systems designs

Need for an In-Space Experiment

- Time dependent flight data is required to understand the non-linear nature of materials degradation.
- Some effects of the space environment on materials are reversible when returned to the terrestrial environment.
- The space environment cannot be fully simulated in the laboratory.
- There is significant disagreement between flight and laboratory simulation testing of materials.
- Flight tests of new and improved materials are required before full acceptance of these materials by space hardware designers.

OPM Experiment Objectives

To study the effects of the space environment, both natural and induced, on optical, thermal control, solar array and other materials.

- Determine the effects and damage mechanisms of the space environment on materials.
- Provide data to validate lifetime prediction models.
- Perform flight testing of critical spacecraft and instrument materials.
- Provide data to validate space simulation test facilities and techniques.
- Develop a reusable multifunctional flight instrument for optical studies.

OPM Experiment Concept

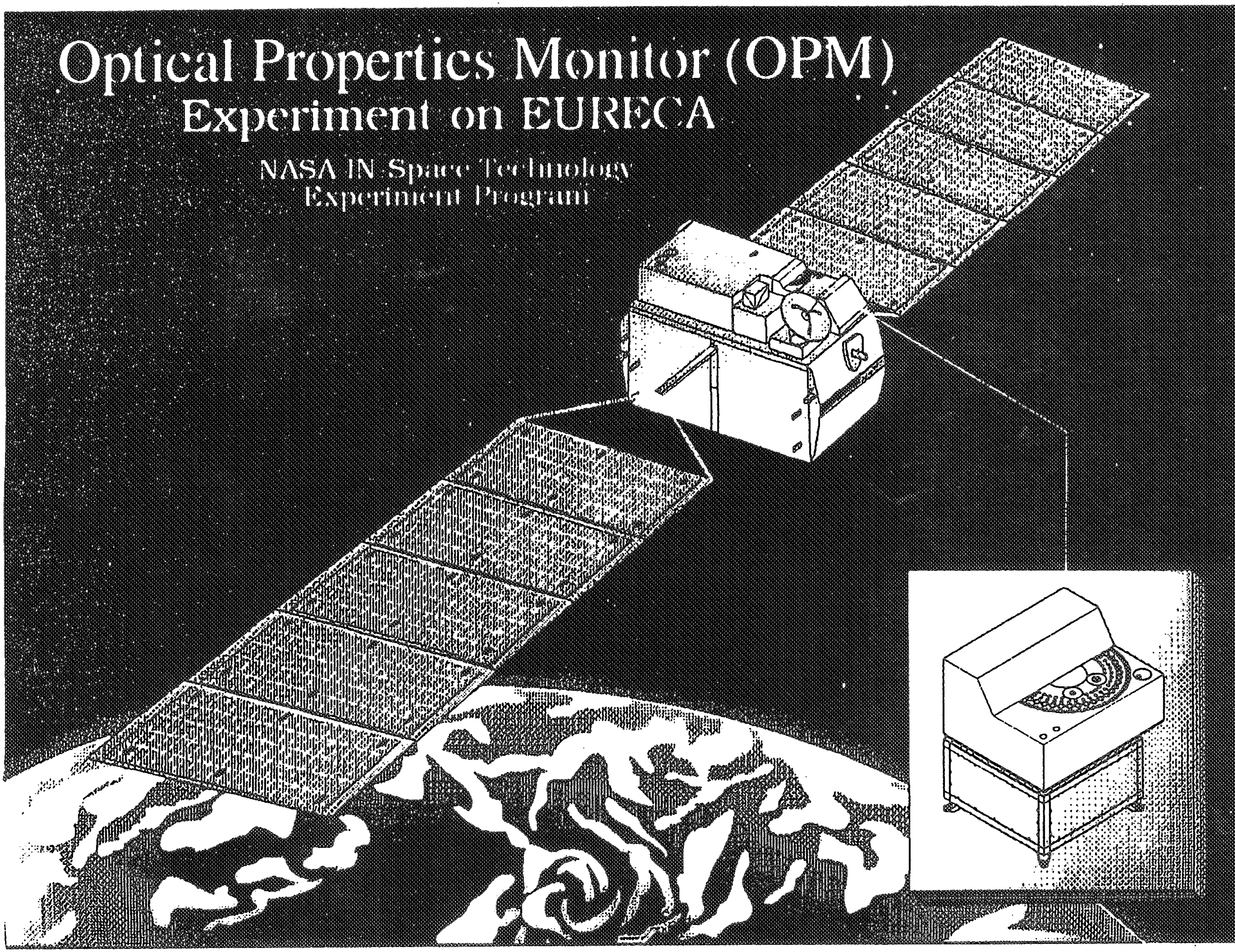
- The OPM is a multifunction in-flight laboratory for in-situ optical studies of materials.
- Many independent and related studies can be carried out on EURECA with the OPM instruments to address the experiment objectives.
- Test Samples will be selected to address the materials and issues of the greatest interest to NASA, ESA, DoD, and the aerospace community. A Sample Selection Advisory Committee (SSAC) will be formed and chaired by the OPM PI and MSFC Project Scientist.

Experiment Summary

Selected materials will be exposed to the low earth orbit space and EURECA environment and their effects measured through in-situ measurements and post-flight analyses.

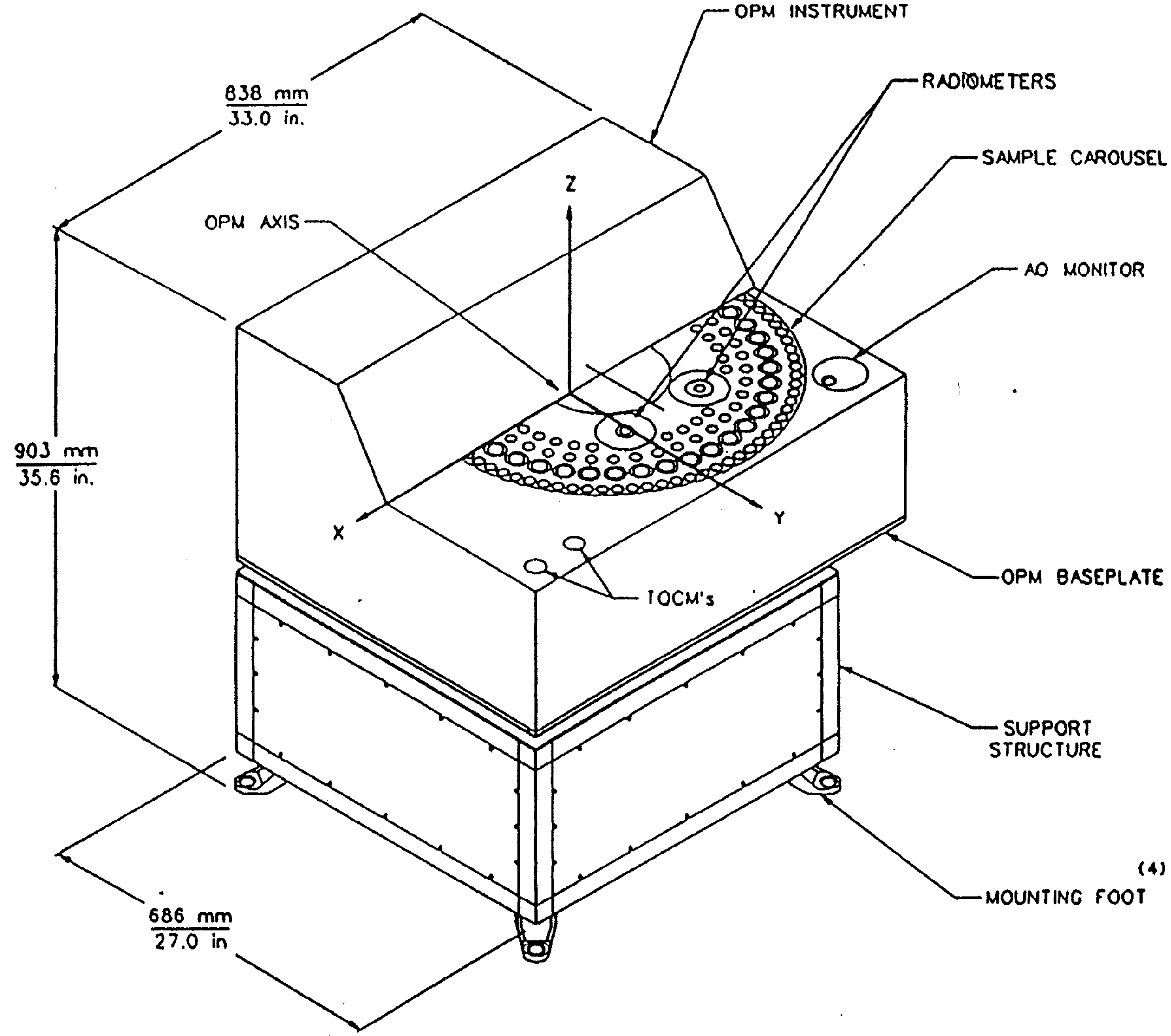
- Optical and thermal properties are measured by in-situ measurement subsystems
  - Spectral total hemispherical reflectance
  - Total Integrated Scatter (TIS)
  - Vacuum Ultraviolet (VUV) reflectance/transmittance
  - Total emittance
- Environmental monitors measure selected components of the exposure environment
  - Solar/earth irradiance
  - Molecular contamination
  - Atomic oxygen
- Detailed optical and thermal properties, surface degradation, and contamination are determined by post-flight analysis.

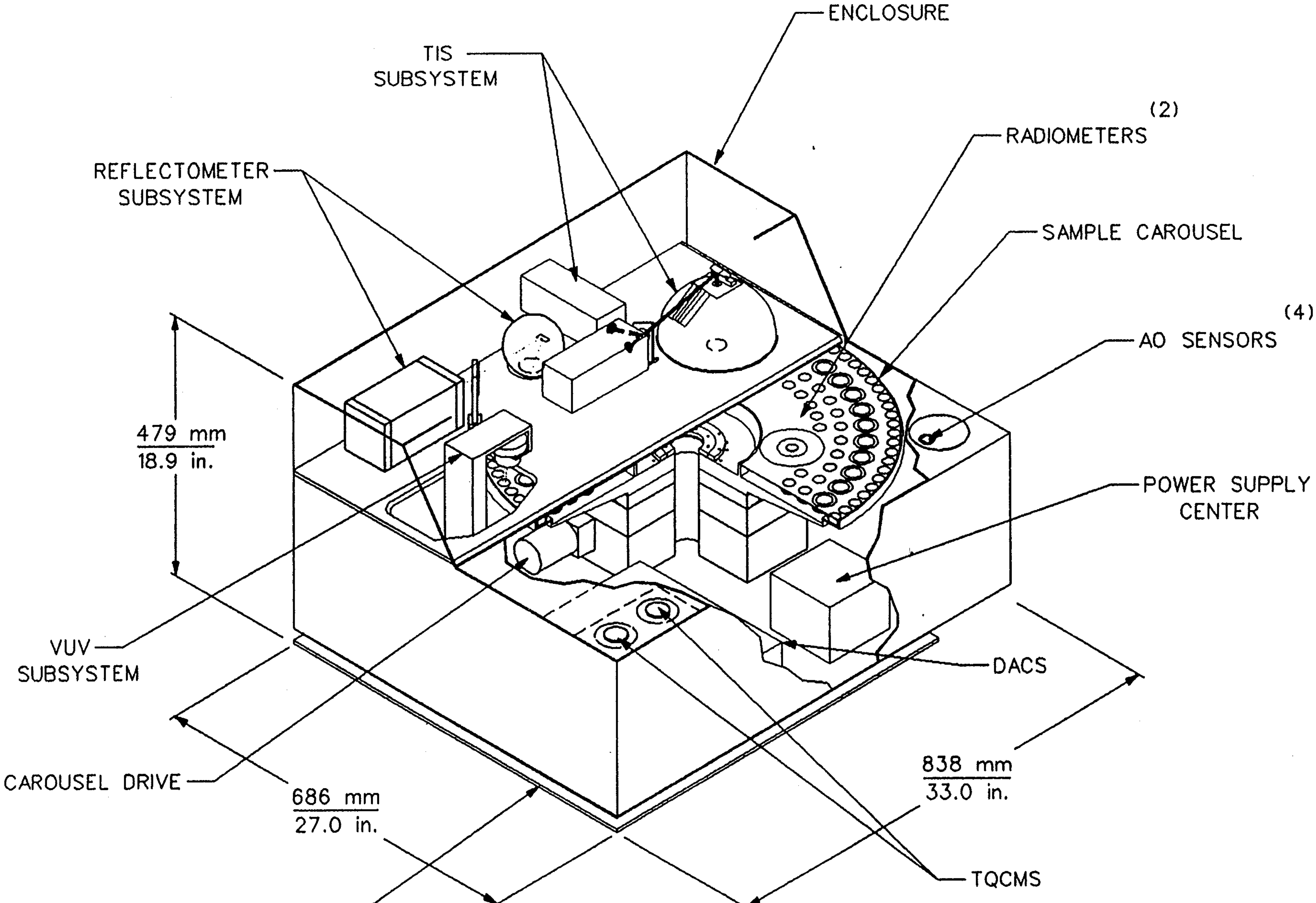
- Experiment results will be disseminated to the aerospace community through IN-STEP conferences, technical conferences and publications, space materials handbooks, and materials databases.



Optical Properties Monitor (OPM)
Experiment on EURECA
NASA IN Space Technology
Experiment Program

OPM Instrument and Support Structure





OPM ASSEMBLY

OPM Hardware Summary

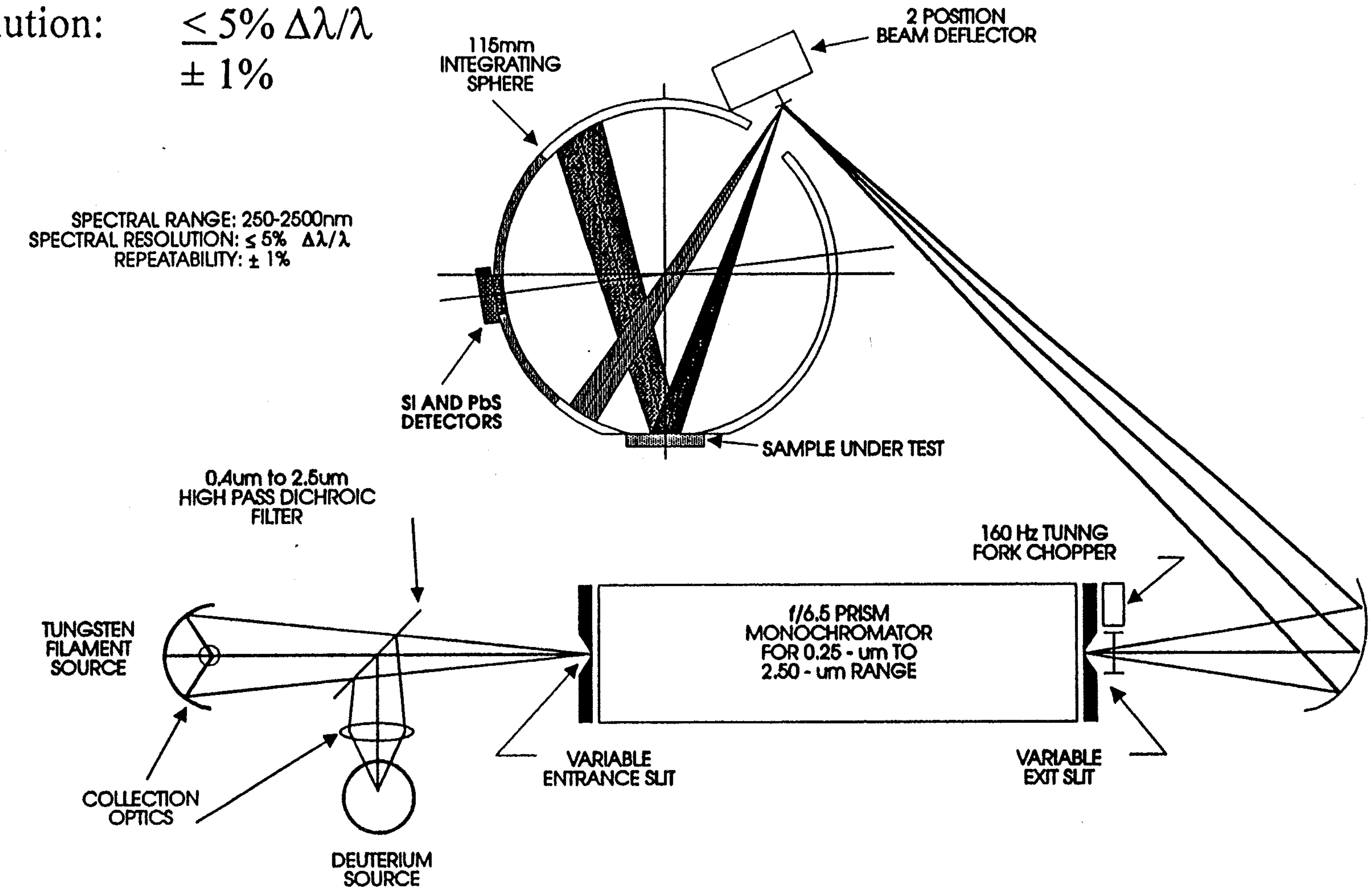
Size	686mm x 838mm x 903mm (27" x 33" x 35.6")
Mass	127 Kg (279 lbs)
Number of Test Samples	188
Power Source	28VDC S/C Power
Power - Average	43 watts
- Peak	258 watts
Command/Data Interface	EURECA RAU (Serial)
Data Rate - Average	5 bps
- Peak	50 bps

Reflectometer Optical Schematic

Spectral Range: 250 - 2500 nm

Spectral Resolution: $\leq$ 5% $\Delta\lambda/\lambda$

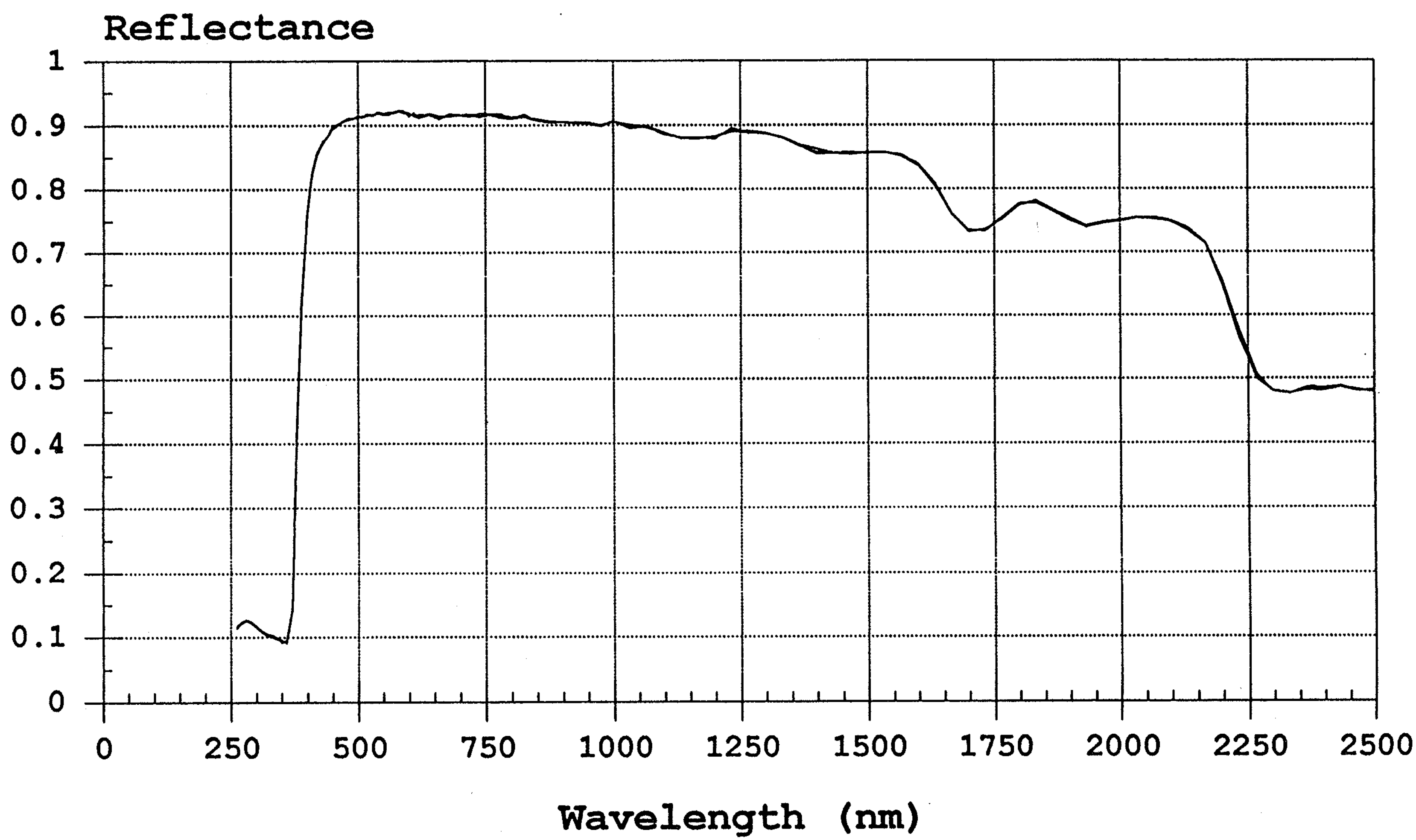
Repeatability: ± 1%



STIG Flight Experiments
Technical Interchange Meeting

OPM

Performance of the updated reflectometer design is demonstrated in this plot of three separate measurements of S13G/LO white paint.



Monterey

AZ Tek AZ TECHNOLOGY	STIG Flight Experiments Technical Interchange Meeting	**OPM**

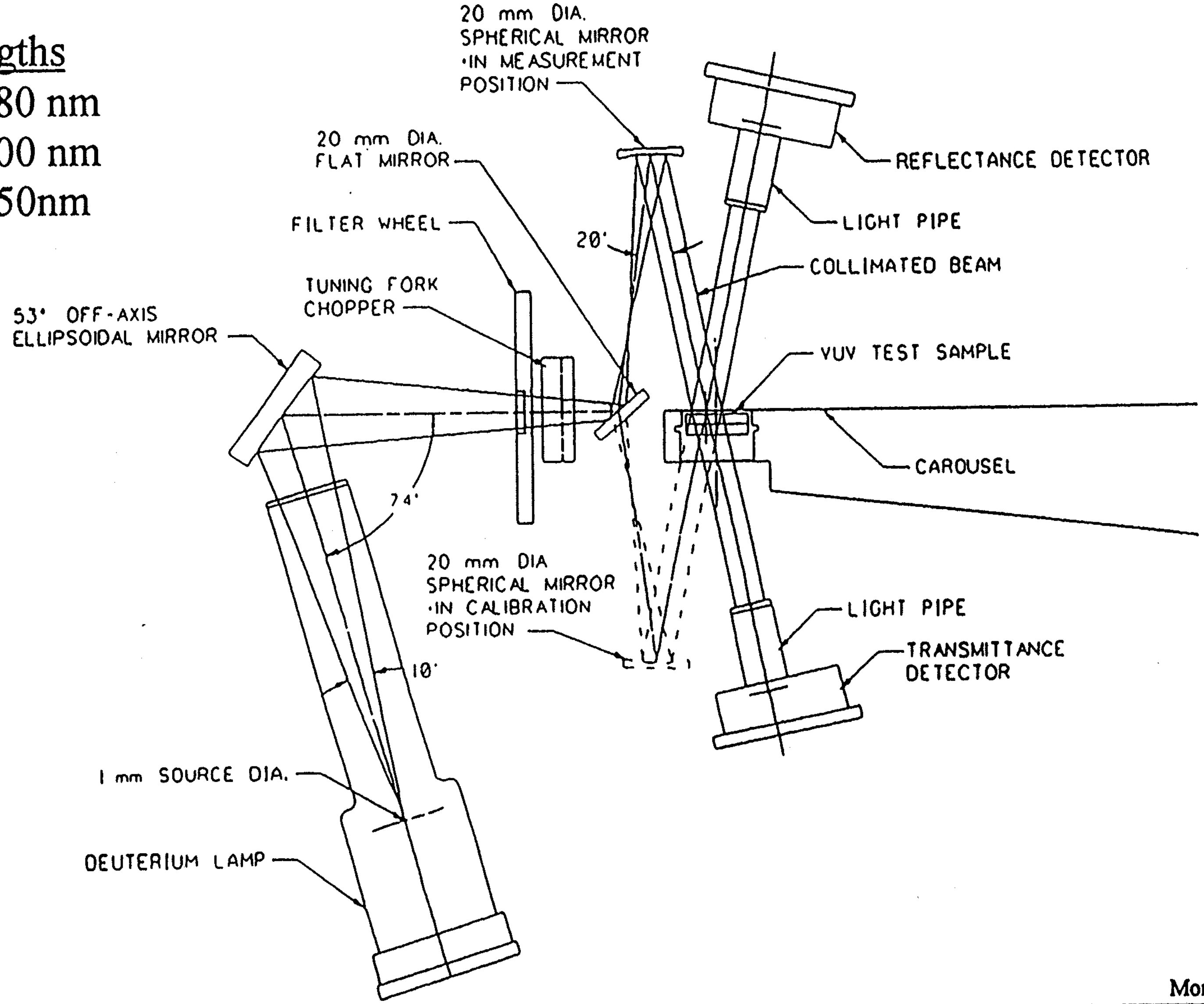
VUV Spectrometer Design

Measurement Wavelengths

- 121.6 nm
- 160.6 nm
- 170 nm
- 180 nm
- 200 nm
- 250nm

Accuracy: ± 5%

Repeatability: ± 2%



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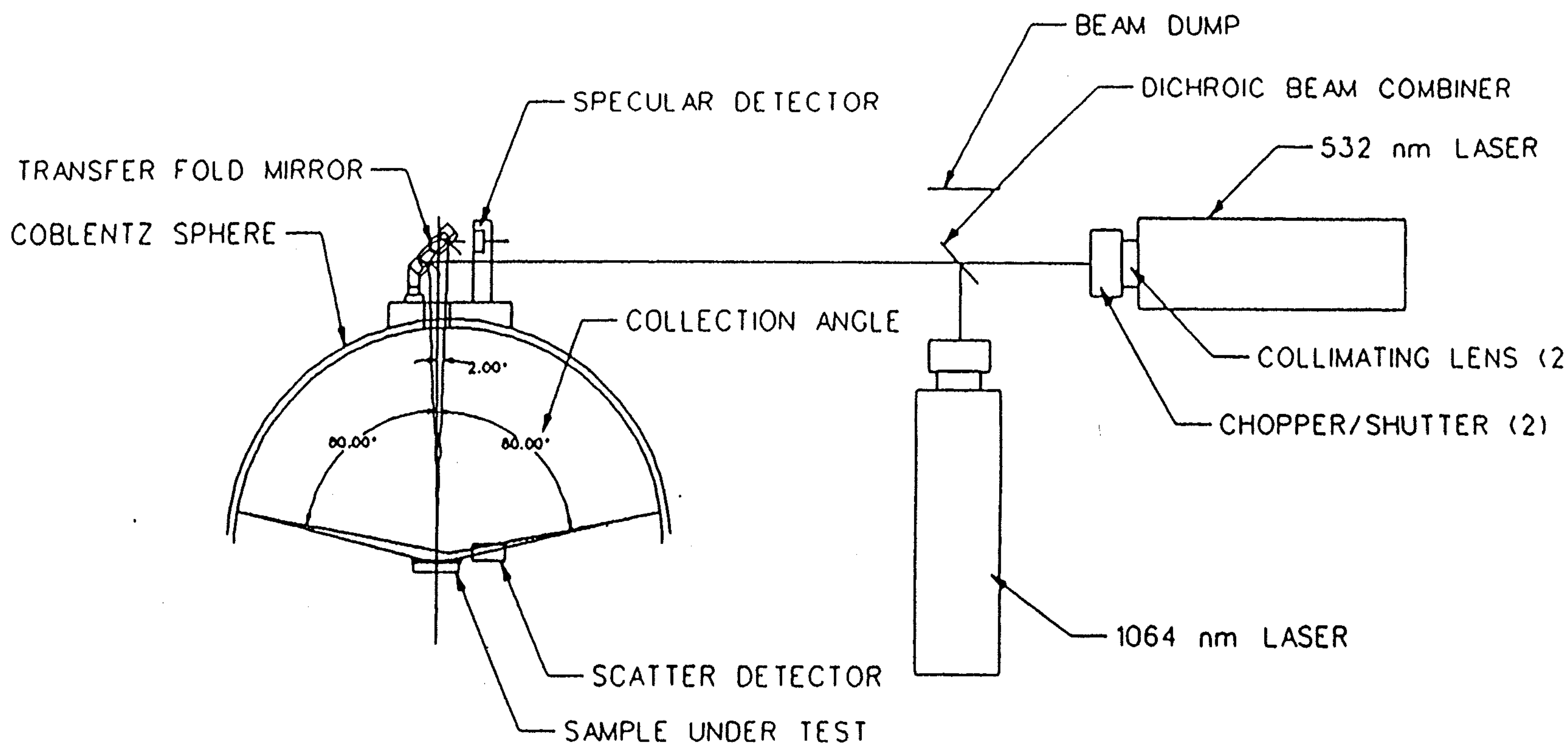
TIS Optical Schematic



TIS = $V_{scattered}/V_{specular}$ RMS = $(\lambda/4\pi)(TIS)^{1/2}$

Scatter Collection Angle: 2.5° to 80°

Accuracy : ± 10%

Repeatability : ± 2%

RMS Measurement Range: 5 - 500 Å

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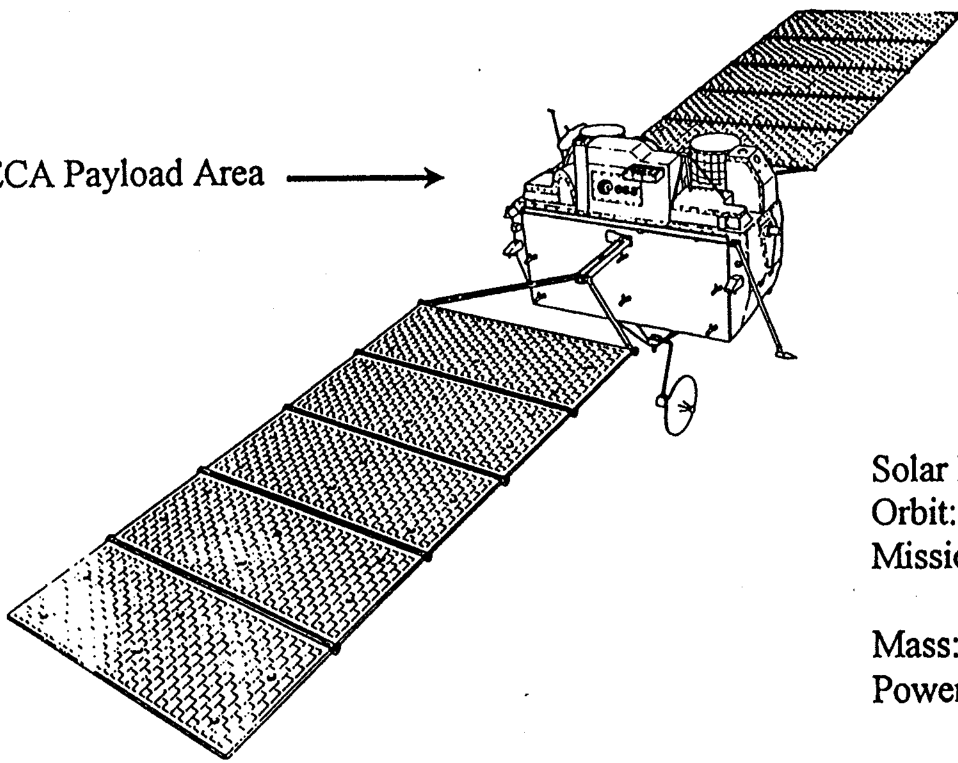
EURECA Summary

EURECA Payload Area →



EURECA

EUropean REtrievable CArrier

Solar Pointing Shuttle Launched Free Flier
Orbit: 525 km at 28.5^{o} inclination
Mission Duration: 6 Months Operational (Nominal)
3 Months Dormant
Mass: Total: 4500 kg
Power: Available to Payload: 1000 W
Peak: 1500 W
Thermal Control: Passive or Liquid Freon Loop (1000 W)
Data: High Speed Down-Link: 256 kbps
Low Speed Down-Link: 2 kbps
Solar Pointing Accuracy: 1^{o}
Microgravity: 10^{-5}g < 1 Hz
10^{-3}g > 100 Hz

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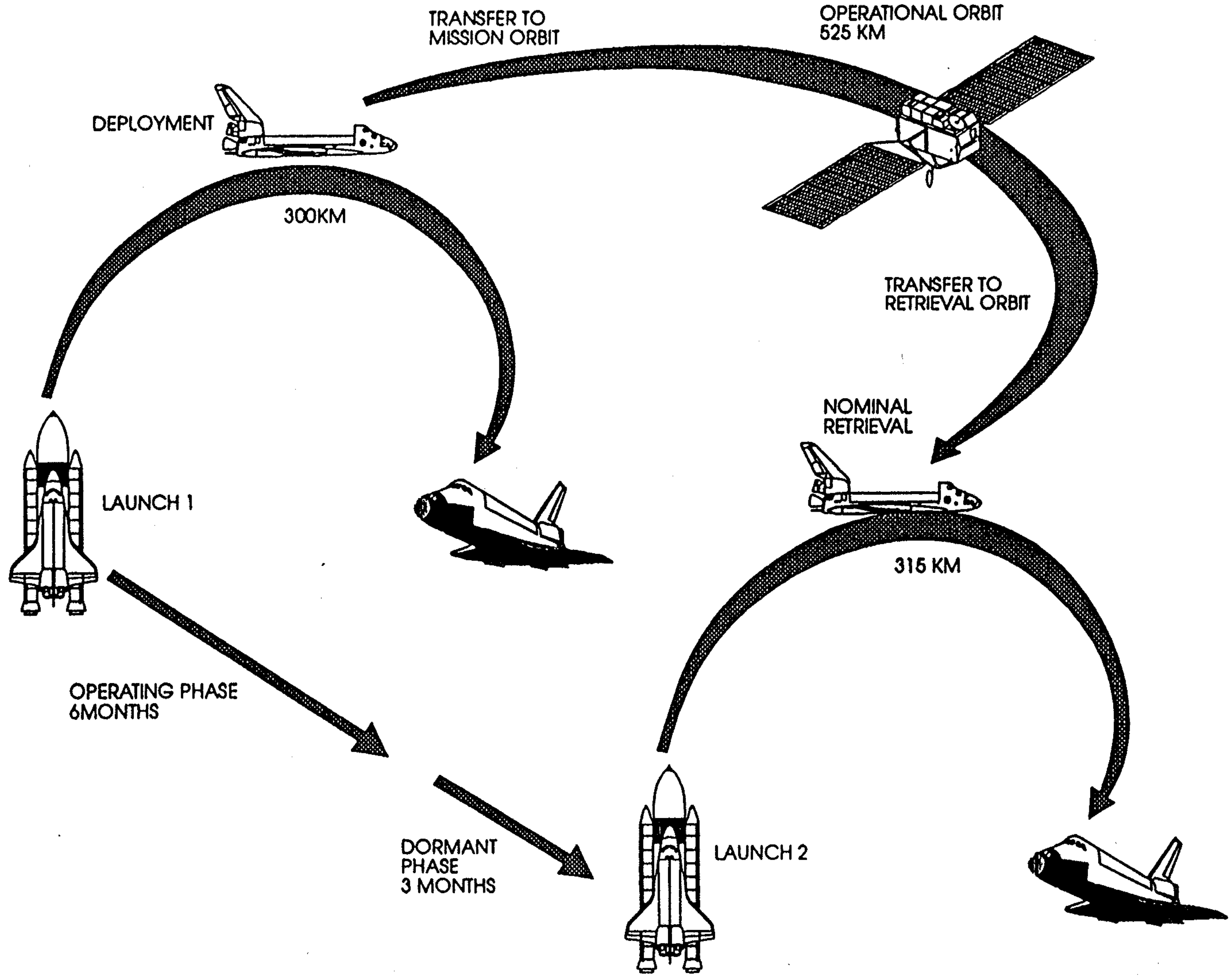
STIG Flight Experiments
Technical Interchange Meeting

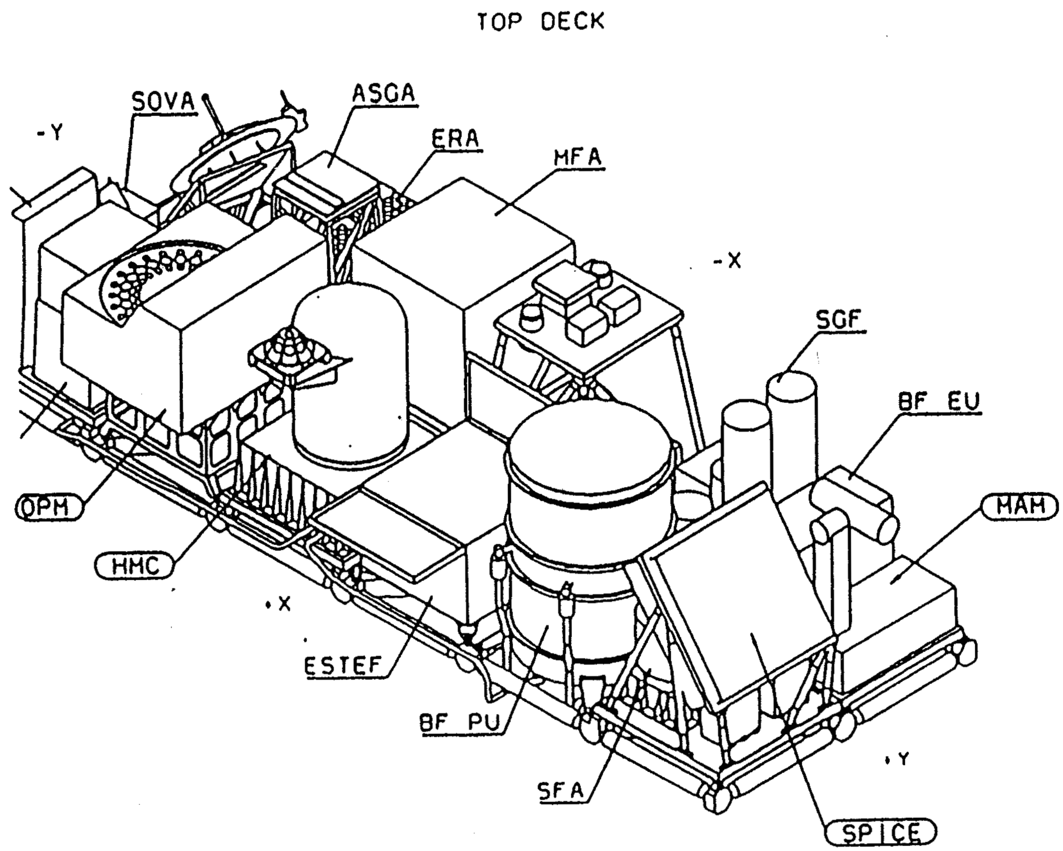
OPM

EURECA Flight Scenario



Monterey

OPM Accommodation on EURECA Payload Deck



EURECA Payload Exposure Environment

Operational Orbit (525 km)

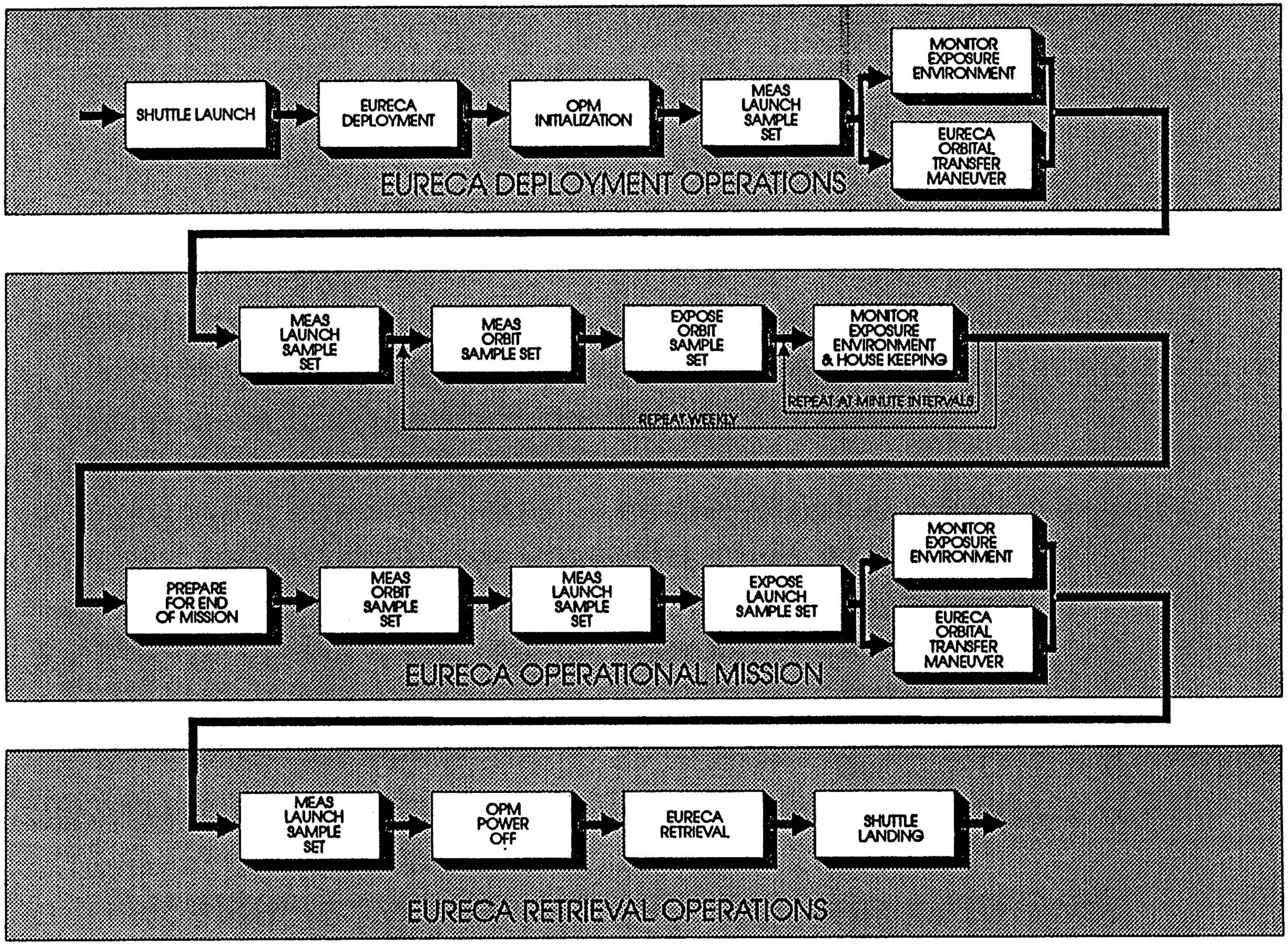
- Six months total exposure
- Solar exposure: ~3,000 direct sun hours
  (Equivalent to 3.3 x exposure of non-solar oriented vehicle such as the Space Station or 1.7 years)
- Atomic Oxygen Fluence: ~ 10^{20} atoms/cm^2

Parking Orbit (300 km)

- Approximately one month exposure
- Solar exposure: ~ 150 direct sun hours
- Atomic Oxygen: ~ 10^{20} atoms/cm^2

Boost/De-Boost

- Monopropellant hydrazine residue contamination - EURECA offers the additional opportunity to study the in-situ effects of boost/deboost operations



OPM Mission Operations

opm-ms2.cd

Example OPM/EURECA Technology Studies

Many different technical studies can be carried out on the OPM/EURECA mission.

- Flight tests of selected spacecraft and instrument materials
- Sensitivity of optical scatter to the space environment
- Synergistic effects of the contituents of the space environment
- Launch, retrieval and orbital manueuver effects on materials
- The effects of processing variables on the stability of anodized coatings
- Effectiveness of AO protective coatings
- Effectiveness of AO cleaning of molecular contamination
- Performance of different adhesives and application methods for silver Teflon
- Characterize the environment of a Shuttle launched free-flier

OPM Milestones

Preliminary Design Review	August, 1992
Non-Advocate Review	September, 1992
Critical Design Review	December, 1993
OPM Flight Hardware Delivery	March, 1996
EURECA Launch	September, 1997
EURECA Retrieval	March, 1998
OPM Quick Look Report	June, 1998
OPM Final Report	March, 1999

N93-28730

RADIATION MONITORING EQUIPMENT DOSIMETER EXPERIMENT

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INTRODUCTION

Spacecraft crews risk exposure to relatively high levels of ionizing radiation. This radiation may come from charged particles trapped in the Earth's magnetic field, charged particles released by solar flare activity, galactic cosmic radiation, energetic photons and neutrons generated by interaction of these primary radiations with the spacecraft and crew, and man-made sources (e.g., nuclear power generators). As mission are directed to higher radiation level orbits, viz., higher altitudes and inclinations, longer durations, and increased flight frequency, radiation exposure could well become a major factor for crew stay times and career lengths. To more accurately define the radiological exposure and risk to the crew, active, real-time radiation monitoring instrumentation must be flown capable of identifying and measuring the various radiation components. This presentation describes a radiation dosimeter instrument which has been successfully flown on the Space Shuttle, the RME-III.

The RME-III is a self-contained, portable, active (real-time) dosimeter system developed by E.G.&G., Inc. for the U.S. Air Force (USAF) and is adapted for use in measuring the ionizing radiation environment on the Space Shuttle. RME stands for "radiation monitoring equipment", the title given to 2 prototype radiation instruments sucessfully flown by the USAF on several pre-Challenger missions (1). These prototype instruments had relatively short battery life and very limited memory capacity; thereby, requiring an inordinate amount of crew interaction time. The RME-III was developed to incorporate the capabilities of both radiation instruments into a single unit and to minimize crew interaction times with longer battery life and expanded memory capacity. A description of the RME-III instrument and the results from flight of several Space Shuttle missions is presented in this report.

INSTRUMENT DESCRIPTION

The RME-III is a portable, self-contained, active dosimeter system. The system features a 3-channel tissue equivalent proportional counter (TEPC) which measures particle fluence and computes dose and dose equivalent at operator selected time intervals. The total accumulated absorbed dose and dose equivalent are displayed real-time on a liquid crystal display while the data and time of the interval dose readings are stored in memory modules for future analysis. Analysis of the time-resolved data permits correlation of the radiation exposure with geographic position, altitude and instrument location in the space craft. The instrument consists of 2 components: (1) the

Main Module, and (2) the Memory Module. The memory module is designed to plug into the Main Module and is exchanged with a new module when the memory capacity or primary system battery life is reached. The overall dimensions of the system are 28.6 cm x 10.2 cm x 5.1 cm. The combined total weight is 1550 g (3.3 lbs).

Main Module Description

The main module contains the analog and digital operating circuitry of the instrument and the TEPC. At the top of the module are 2 locking toggle switches. One is the system power "ON-OFF" switch. The other is a keyboard locking switch, which prevents accidental entries via the keyboard during system operation. The display is a 6.4 cm x 3.8 cm liquid crystal display (LCD), displaying 8 lines of data with 20 characters per line. The system operating parameters are entered via a 12 key keyboard. Primary operating power is provided by 5 AA alkaline batteries in the memory modules. Zinc-air batteries are used in the Main Module to maintain the instrument operating parameters and internal operrating clocks during Memory Module replacement.

Memory Module Description

The Memory Module contains a 0.5 Megabyte Random Access Memory (RAM) into which the RME-III data is stored for future analysis. The RAM is also backed up by zinc-air batteries. The primary operating power for the entire system is provided by 5 AA batteries in the Memory Module, providing approximately 42 hours of continous ooperation. The dimensions of the Memory Modules are 10.2 cm x 10.7 cm x 4.4 cm. The stand-alone weight of the Memory Module is 595 grams. Several Memory Modules are flown

with the Main Module on a Shuttle mission, sufficient enough to cover the entire mission.

TEPC Description

The heart of the system is the cylindrical tissue equivalent proportional counter developed by E.G.&G. The TEPC is a cylindrical (1.27 cm ID, 5.05 cm long) ionization chamber, lined with tissue equivalent plastic (type A-150), and filled with methane gas at reduced pressure (80 mmHg). The reduced gas pressure allows the chamber to simulate a tissue volume of about 2 micrometers in diameter. The TEPC is designed to measure the rate of energy loss of individual particles passing through the detector. The RME-III TEPC permits measurement of the radiation energy loss in a simulated volume on the order of the size of a typical human cell. Thus, the RME-III is a microdosimetry based instrument.

RME System Operation

The system block diagram of the RME is shown in the next figure. Pulses from the TEPC are amplified and shaped by the pr-amplifier and linear amplifier, and are fed into comparators that separate the spectrum into 3 pulse height or linear energy transfer (LET) bins: 0.35 to 6.5 keV/um, 6.5 to 30 keV/um, and 30 to 120 keV/um. Pulses greater than120 keV/um are placed in the highest bin. The comparators in effect serve as a 3-channel pulse height analyzer. The actual counts in each pulse height bin are stored in the memory and can be used after-the-fact to more accurately estimate exposure, if required.

The instrument uses an algorithm to convert counts in each

pulse height bin into absorbed dose exposure values. For rads tissue a multiplicative constant is obtained for each of the 3 bins by exposing the instrument of Cs-137 and Cf-252 sources. The Cs-137 gamma source is used to determine the conversion constant in the low pulse height enerrgy bin. The Cf-252 spontaneous fission source permits determination of the constants for the two higher energy pulse height bins. The constants determined by the calibration procedure are stored in the Main Module processor. The instrument calibration is checked before and after each mission.

The dose equivalent is computed by multiplying the measured absorbed dose per channel by a present quality factor. The quality factors used in the Shuttle missions presented in this report were 1 for the lowest pulse height channel, 9 for the intermediate channel, and 15 for the upper channel.

RME-III Setup Menu

The RME-III is menu driven. When the RME is initially turned on the set-up menu appears as shown in DISPLAY 1. The operator can the select the following operational parameters:

a. Time interval for dose integration. Ten seconds is the value that is used on the Shuttle.

b. Alarm levels for total dose and dose rate. These are normally set to OFF on the Shuttle flights.

c. Calendar Date

d. Coordinated universal Time (UTC)

e. Mission Elapsed Time (MET)

Normally all of the operating parameters are preset by the

ground crew at L-5 with the exception of the MET. The MET is entered by the Shuttle crewmember when the unit is turned on after launch. After presetting by the ground crew, the system is turned off and stowed in a mid-deck locker on the Shuttle. After lift-off the RME-III is removed from the locker by the crewmember, the power turned on, the keyboard unlocked and the MET entered. The crewmember then presses the START button, locks the keyboard and places the RME in the appropriate operating location. DISPLAY 3 is the display the operator normally sees during operation. The Shuttle cremwmbers are trained in the operation of the RME and can be called upon to change or correct the operating parameters during flight, if required.

DATA

Data may be read out directly from any memory module by using the Main Module display and control keys. This function is a lengthy process since there are usually many sets of data (30,000 to 50,000) for a typical Shuttle mission. The normal procedure is to download the data from the Memory Modules to a personal computer (PC) after the mission. This task is accomplished by attaching the Memory Module to a special readout module which transfers the data from the Memory Module to a file in the PC via an RS-232 protocol. An 80C31 processor with data conversion algorithms is used to handle the transfer and conversion of the Memory Module data files. These files contain the raw data, the timing information (date, UTC, MET, and interval length), the absorbed dose in rads and the dose equivalent in rem for each time interval, and the integral dose in rads and rem for the

whole time the Memory Module recorded data.

The next figure shows representative data files obtained during Shuttle missions. The top file shows the data obtained at the start of one data set from a memory module. The counts in channel #1 are the low LET counts. The counts in Channels #2 and #3 are the intermediate and the high LET counts. The data in this file is representative of the count rate data obtained while traversing the equatorial latitudes, where the radiation background is due primarily to high energy cosmic rays. The middle file is an example of count rate data obtained at higher latitudes, where weakening of the Earth's magnetic filed permits more cosmic ray particles to reach the Shuttle. The bottom file is an example of data collected during a pass through the South Atlantic Anomaly (SAA) on STS-31. The bulk of the dose is due to the high energy, low LET component of the inner Van Allen Belt protons, but the increased counts in the intermediate and high-LET channels indicate the presence of either high-LET, low energy protons, and/or the presence of high LET secondaries, which have a marked effect on the resulting dose equivalent.

The data for the first six Shuttle missions flown by the RME-III are summarized in Table I. Shown for comparison are the average Passive Radiation Dosimeters (PRD) doses measured on the Shuttle with thermoluminescent dosimeters by NASA/JSC. Six PRDs are routinely flown at different locations in the Shuttle. The highest doses/dose rates are measured in the high altitude missions which penetrate the SAA. In these orbits the dose contribution is due primarily to the Van Allen Belt protons. In the lower altitude orbits, the Galactic Cosmic Ray component predominates.

The data files are in ASCII or binary format and can readily be manipulated to produce plots of counts or dose vs. time or position. The next figures represent data obtained from various Space Shuttle missions displayed in this format. Figure 8 illustrates dose rate at 10s intervals vs MET for a low altitude, high inclination mission in which there is little interaction with the SAA and the exposures are dominated by the cosmic ray background. Notice the cyclic nature of the cosmic ray background with the peaks occurring at the high latitudes and the valleys at the equator caused by the geomagnetic shielding effect of the Earth. Figure 9 illustrates dose rate vs. MET for a Shuttle mission in which the altitude was sufficiently high to penetrate deeply into the SAA. Notice the peaks due to the penetration of the SAA. Dose rates on the order of several hundred millirad per hour were measured on this mission during some of the SAA passes.

The RME-III data readily lends itself to other forms of analysis. Figure 10 is a plot of dose rate as a function of orbiter geographic position (longitude and latitude) which were made from 38241 individual measurements taken during 113 hours of operation on the STS-28 Space Shuttle mission. The next figure (fig. 11) illustrates a plot of the South Atlantic Anomaly in two dimensions obtained from Space Shuttle RME-III data.

CONCLUSION AND DISCUSSION

Flight data from the Space Shuttle missions has demonstrated that a micro-dosimetric based radiation instrument such as the RME-III with the capability of registering counts and exposures vs. MET and UTC can be used to accurately assess dose form various sources of exposure, such as that encountered in the complex radiation environment of space. The RME-III is presently manifested to fly on STS-53 next month, and on STS-56 and -51 in 1993. Our goal is to fly

the RME-III on at least two or three more flights in 1994, the point of solar minimum. The RME-III flew its first mission on STS-27 in Dec 88 just before solar maximum. The RME-III is the only real-time radiation instrument to fly on the Shuttle from Solar Maximum to Solar Minimum. The data from all of these mission are being analyzed. The data is being used to enhance spacecraft shielding and space radiation models; and to investigate phenomena such as the drift of the SAA with time, the variation of the GCR background in LEO with the solar cycle, and diurnal (day vs. night) variations in the space radiation quality factors.

"Smart" dosimetric instrumentation such as the RME-III have definite applicability to the Aerospace operational environment. Newer instruments with larger memory capacities and greater spectral resolution are currently being developed which will ultimately be flown on the Space Station Freedom and future long duration space missions. Modified versions of the RME-III will soon be used by the Air Force to investigate the cosmic ray and solar radiation exposure of air crews at high altitudes.

REFERENCES

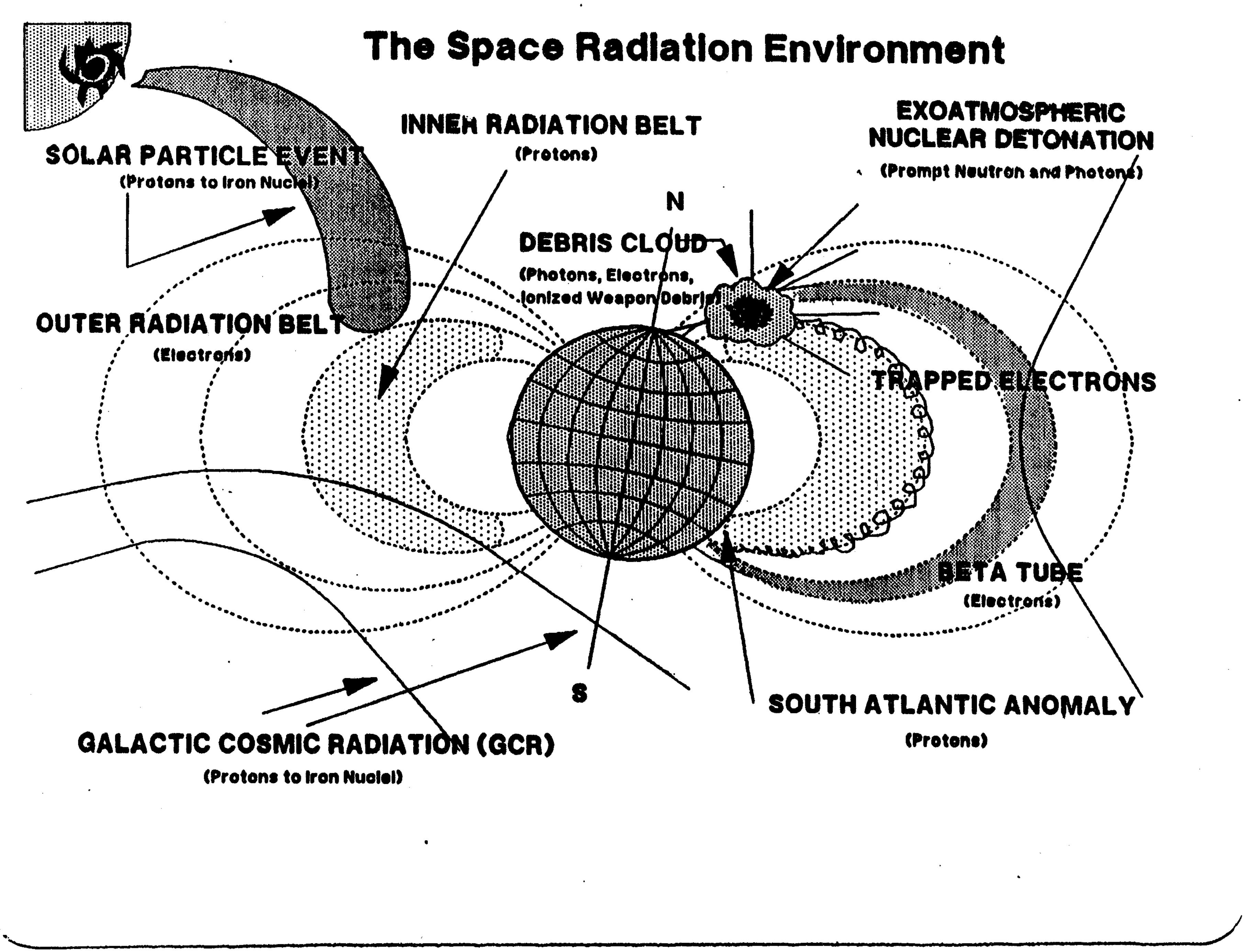
1. Cash, S. E., Madonna, R. G., McClellan, M. R., Steskal, M. J., and Fields, M. E. Results for Radiation Monitoring Equipment Experimens of STS-41-C, 41D, 41G, and 51A. AFTAC-TR-85-4, April 1985.

RME-III

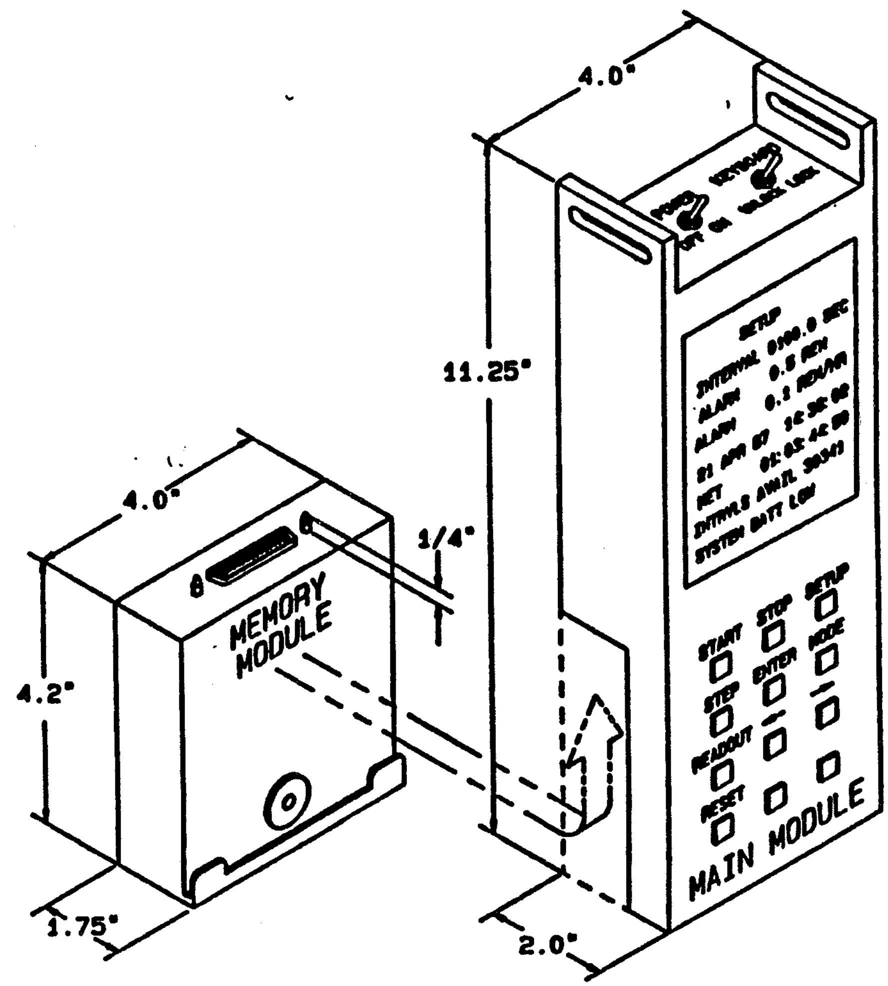
RADIATION MONITORING EQUIPMENT

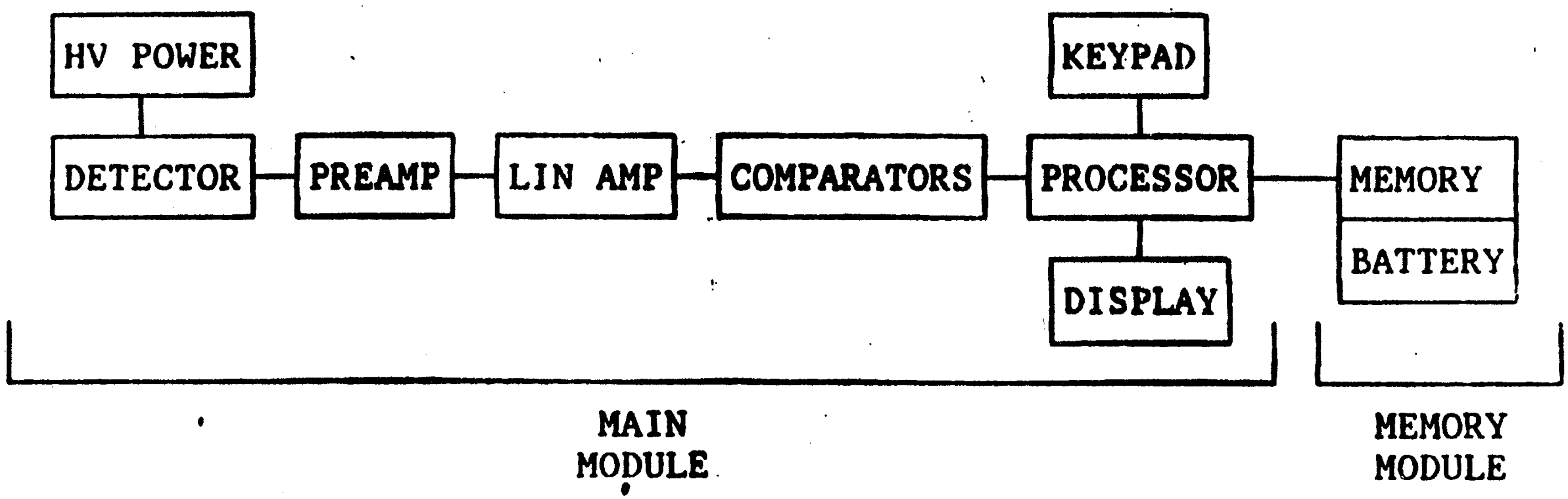
KENNETH A. HARDY

Radiation Physicist
Dosimetry Function
Occupational Environmental
Health Directorate
USAF Armstrong Laboratory



The Space Radiation Environment
SOLAR PARTICLE EVENT
(Protons to Iron Nuclei)
INNER RADIATION BELT
(Protons)
EXOATMOSPHERIC
NUCLEAR DETONATION
(Prompt Neutron and Photons)
N
DEBRIS CLOUD
(Photons, Electrons,
Ionized Weapon Debris)
OUTER RADIATION BELT
(Electrons)
TRAPPED ELECTRONS
BETA TUBE
(Electrons)
S
SOUTH ATLANTIC ANOMALY
(Protons)
GALACTIC COSMIC RADIATION (GCR)
(Protons to Iron Nuclei)


4.0"
11.25"
SETUP
INTERVAL
ALARM
ALARM
NET
SYSTEM BATT LOW
START
STOP
SETUP
ENTER
MODE
STEP
READOUT
RESET
MAIN MODULE
2.0"
4.0"
1/4"
MEMORY
MODULE
4.2"
1.75"



RME III BLOCK DIAGRAM

SETUP

INTERVAL 10 SEC
ALARM OFF mREM
ALARM OFF mREM/HR
29 SEP 88 14:30:02
MET 000/03:44:50
INTERVALS AVAIL 32765
SYSTEMS BATT LOW

DISPLAY 1

TOTAL DOSE

μRAD	00.00
μREM	00.00
μREM/HR	00.00

IN LAST INTERVAL
TOTL TIME 0/01:47:30
MEMORY BKUP BATT LOW

DISPLAY 2

INTERVAL DOSE

μRAD	0.00
μREM	0.00
μREM/HR	0.00
INTERVAL	100 mSEC

REMAINING TIME IN
INTERVAL 0 SED
MAIN BACKUP BATT LOW

DISPLAY 3

Figure 3. RME-III operating displays.

START OF FILE MET = 002/02:33:17 GMT = 041288 17:03:50:0JI0

MET	GMT	RAD	REM	COUNTS#1	CNT#2	CNT#3	INV
002/O2:33:27:0	041288 17:04:00:0	2.941e-07	2.941e-07	2	0	0	10
002/02:33:37:0	041288 17:04:10:0	1.029e-06	1.029e-06	7	0	0	10
002/02:33:47:0	041288 17:04:20:0	7.353e-07	7.353e-07	5	0	0	10
002/02:33:57:0	041288 17:04:30:0	1.241e-06	5.267e-06	5	1	0	10
002/02:34:07:0	041288 17:04:40:0	5.883e-07	5.883e-07	4	0	0	10
002/02:34:17:0	041288 17:04:50:0	1.177e-06	1.177e-06	8	0	0	10
002/02:34:27:0	041288 17:05:00:0	5.883e-07	5.883e-07	4	0	0	10
002/02:34:37:0	041288 17:05:10:0	2.941e-07	2.941e.07	2	0	0	10
002/02:34:47:0	041288 17:05:20:0	7.353e-07	7.353e-07	5	0	0	10
002/02:34:57:0	041288 17:05:30:0	5.883e-07	5.883e-07	4	0	0	10
002/02:35:07:0	041288 17:05:40:0	4.412e-07	4.412e-07	3	0	0	10

Figure 4a. Count rate data at equatorial latitudes (STS-27)

002/03:45:37:0	041288 18:16:10:0	2.353e-06	2.353e-06	16	0	0	10
002/03:45:47:0	041288 18:16:20:0	3.088e-06	3.088e-06	21	0	0	10
002/03:45:57:0	041288 18:16:30:0	2.647e-06	2.547e-06	18	0	0	10
002/03:46:57:0	041288 18:16:40:0	3.236e-06	3.236e-06	22	0	0	10
002/03:46:17:0	041288 18:16:50:0	2.794e-06	2.794e-06	19	0	0	10
002/03:46:27:0	041288 18:17:00:0	3.088e-06	3.088e-06	21	0	0	10
002/03:46:37:0	041288 18:17:10:0	3.741e-06	7.767e-06	22	1	0	10
002/03:46:47:0	041288 18:17:20:0	3.530e-06	3.530e-06	24	0	0	10
002/03:46:57:0	041288 18:17:30:0	3.530e-06	3.530e-06	24	0	0	10
002/03:47:07:0	041288 18:17:40:0	2.500e-06	2.500e-06	17	0	0	10
002/03:47:17:0	041288 18:17:50:0	3.383e-06	3.383e-06	23	0	0	10
002/03:47:27:0	041288 18:18:00:0	3.377e-06	3.677e-06	25	0	0	10

Figure 4b. Count rate data at high latitudes (STS-27).

2/7:39:38:0	900426 20:13:37:0	1.660e-03	1.963e-03	11058	53	10	10
2/7:39:58:0	900426 20:13:47:0	1.655e-03	2.001e-03	10985	68	8	10
2/7:39:58:0	900426 20:13:57:0	1.701e-03	2.018e-03	11318	61	8	10
2/7:40:8:0	900426 20:14:07:0	1.704e-03	2.063e-03	11310	69	9	10
2/7:40:18:0	900426 20:14:17:0	1.694e-03	2.074e-03	11229	72	10	10
2/7:40:28:0	900426 20:14:27:0	1.741e-03	2.167e-03	11514	81	11	10
2/7:40:38:0	900426 20:14:37:0	1.699e-03	2.036e-03	11286	68	7	10
2/7:40:48:0	900426 20:14:47:0	1.691e-03	2.118e-03	11173	77	13	10
2/7:40:58:0	900426 20:14:57:0	1.659e-03	1.973e-03	11026	69	4	10

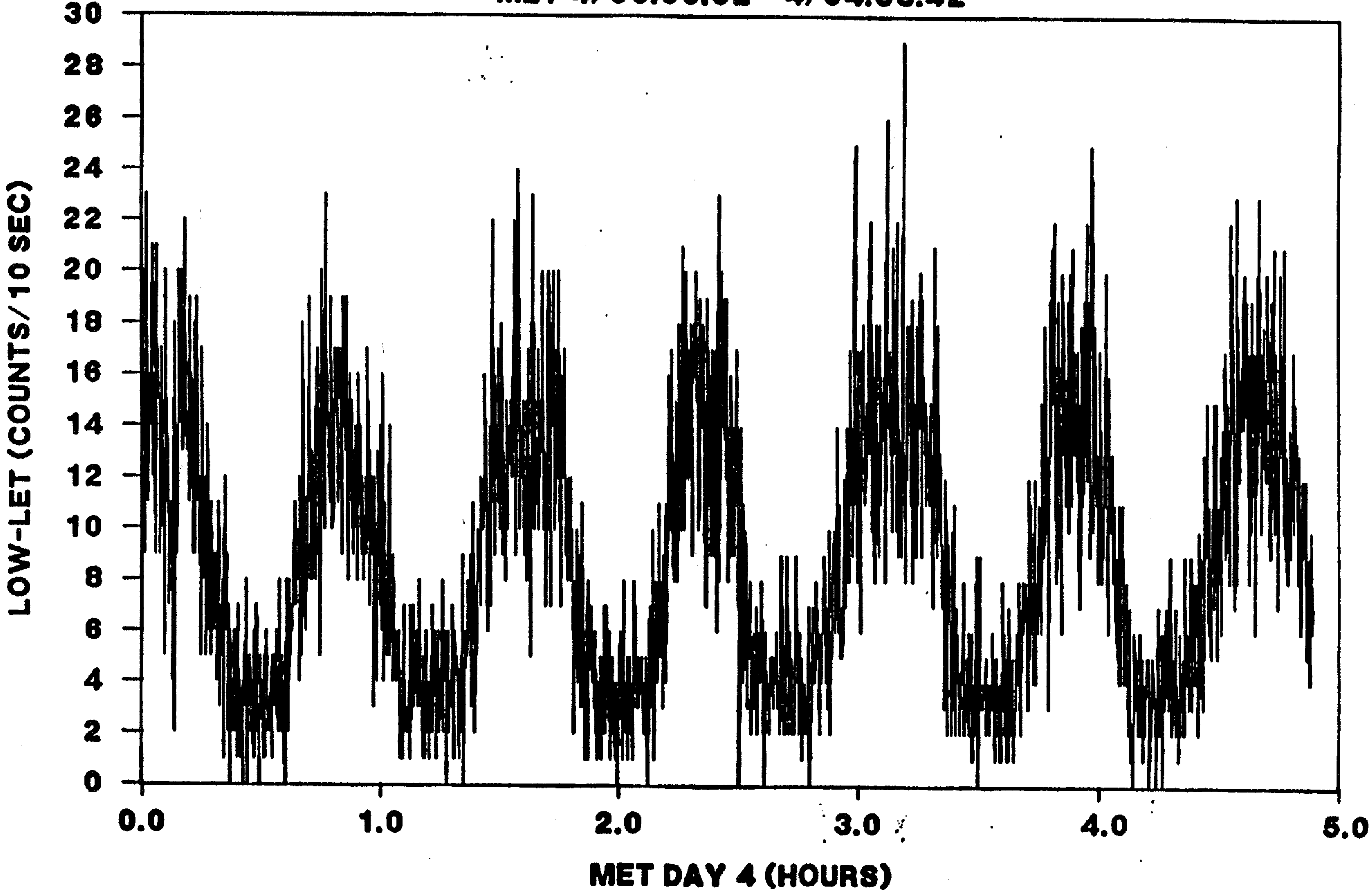
Figure 4c. Count rate data in South Atlantic Anomaly (SAA) measured on (STS-31).

Table 1. RME-III Shuttle Data

Shuttle Mission	Orbital Parameters	RME run Time (hrs.)	RME Total Dose (mrad)	RME Total Dose Equiv. (mrem)	Average Q.F.	Av. PDR Dose (mrad)
*******	*************	*******	**********	*********	******	*******
STS-27	Inc.: 57 deg. Alt.: 450 km circular	101.5	157.9	247.3	1.566	256.4
STS-28	Inc.: 57 deg. Alt.: 300 km circular	116.1	63.4	109.2	1.722	119.9
STS-33	Inc.: 28.5 deg. Alt.: 230 km x x 555 km elliptical	102.7	370.7	427.7	1.154	762.8
STS-36	Inc.: 62 deg. Alt.: 240 km circular	104.6	20.9	33.5	1.603	32.5
STS-31	Inc.: 28.5 deg. Alt.: 685 km circular	112.9	2033.5	2486.5	1.223	1380.9
STS-41	Inc.: 28.5 deg. Alt.: 300 km circular	85.3	12.0	16.1	1.346	17.8

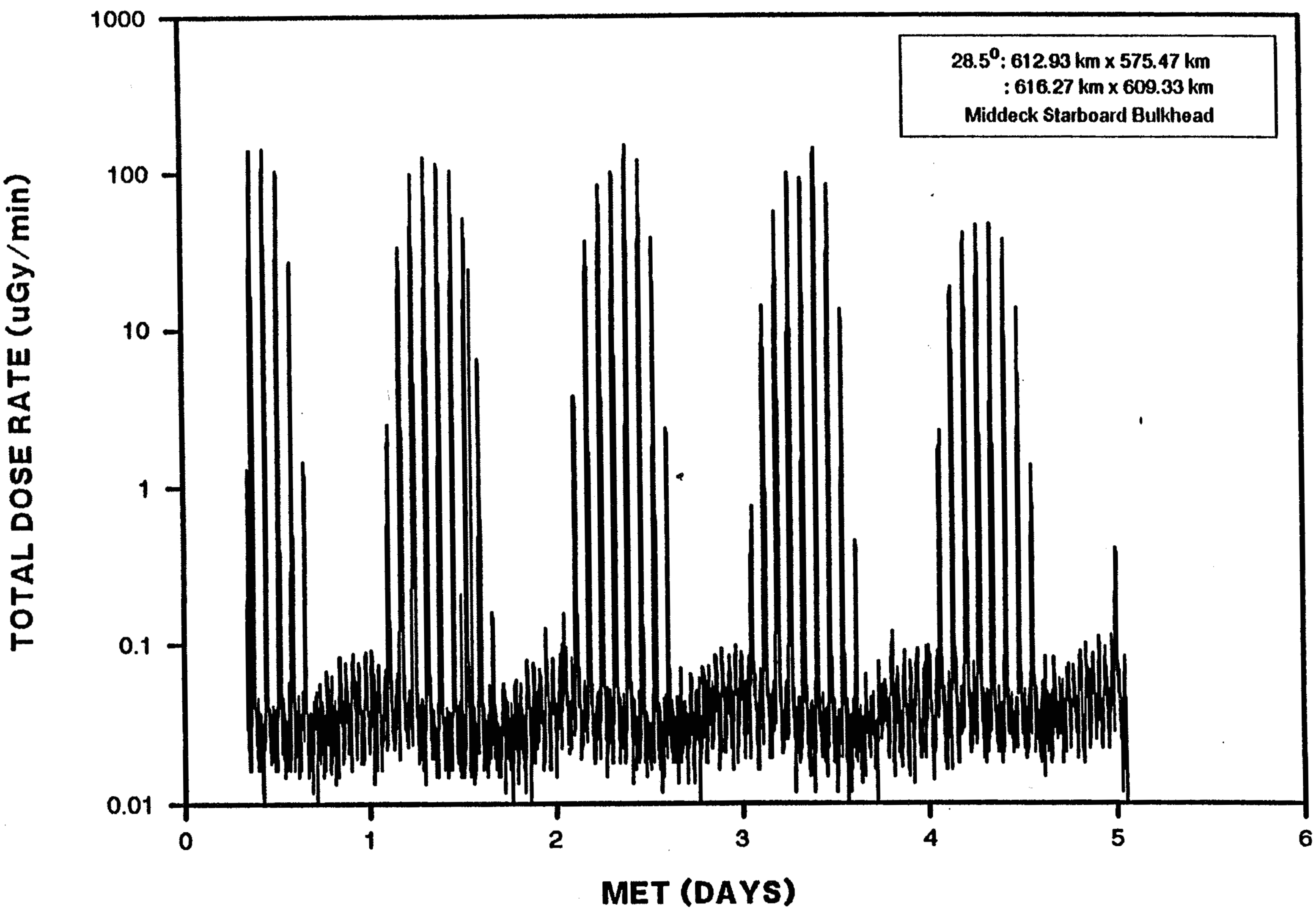
STS-28 RME-III DATA

MET 4/00:00:02 - 4/04:53:42



STS-31 RME-III DATA

MET 0/08:11:36:0 TO 5/03:59:43:0



M. Golightly/NASA JSC

STS-28
RME-III Measured Dose Rate

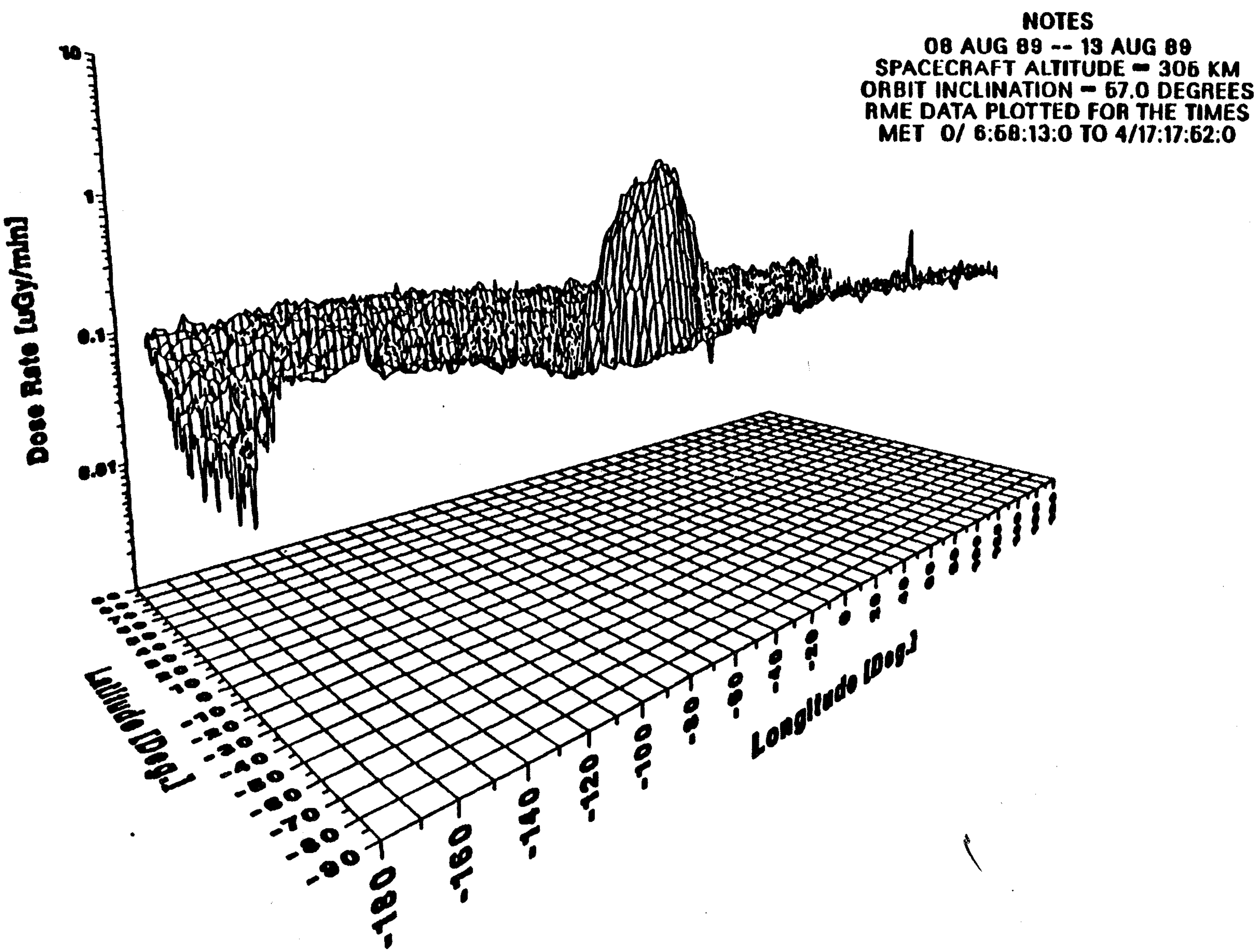
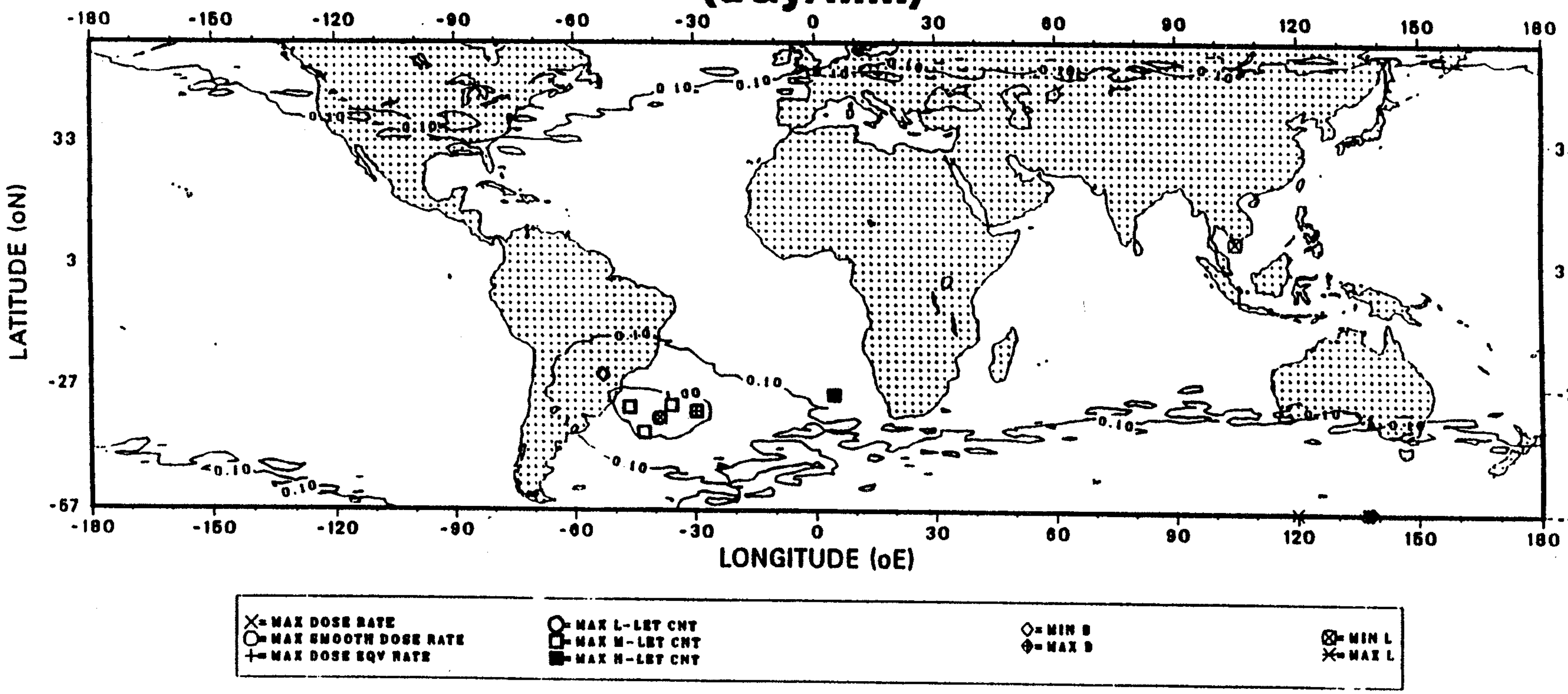


Fig. 1. Isodose contours derived from the STS-28 RME-III data. The contours were derived from 38,000 data points which were smoothed over a 249 x 249 element latitude-longitude array.

STS-28
RME-III Measured Dose Rate
(uGy/min)



NOTES

MISSION DURATION: 08 AUG 89 -- 13 AUG 89
ORBITAL INCLINATION: 57.0 DEGREES / AVERAGE ALTITUDE: 302 KM
PLOTTED DATA SET DURATION: MET 0/ 6:56: 3 TO 4/17:17:52
RECORDS IN DATA SET: 38240
MAP PROJECTION: CYLINDRICAL EQUIDISTANT
DATA SMOOTHING GRID (LATxLON): 249 x 249
FLUX CONTOURS: uGy/min AT x10 INTERVALS
MAGNETIC FIELD MODEL: IGRF 1985
MAGNETIC FIELD EPOCH: 1989.7

DATA SUMMARY

MAXIMUM DOSE RATE: 2.40 uGy/min
MAXIMUM SMOOTHED DOSE RATE: 1.94 uGy/min
MAXIMUM DOSE EQUIVALENT RATE: 4.17 uSv/min
MAXIMUM L-LET COUNT RATE: 262 cnts/10 sec
MAXIMUM M-LET COUNT RATE: 7 cnts/10 sec
MAXIMUM H-LET COUNT RATE: 3 cnts/10 sec
RANGE OF B VALUES (gauss): 0.2067 -- 0.5768
RANGE OF L VALUES (Earth Radii): 0.95 -- 10.76

M.J. GOLIGHTLY/NASA JSC SN31/(713) 483-6190

Spacecraft Interactions and the APE Experiment

R. A. Viereck, E. Murad and C. P. Pike / Phillips Lab, Boston
S. B. Mende, and G. B. Swenson / Lockheed, Palo Alto
F. Culbertson, R. Springer, S. Lucid, K. T. Thornton / NASA

Outline

Purpose:	To Understand the optical emissions that arise from the interactions between spacecraft in low earth orbit and the ambient atmosphere.
Procedures:	Instrument Description Experiment Description
Results:	Shuttle Plume Emission Shuttle Glow Airglow
Conclusions:	Lessons Learned Future Experiments

Flight Experiments Technical Interchange Meeting

Purpose:

Satellite Assessment:

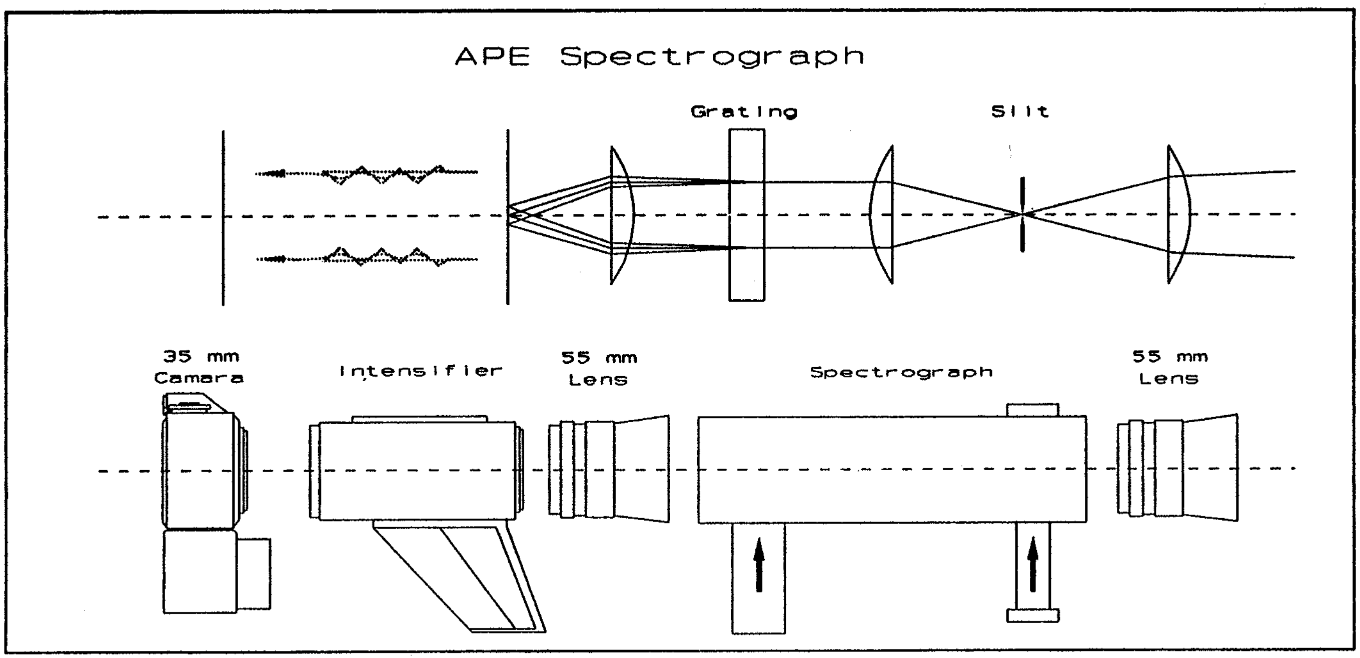
Optical Signatures for Targeting and tracking

Understanding the Physics that give rise to the spectral, spatial, and temporal variations in the optical emissions around space craft.

Characterizing the spectral, spatial, and temporal variations in the background airglow emissions

Instrument Components:

Lenses (2)	50 mm, f-1.2. (NASA Provided Hardware)
Slit	25 by 0.03 mm, 1.1 nm resolution
Grating	600 lines/mm, blazed at ~500 nm
Intensifier	S-20, 25 mm photocathode
Camera	Nikon 35 mm (NASA Provided Hardware)
Film	1600 ASA, Kodak Ektapress, color negative



Experiment Description:

Thruster Plumes
- Spectral Assessment
- Near vs far field observations
- Ram angle Dependance*
- Comparison with other space observations and with ground*

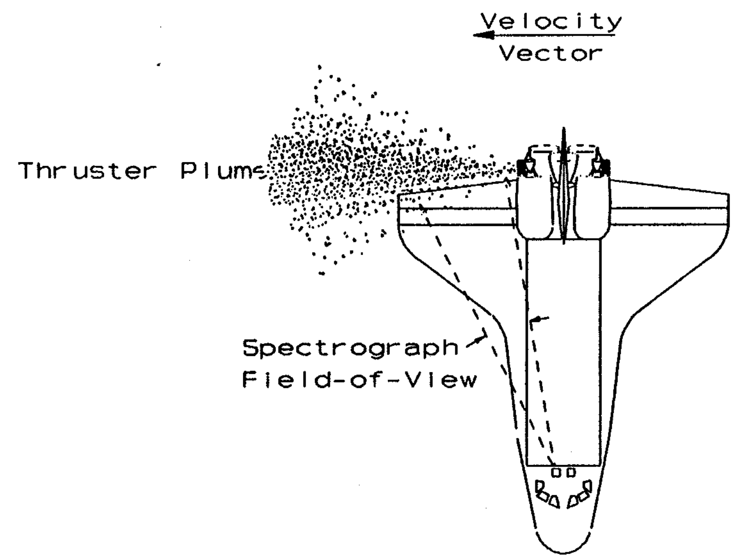
Shuttle Glow
- Spectral Assessment
- Thruster Effects*
- Altitude Effects*

Airglow Observations
- Spectral Assessment
- Identification of Emissions and emission altitudes

* Future Experiment or Analysis

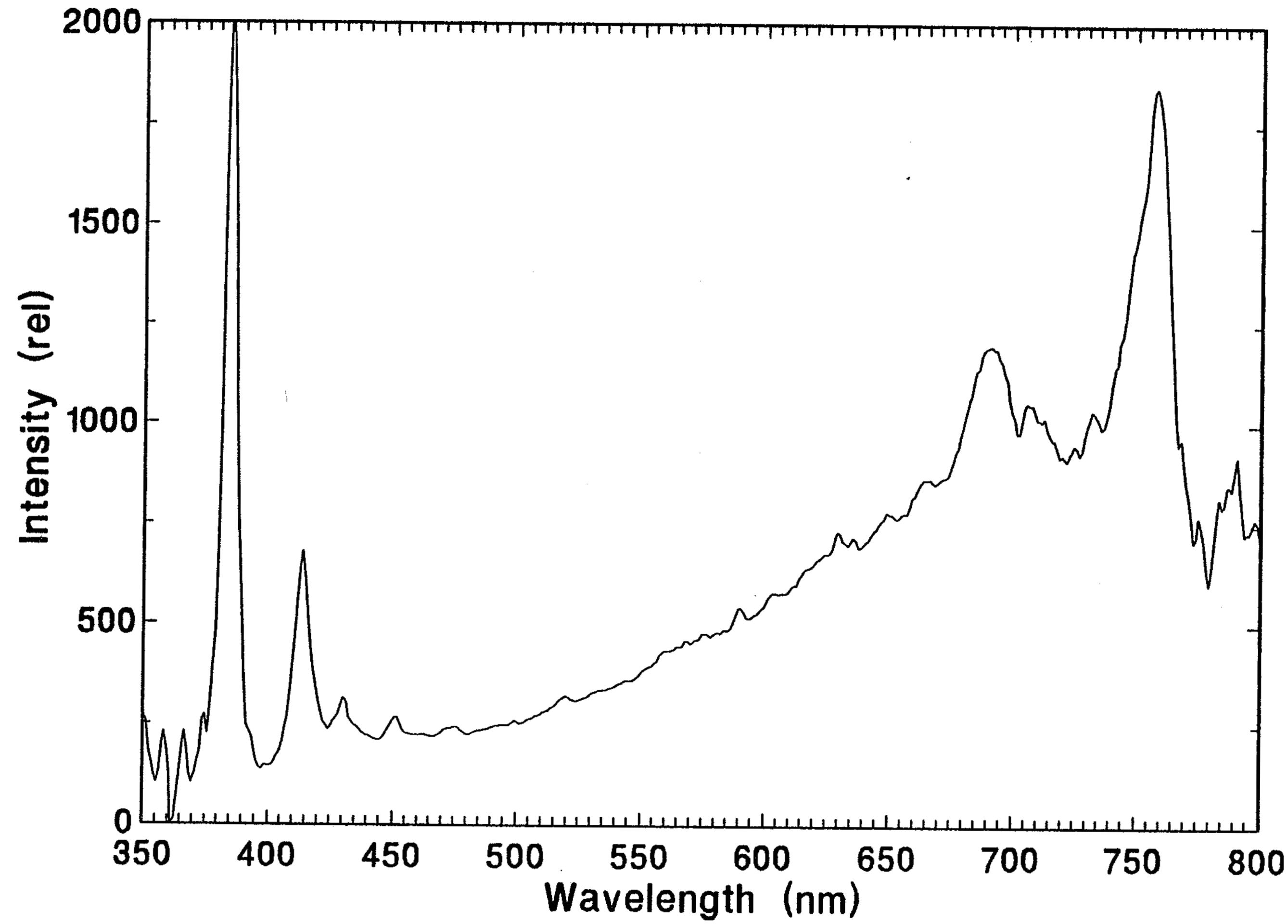
Procedures:

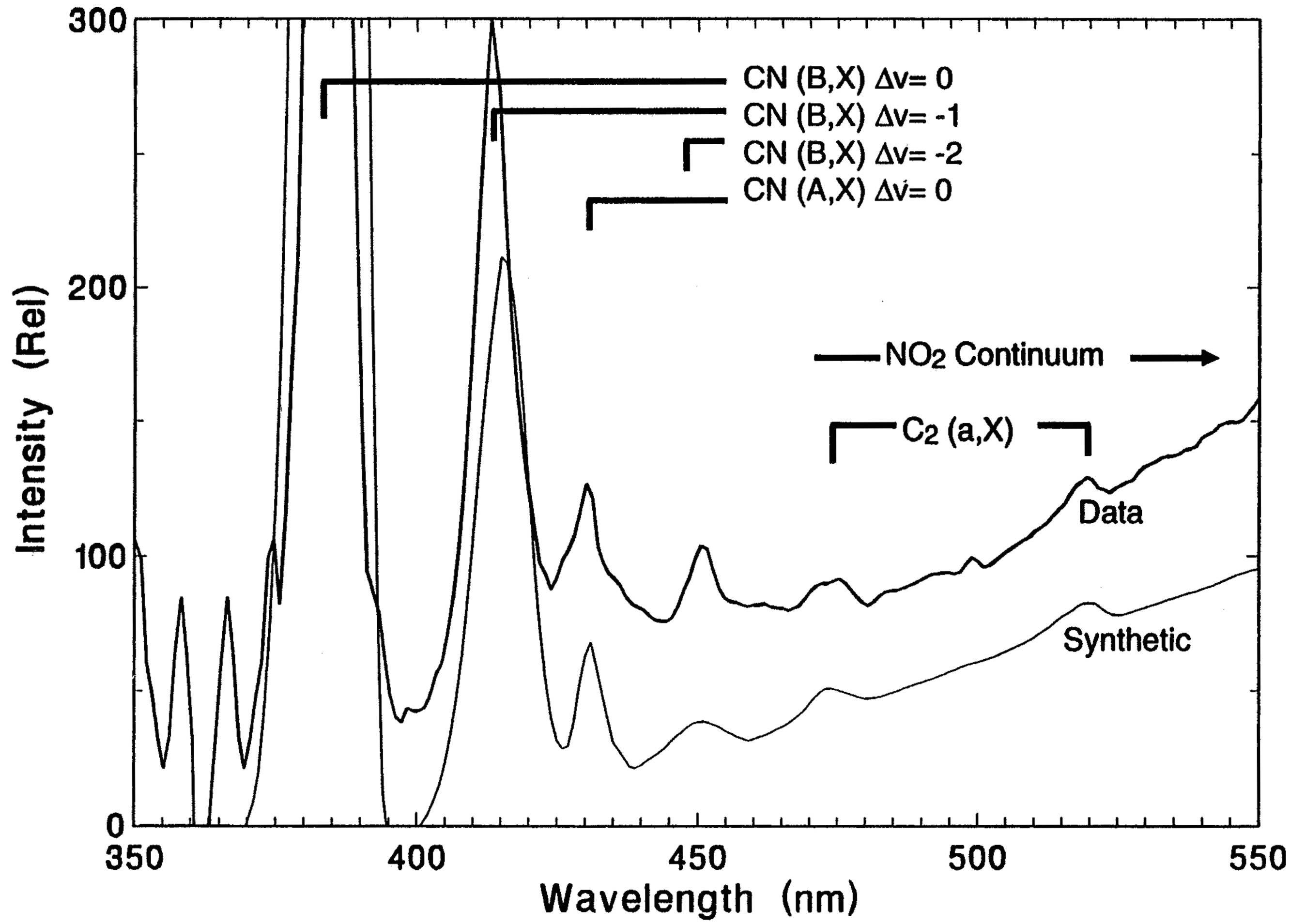
Spectrograph mounted in aft-flight-deck.
Aimed towards tail of orbiter
Slit parallel to centerline of plume
Series of 4 sec exposures



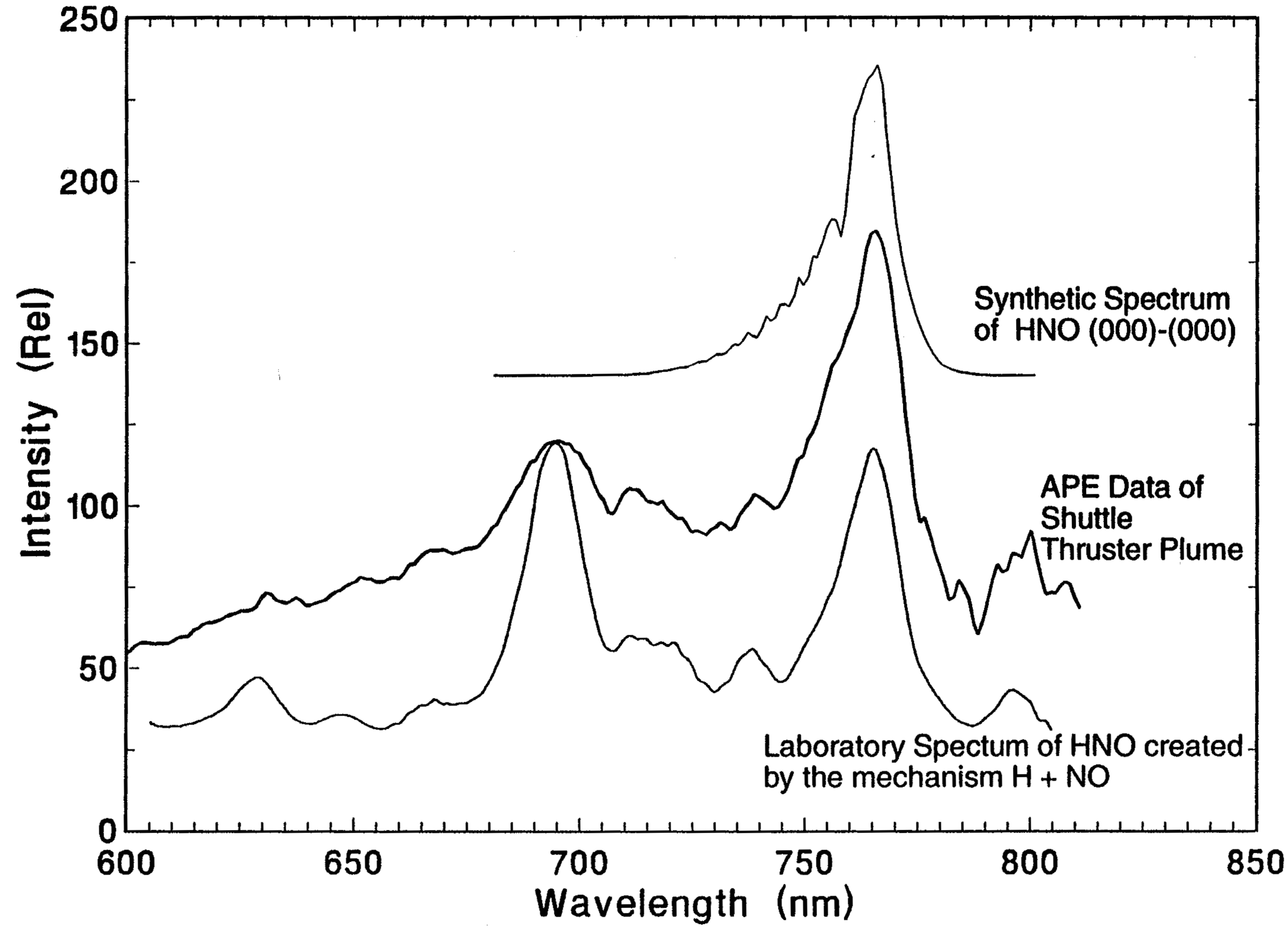
Condition:

Orbiter Altitude	240 km (night)
Orbiter Attitude	Bay to Earth
	Right wing to ram
Thruster	From OMS pod into ram
	820 lb Thruster
	Fired for 4 seconds
	MMH and H_2O_4 film cooled


2000
1500
1000
500
0
Intensity (rel)
350
400
450
500
550
600
650
700
750
800
Wavelength (nm)



CN (B,X) Δv= 0
CN (B,X) Δv= -1
CN (B,X) Δv= -2
CN (A,X) Δv= 0
NO2 Continuum
C2 (a,X)
Data
Synthetic
Intensity (Rel)
Wavelength (nm)
0
100
200
300
350
400
450
500
550


Synthetic Spectrum
of HNO (000)-(000)
APE Data of
Shuttle
Thruster Plume
Laboratory Spectum of HNO created
by the mechanism H + NO
Intensity (Rel)
Wavelength (nm)
0
50
100
150
200
250
600
650
700
750
800
850

Results:

Thruster Plumes

Spectral Features Identified: CN (B→X) Δv=0,-1,-2
CH (B→X) Δv=0
C_2 (A→X) Swan Bands
NO_2 continuum
HNO^*
Na
O^1D and O^1S^*

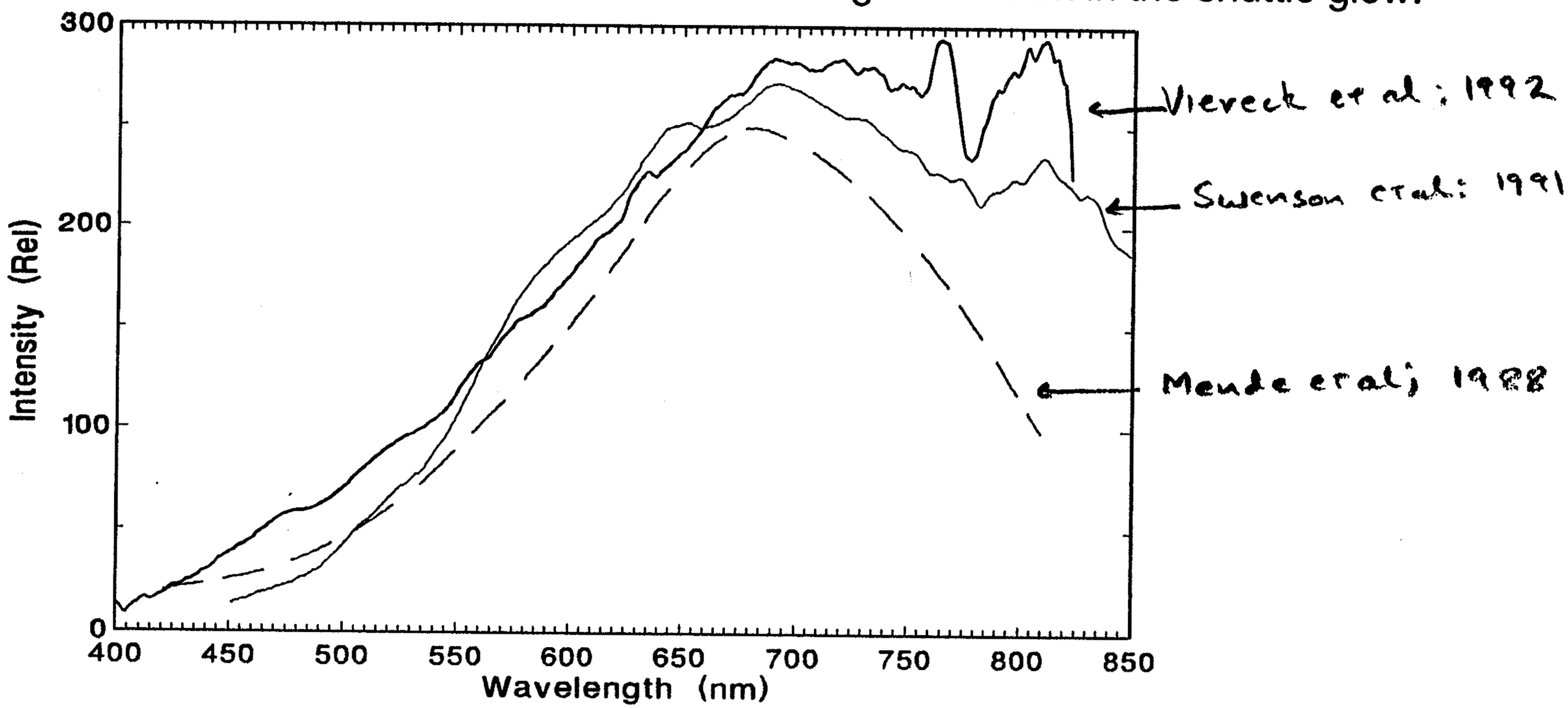
Spatial Features:

CN, CH, C_2, Na	- Decrease Rapidly from nozzle, Down to zero within 1 meter
NO_2 and HNO	- Rise and peak at 1 meter, then fall to 20% at 3 meters
O^1D and O^1S	- Seem to get stronger with increasing distance. O^1S more evident in ram burns than in wake burns.

Flight Experiments Technical Interchange Meeting

Shuttle Glow

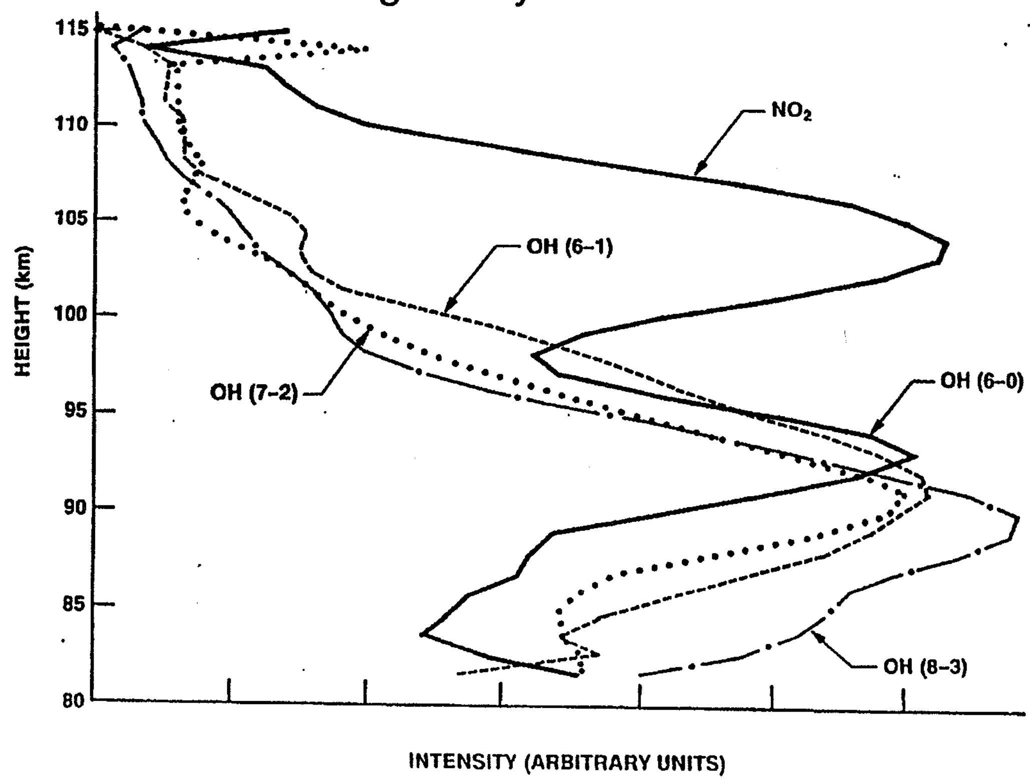
Confirmation of NO_2 as the primary emitting species in the visible
Proper calculation of window transmission resulting in red-shift in the shuttle glow.



Flight Experiments Technical Interchange Meeting

Airglow

Identification of NO_2 in the atmosphere.
Observation of a Double airglow layer.



Conclusions:

Lessons Learned

Small mid-deck locker experiments provide a short turn around and a very high return for the amount of effort.

Shuttle is a difficult platform from which to do science.

- Safety requirements
- Large attitude dead-band and uncertainty
- Windows with poor optical quality for observations

Future Experiments

Several experiments planned for the next 12 months.

- Further map the spatial and temporal distribution of plumes
- Extend the wavelength observations into the UV and IR (APE-C)
- Speed up the data rate by using Video instead of Film.
- Collaborate the observations with other experiments

THE ORBITER EXPERIMENTS (OEX) PROGRAM

David A. Throckmorton
NASA Langley Research Center

NASA/DOD Flight Experiments
Technical Interchange Meeting

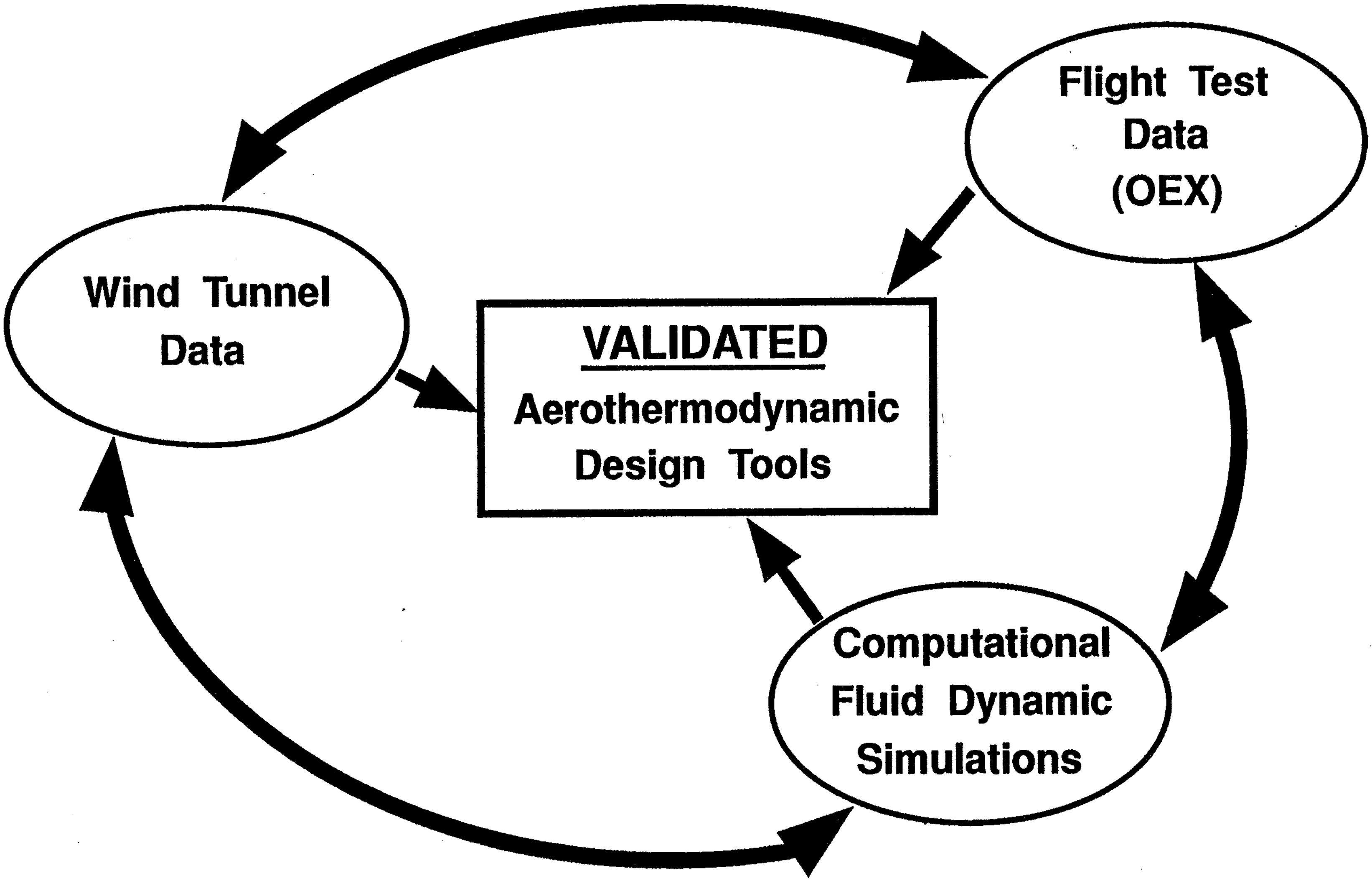
October 7, 1993
Monterey, CA

OEX PROGRAM OBJECTIVE

". . . . to obtain research quality flight data for augmentation and advancement of space transportation technologies. This includes the validation and advancement of analytical theories and of ground-test methods and techniques."

OEX Project Plan

AEROTHERMODYNAMIC DESIGN TOOL DEVELOPMENT AND VALIDATION



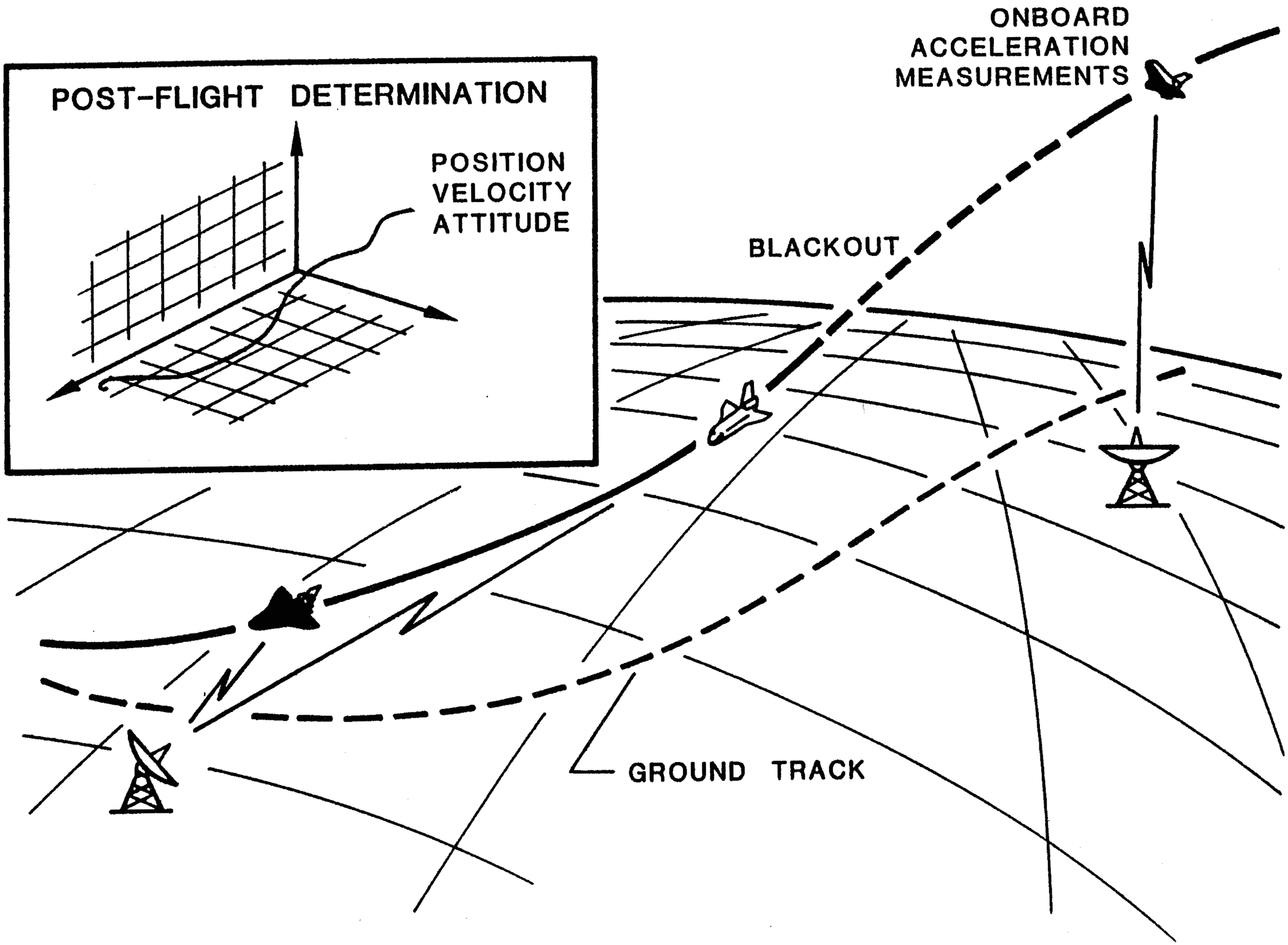
AERODYNAMIC FLIGHT RESEARCH DATA REQUIREMENTS

- Freestream Environment (Including Vehicle Attitude)
- Aerodynamic Forces and Moments
- Surface Pressure and Heat Transfer

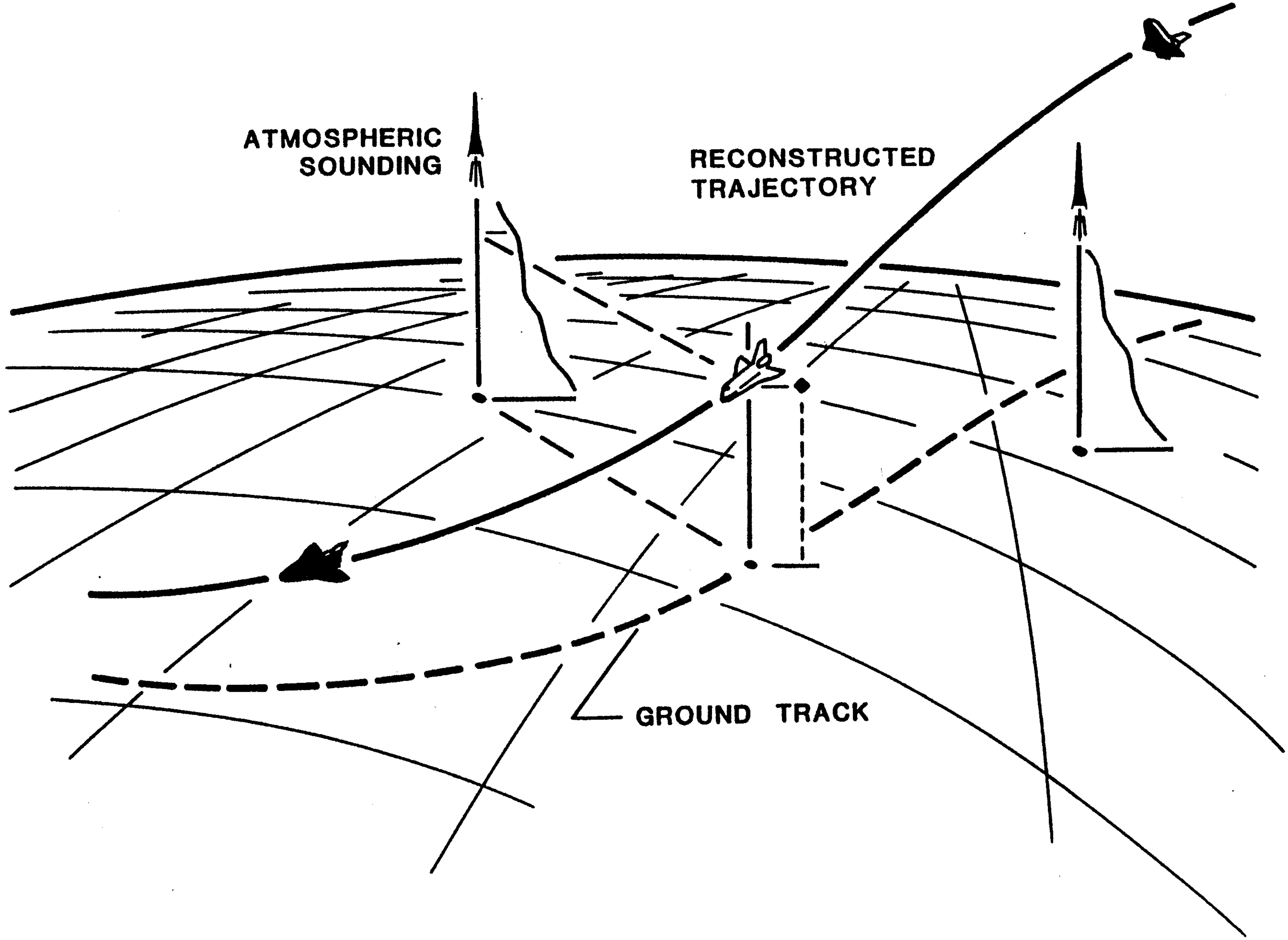
FREESTREAM ENVIRONMENT

- o Best Estimate of Trajectory
- o Shuttle Entry Air Data System (SEADS)
- o Shuttle Upper Atmosphere Mass Spectrometer (SUMS)

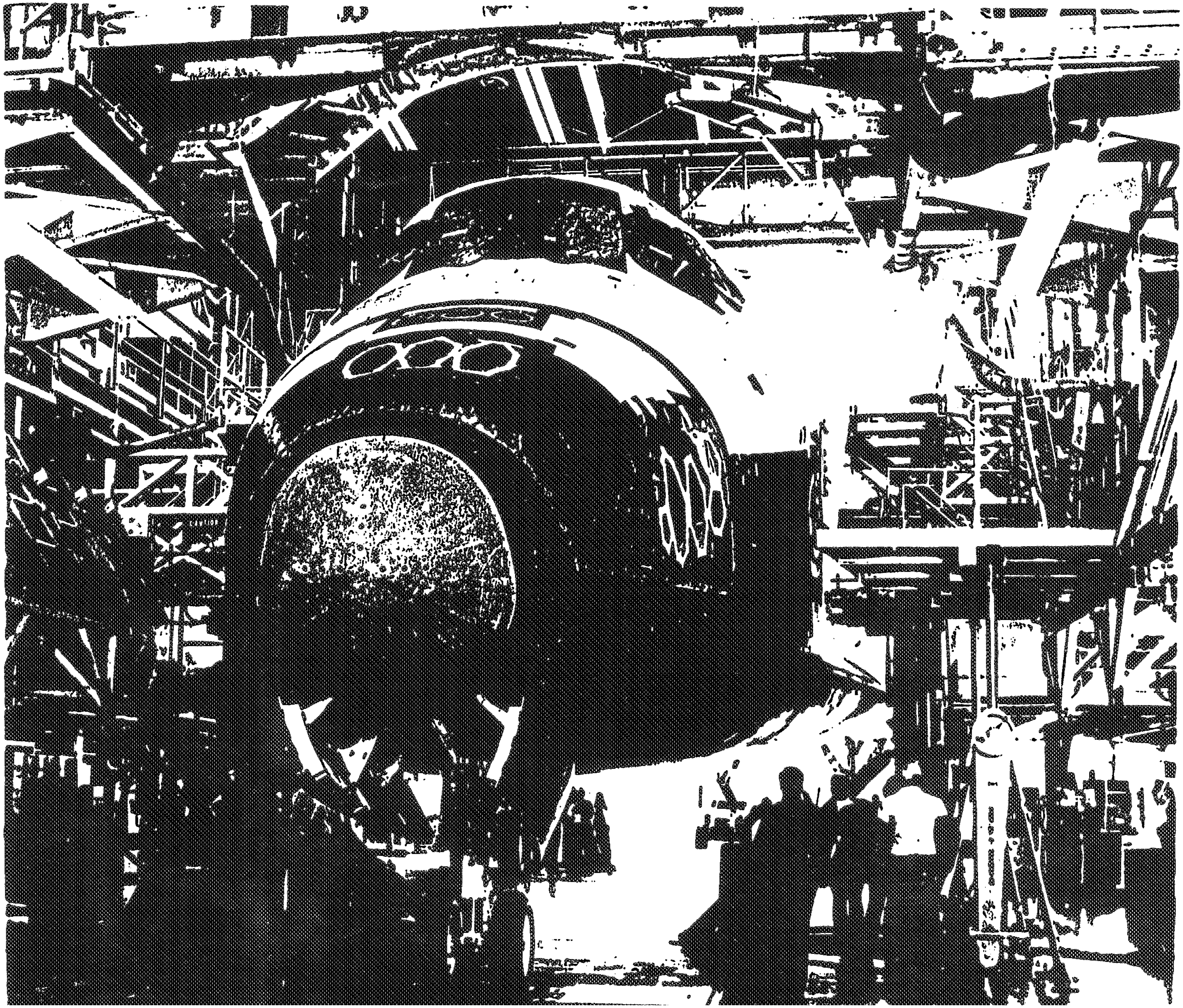
TRAJECTORY RECONSTRUCTION



ATMOSPHERIC RECONSTRUCTION

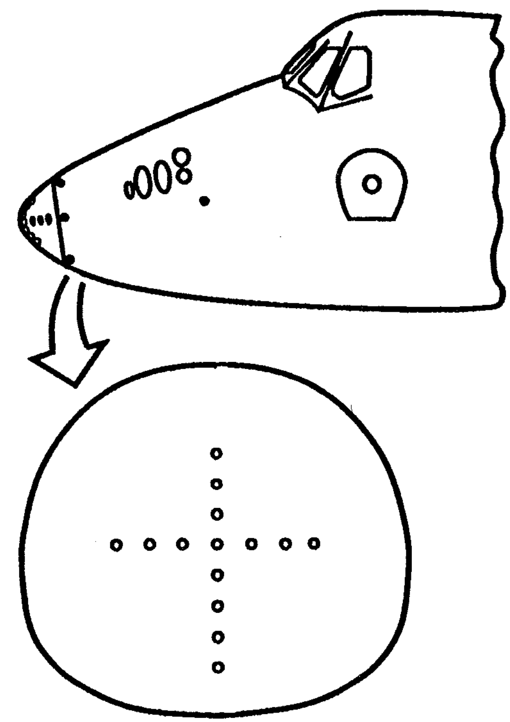


SHUTTLE ENTRY AIR DATA SYSTEM (SEADS)

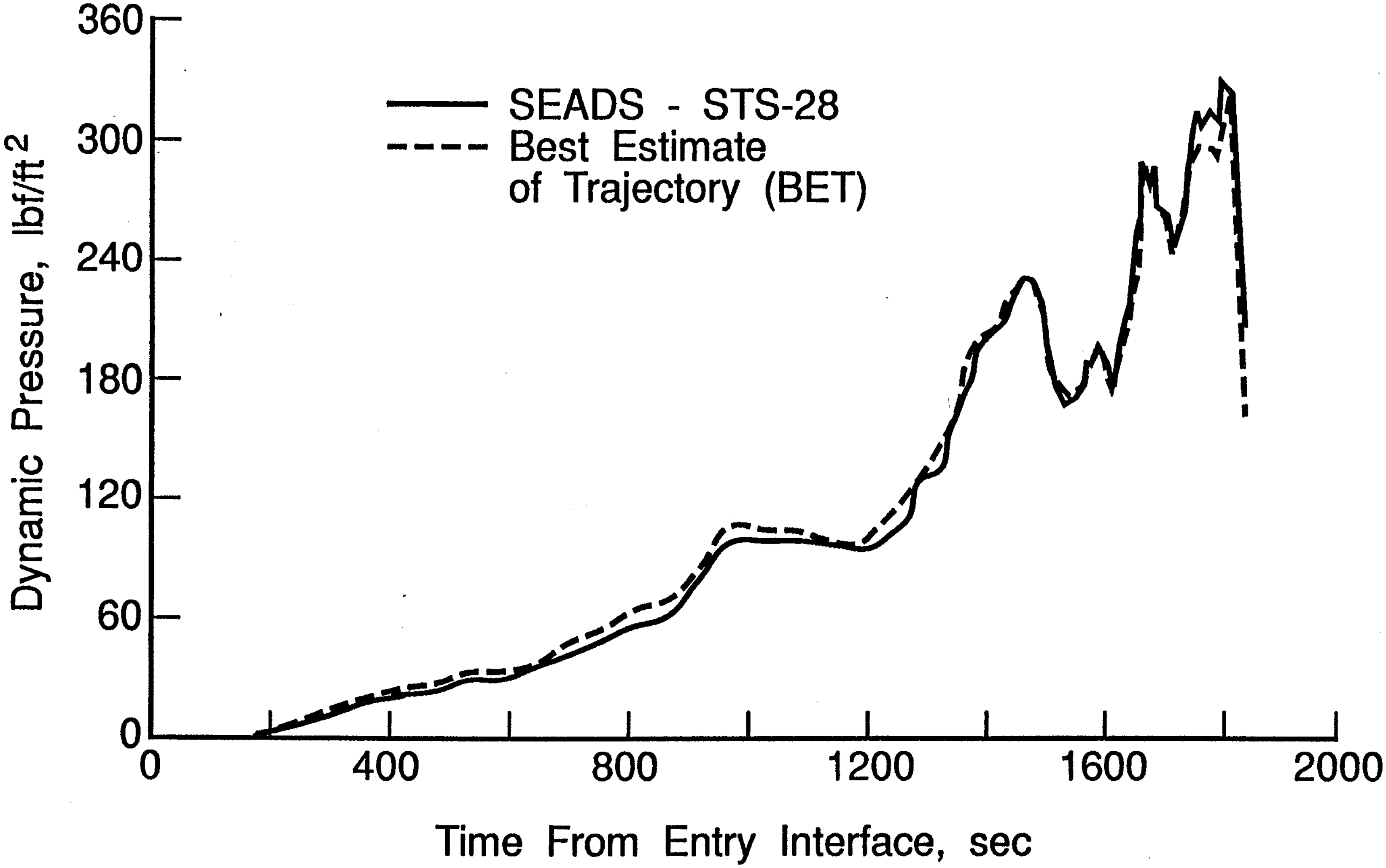


SHUTTLE ENTRY AIR DATA SYSTEM SCHEMATIC

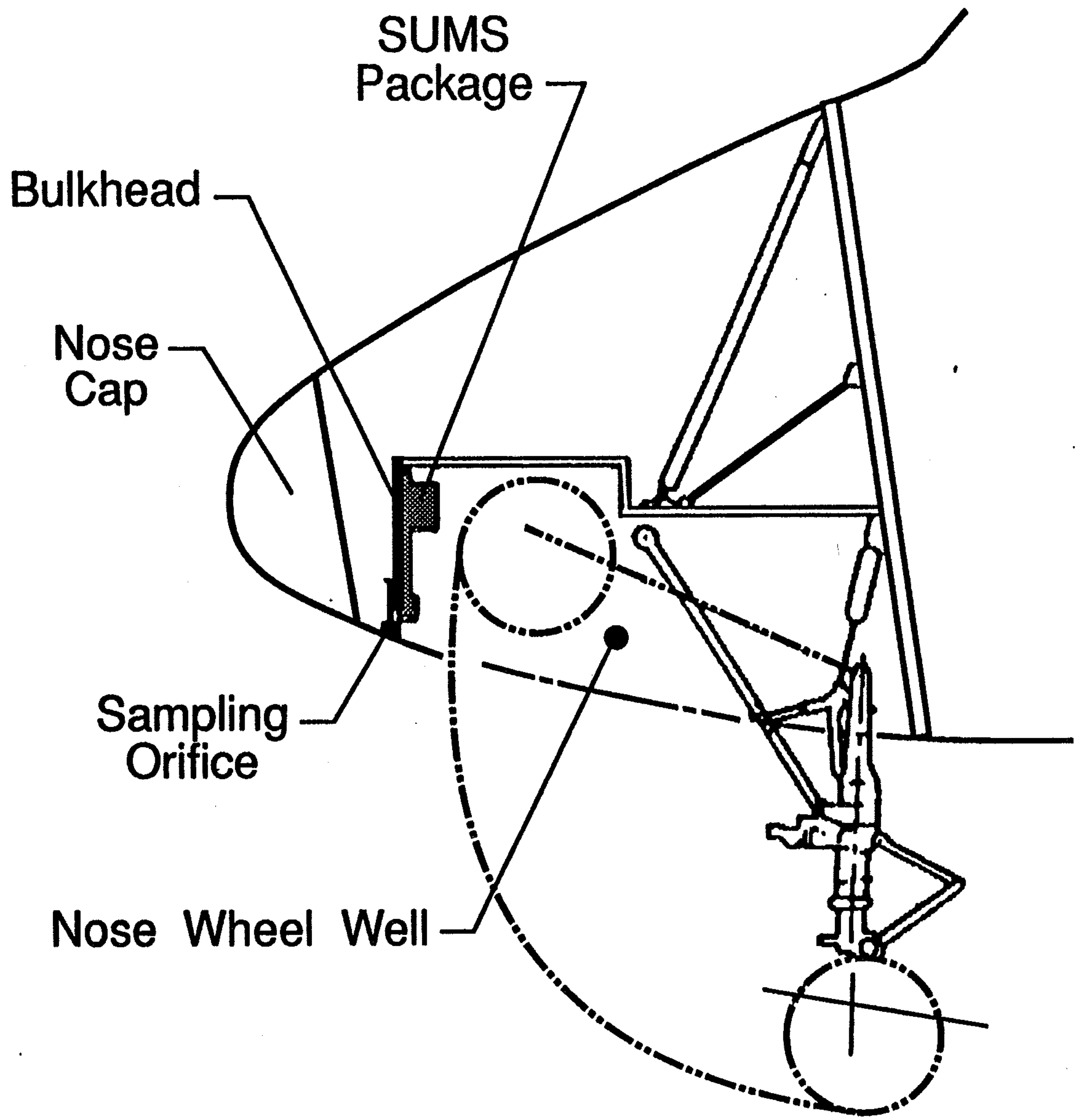
- ∘ Nosecap Orifices (14)
- • "Static" Orifices (6)



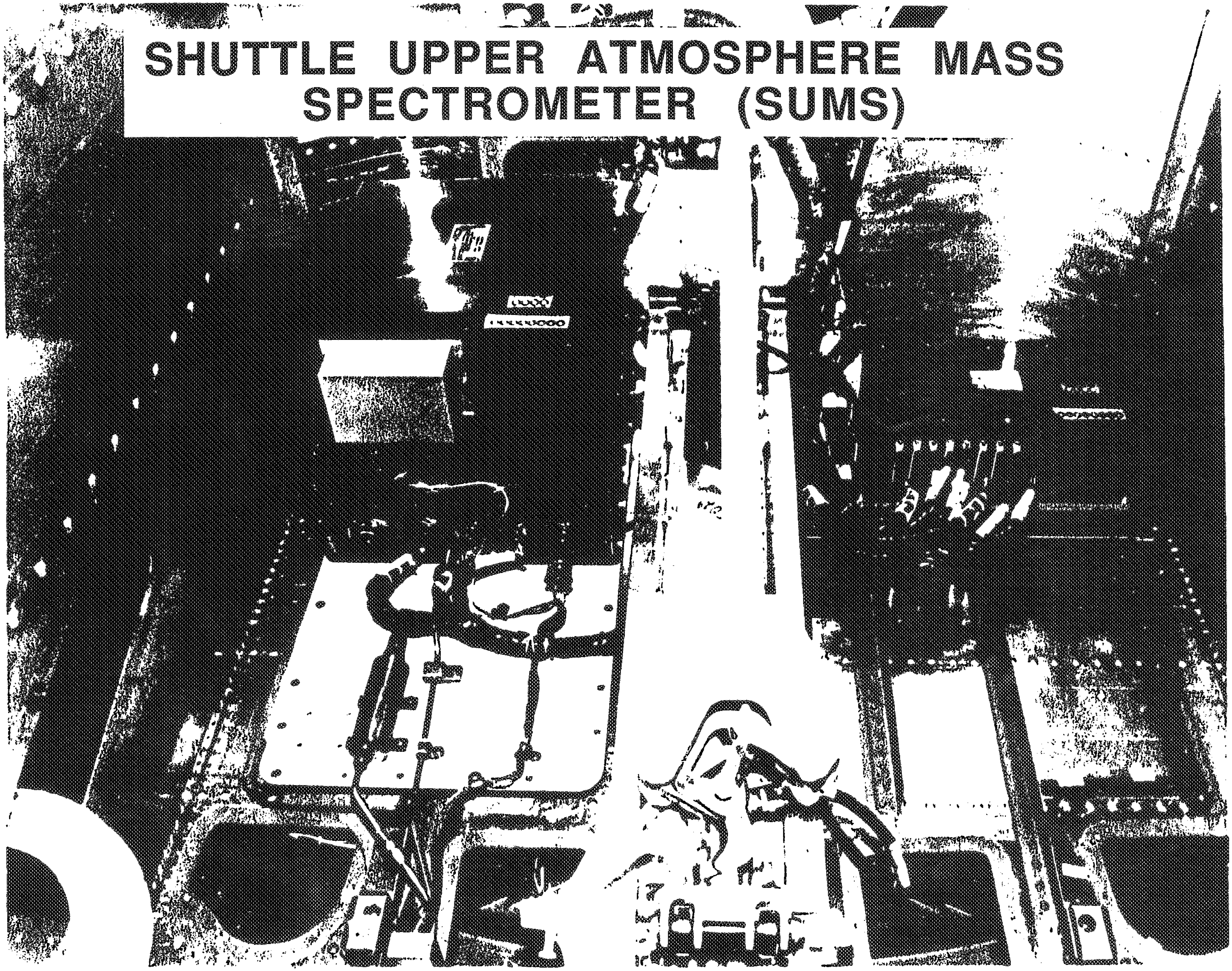
SEADS RESULTS CONFIRM ACCURACY OF BEST ESTIMATE OF TRAJECTORY (BET)



LOCATION OF SUMS ON THE SHUTTLE ORBITER



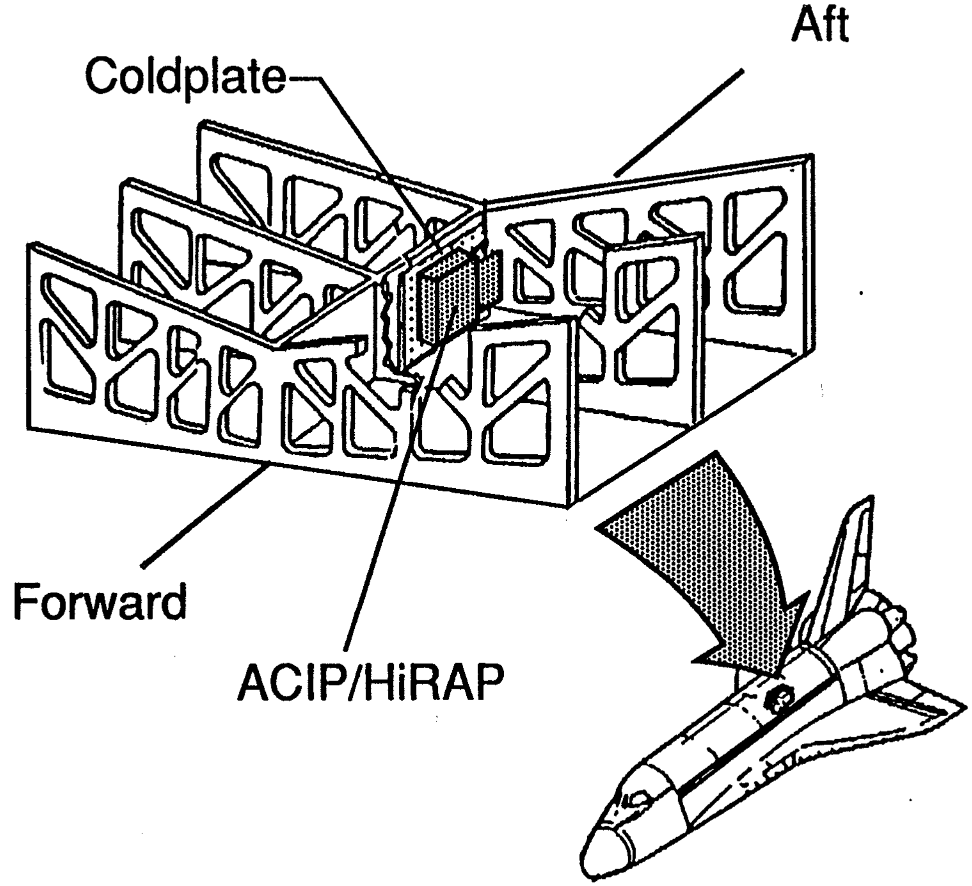
SHUTTLE UPPER ATMOSPHERE MASS SPECTROMETER (SUMS)



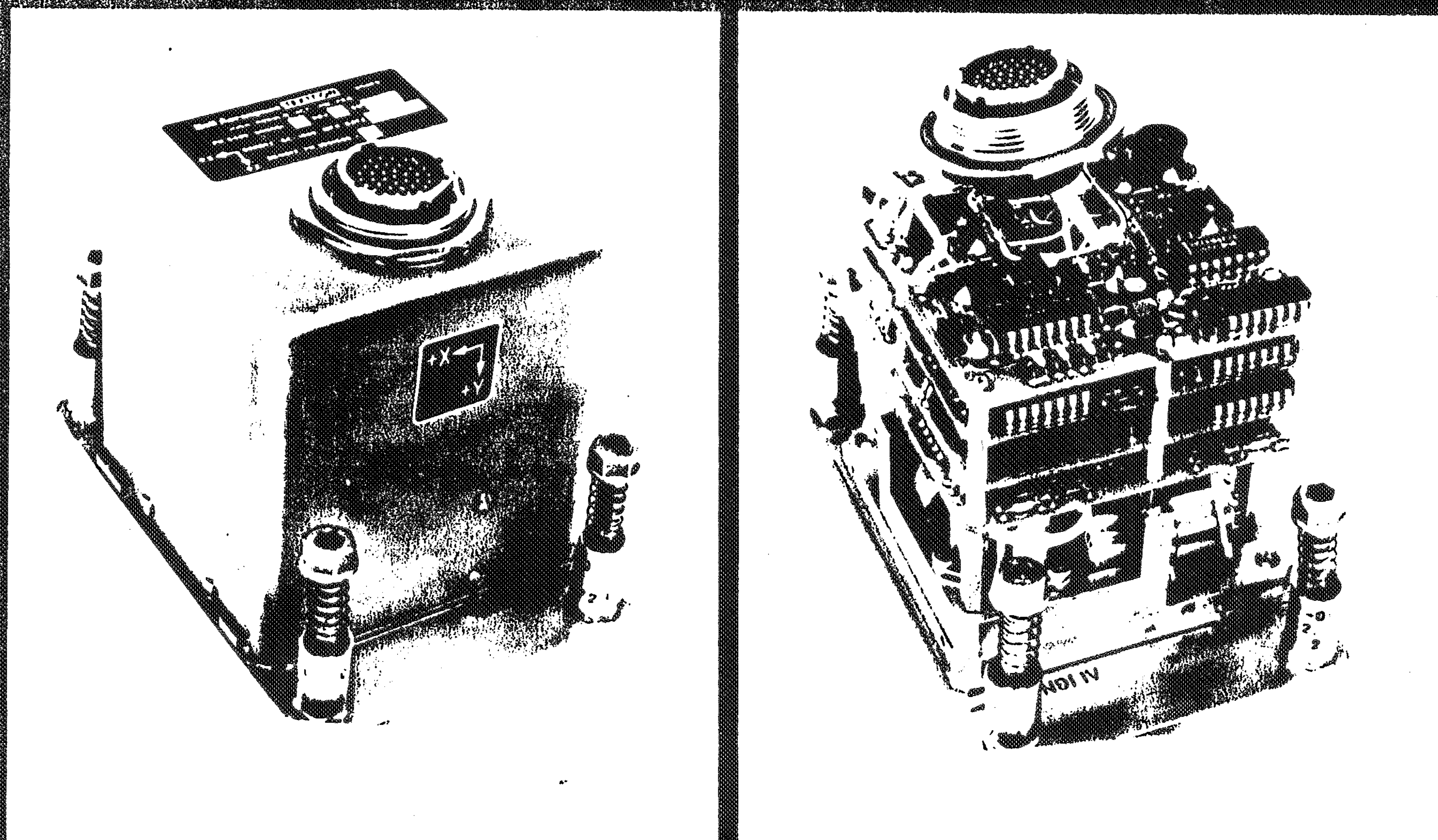
AERODYNAMIC FORCES AND MOMENTS

- o Inertial Measurement Units
- o Aerodynamic Coefficient Identification Package (ACIP)
- o High-Resolution Accelerometer Package (HiRAP)
- o Orbital Acceleration Research Experiment (OARE)

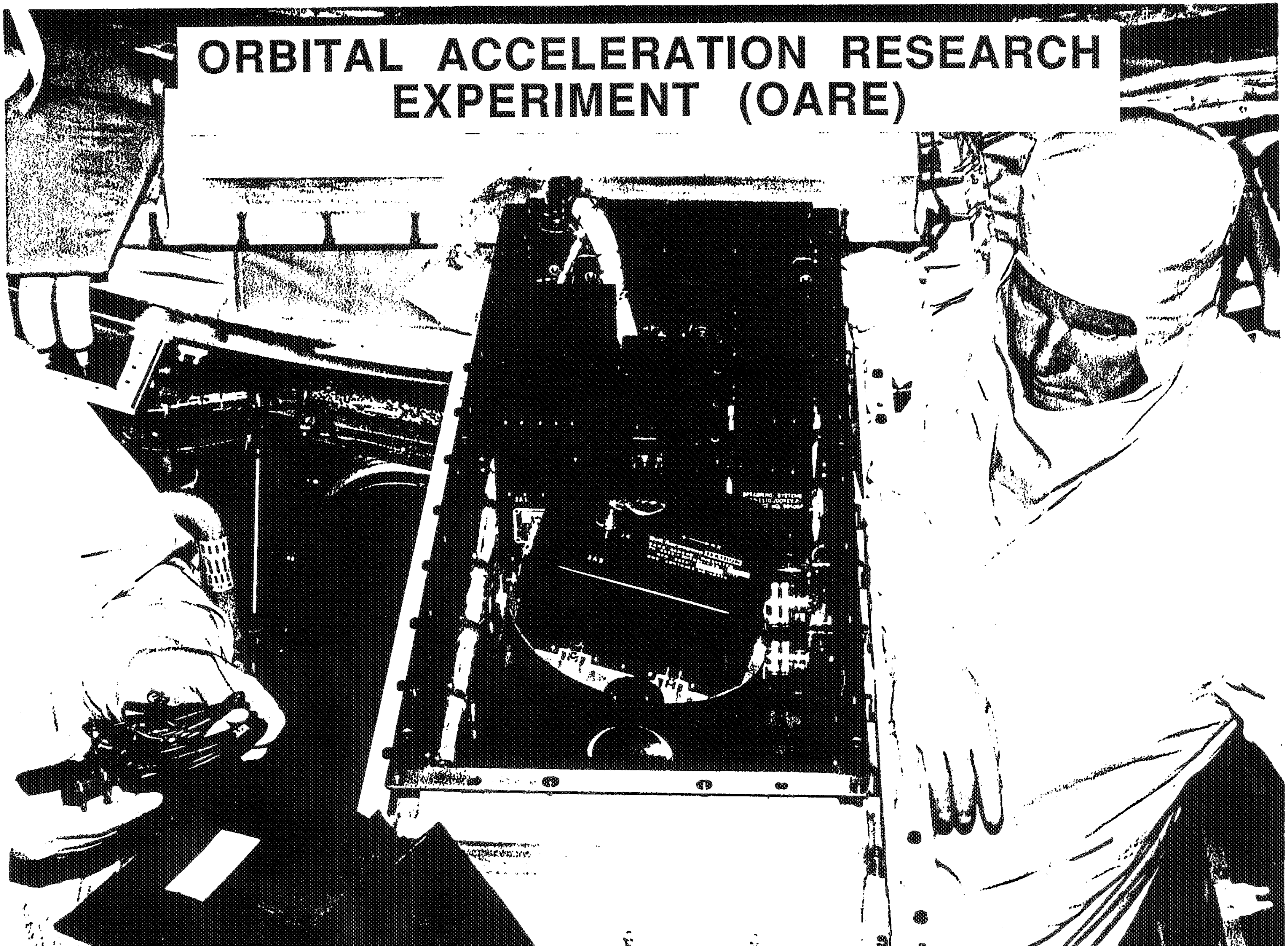
ACIP AND HiRAP LOCATION WITHIN SHUTTLE ORBITER



HiRAP FLIGHT UNIT: SN 002



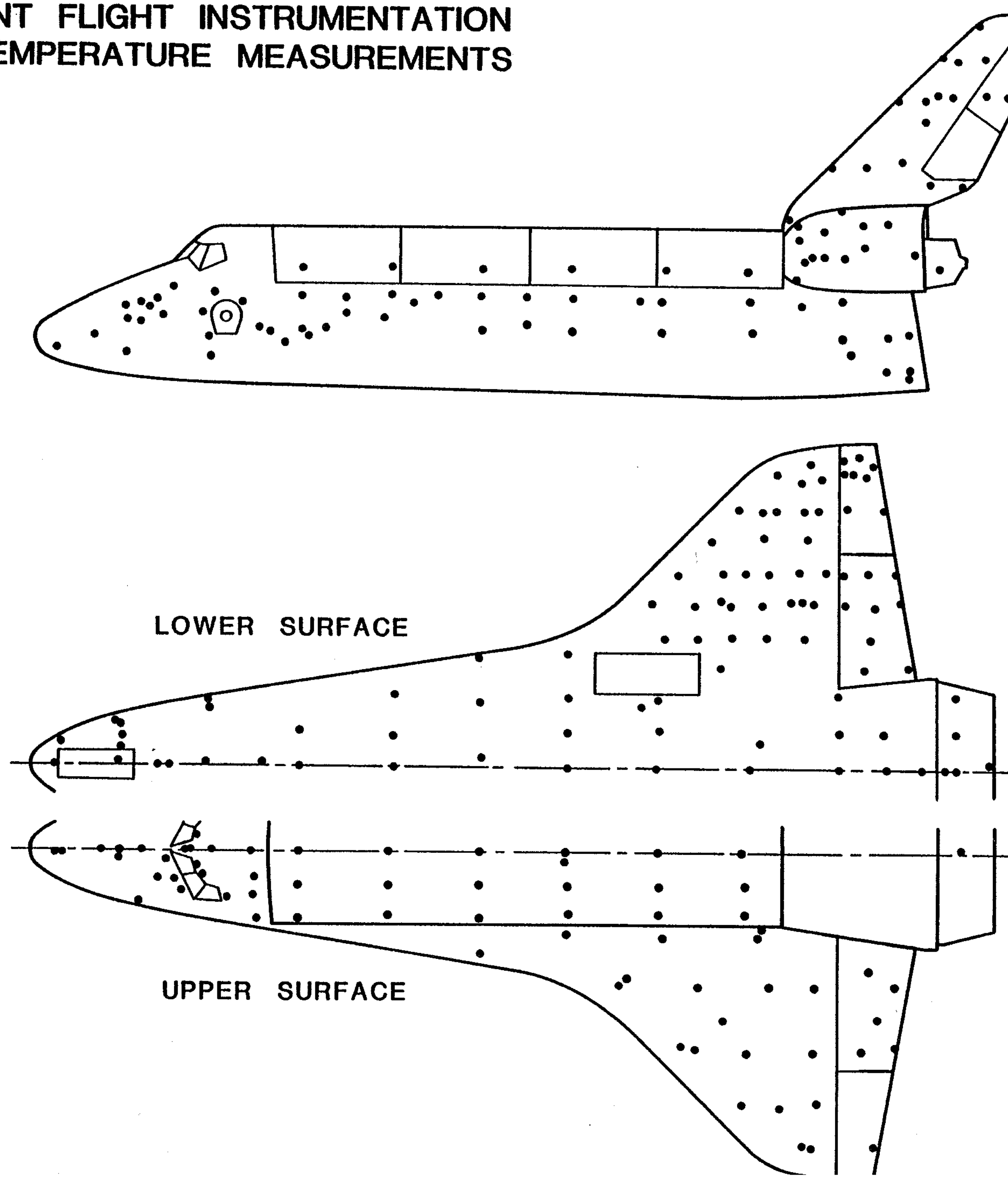
ORBITAL ACCELERATION RESEARCH EXPERIMENT (OARE)



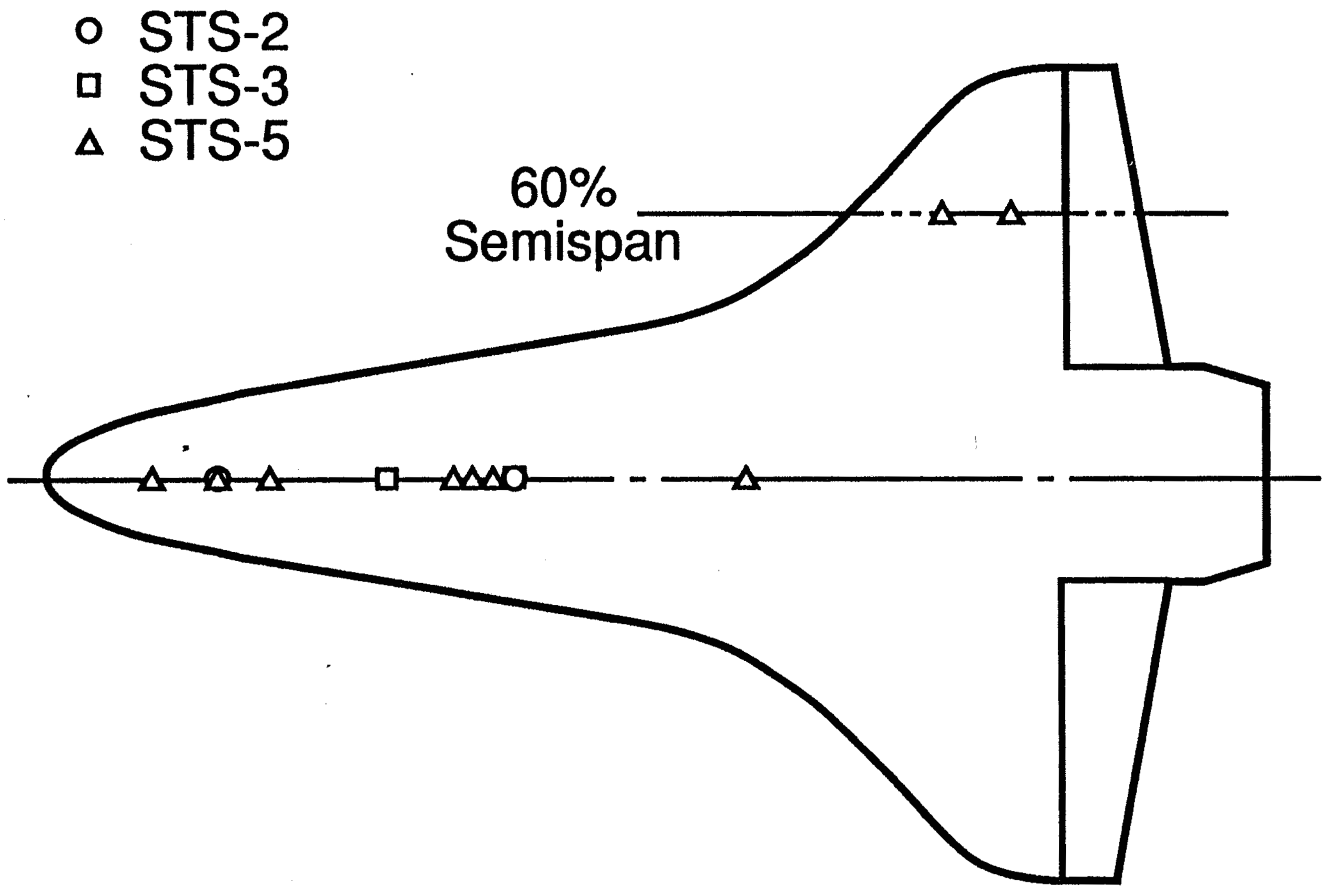
SURFACE PRESSURE AND HEAT TRANSFER

- o Development Flight Instrumentation (DFI)
- o Tile Gap Heating (TGH)
- o Catalytic Surface Effects (CSE)
- o Infrared Imagery of Shuttle (IRIS)
- o Shuttle Infrared Leeside Temperature Sensing (SILTS)
- o Aerothermal Instrumentation Package (AIP)

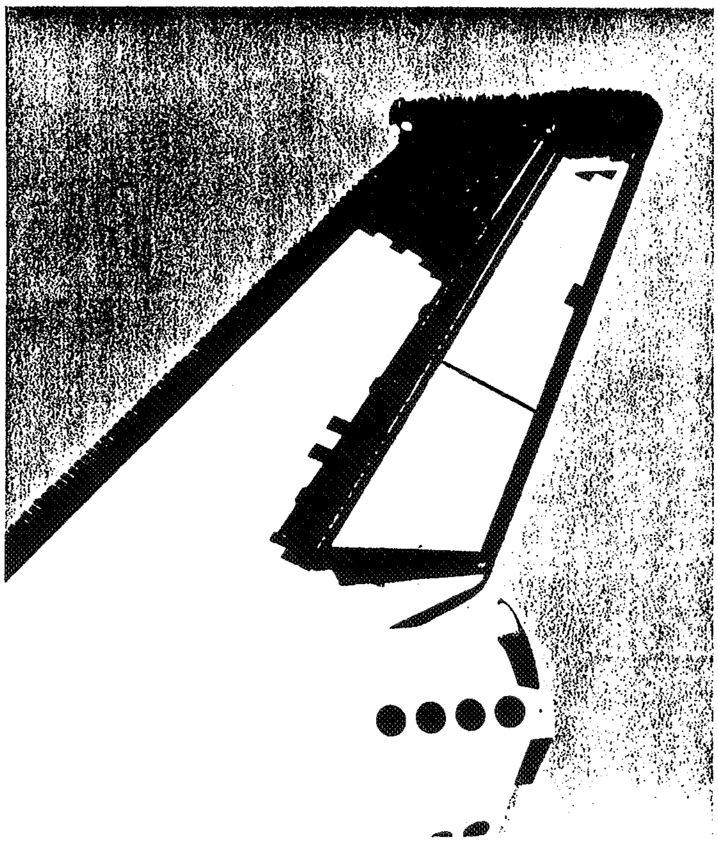
DEVELOPMENT FLIGHT INSTRUMENTATION SURFACE TEMPERATURE MEASUREMENTS



CSE EXPERIMENT MEASUREMENT LOCATIONS

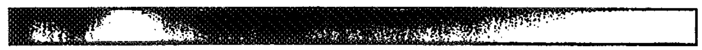


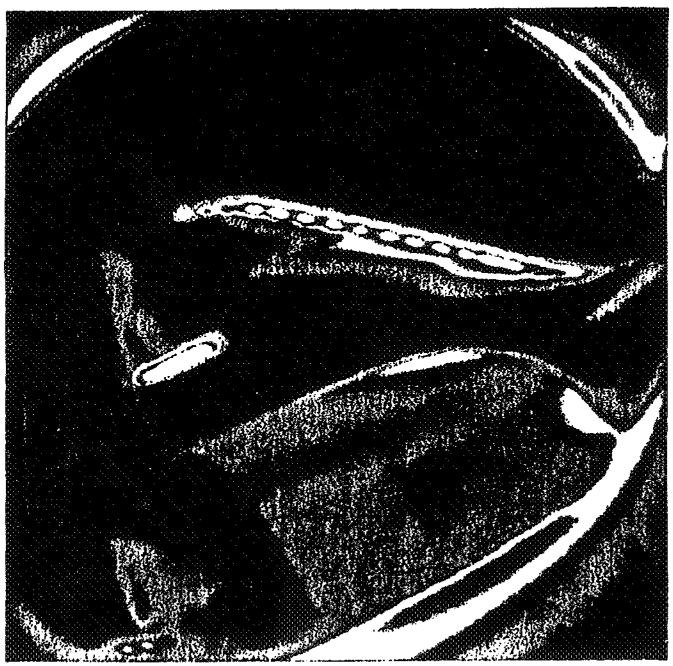
SILTS MODIFICATION TO ORBITER COLUMBIA



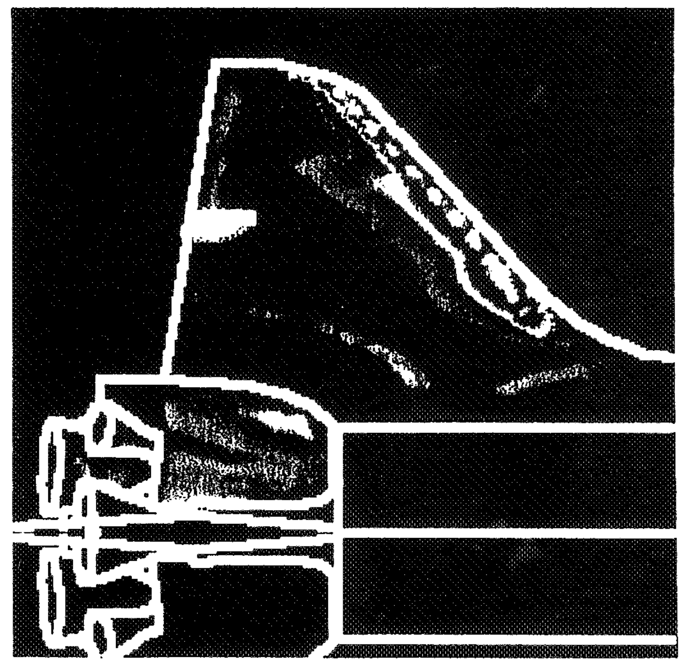
TYPICAL SILTS IMAGE DATA FROM STS-28

Increasing Temperature





Camera View



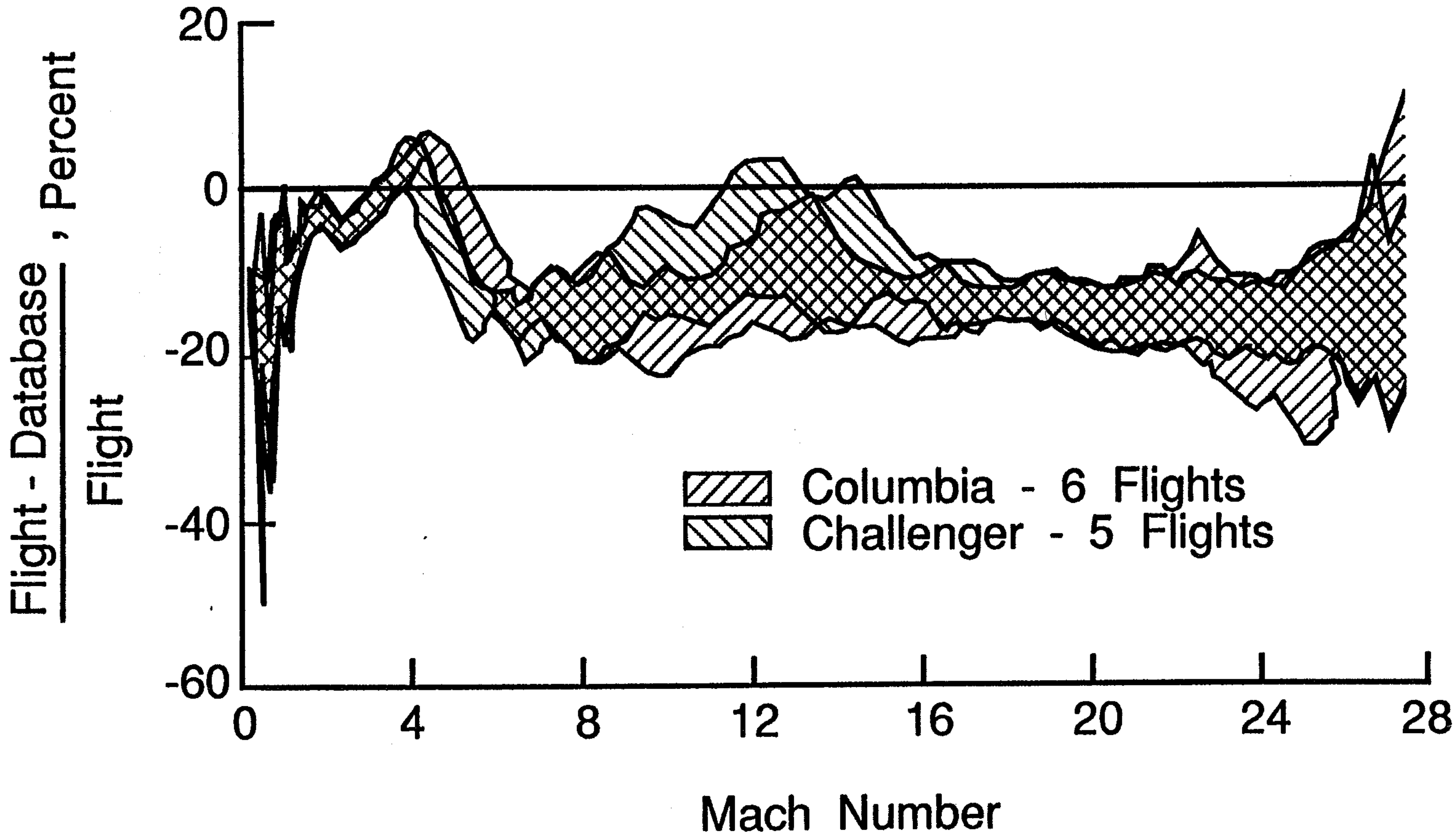
Projection to Planview

NON-AEROTHERMODYNAMIC EXPERIMENTS

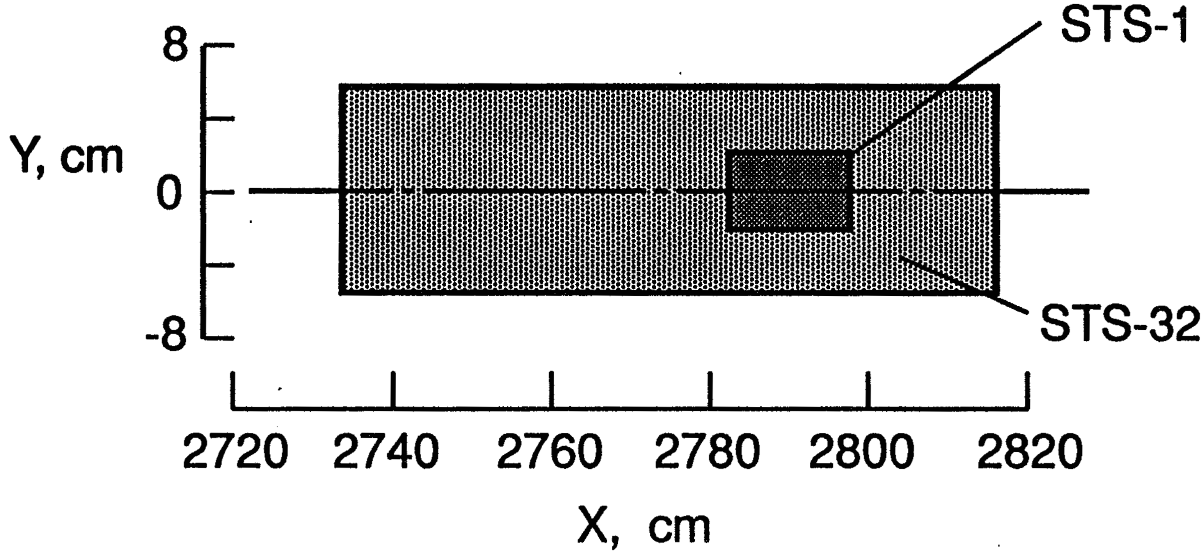
- o Dynamic, Acoustic, and Thermal Environments (DATE)
- o OEX Autonomous Supporting Instrumentation System (OASIS)
- o Advanced Autopilot Experiment (AAPE)
- o Advanced Flexible Reusable Surface Insulation (AFRSI)

OEX FLIGHT DATA UTILIZATION EXAMPLES

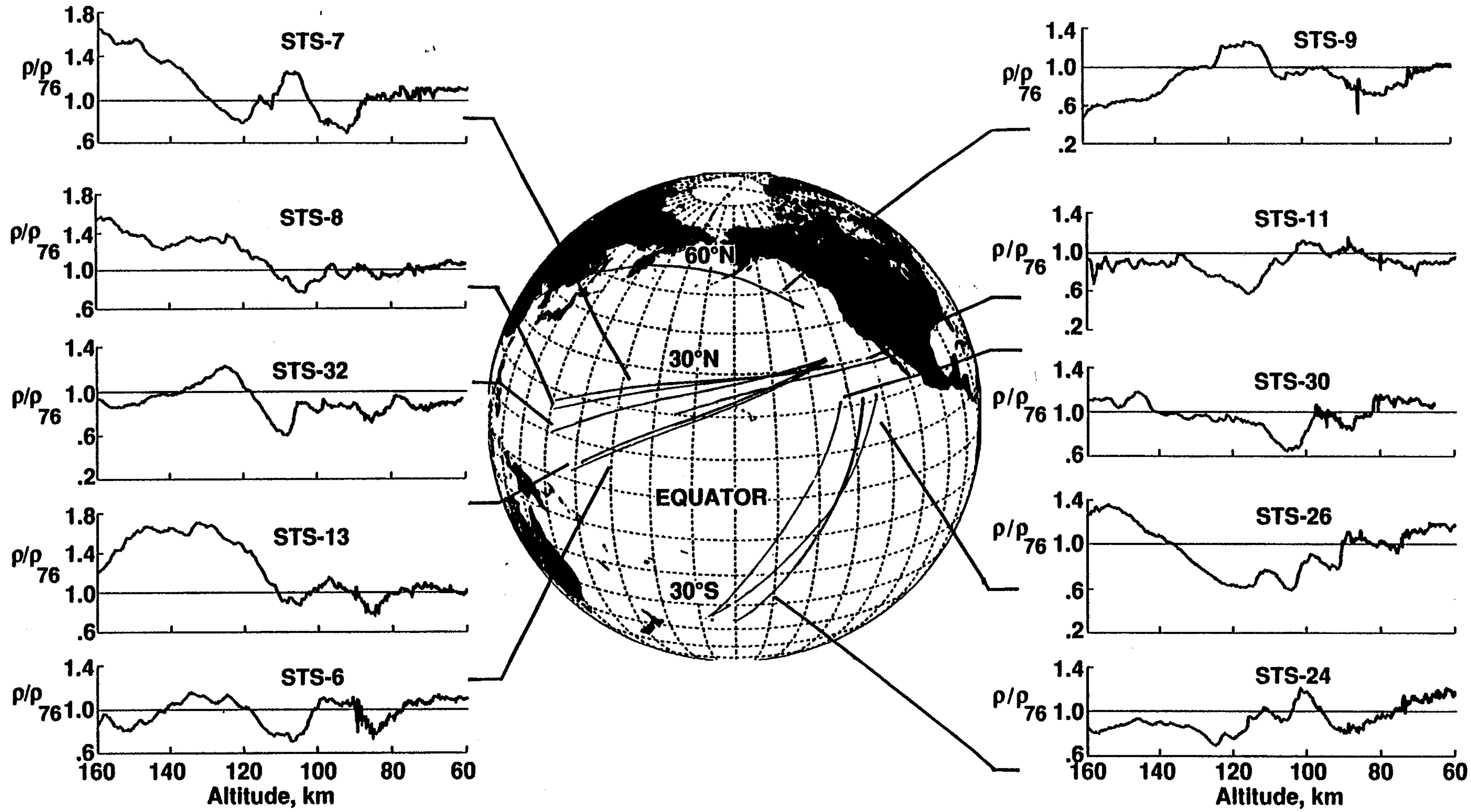
TYPICAL FLIGHT / WIND TUNNEL DATABASE COMPARISON
Axial Force Coefficient



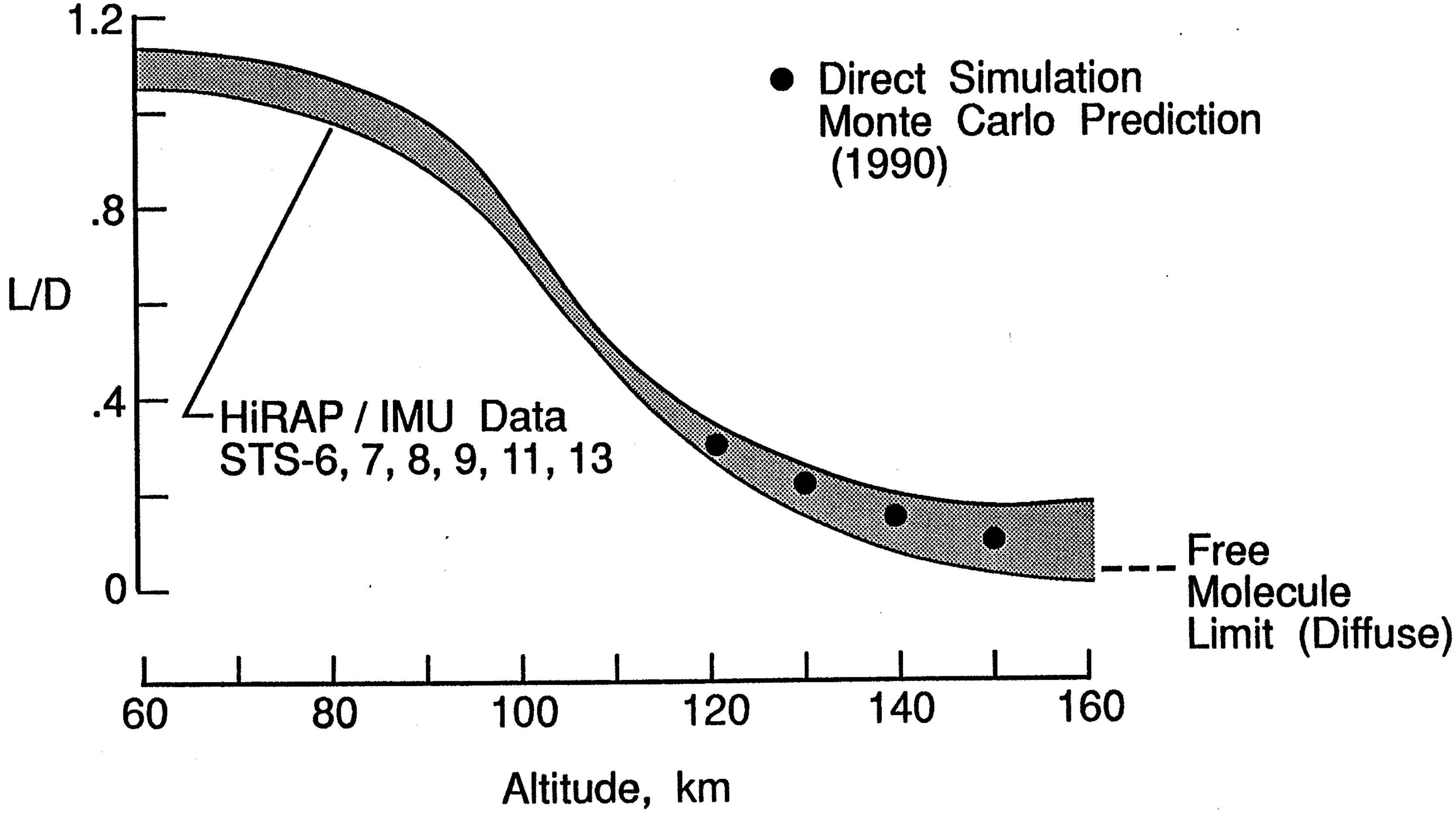
ACIP DATA ENABLED EXPANSION OF ALLOWABLE ORBITER C. G. ENVELOPE



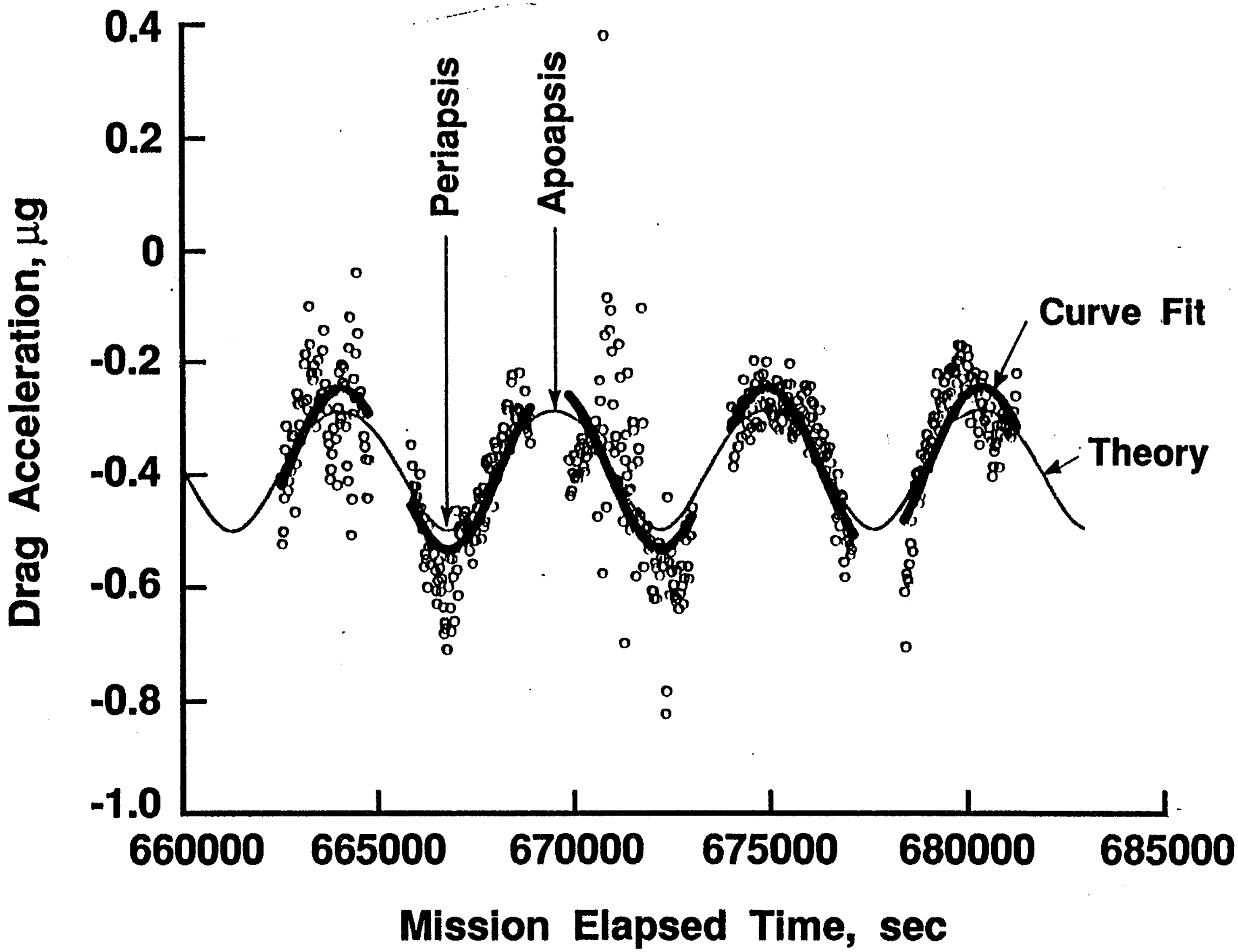
HiRAP/ IMU DENSITY DATA LED TO MODIFICATION OF GLOBAL REFERENCE ATMOSPHERE MODEL (GRAM)



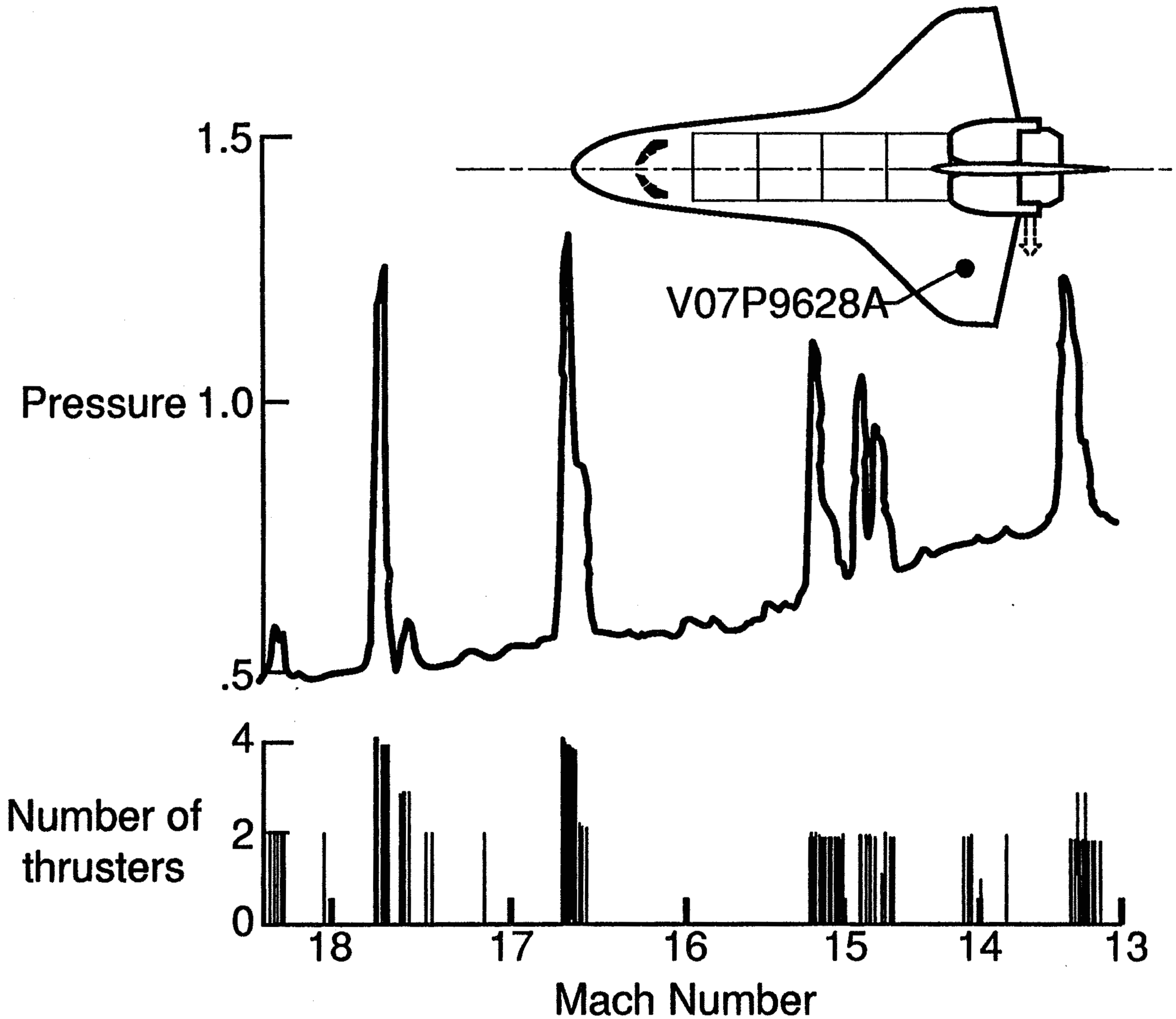
HiRAP PROVIDES VALIDATION DATA FOR RAREFIED FLOW COMPUTATIONAL TOOLS



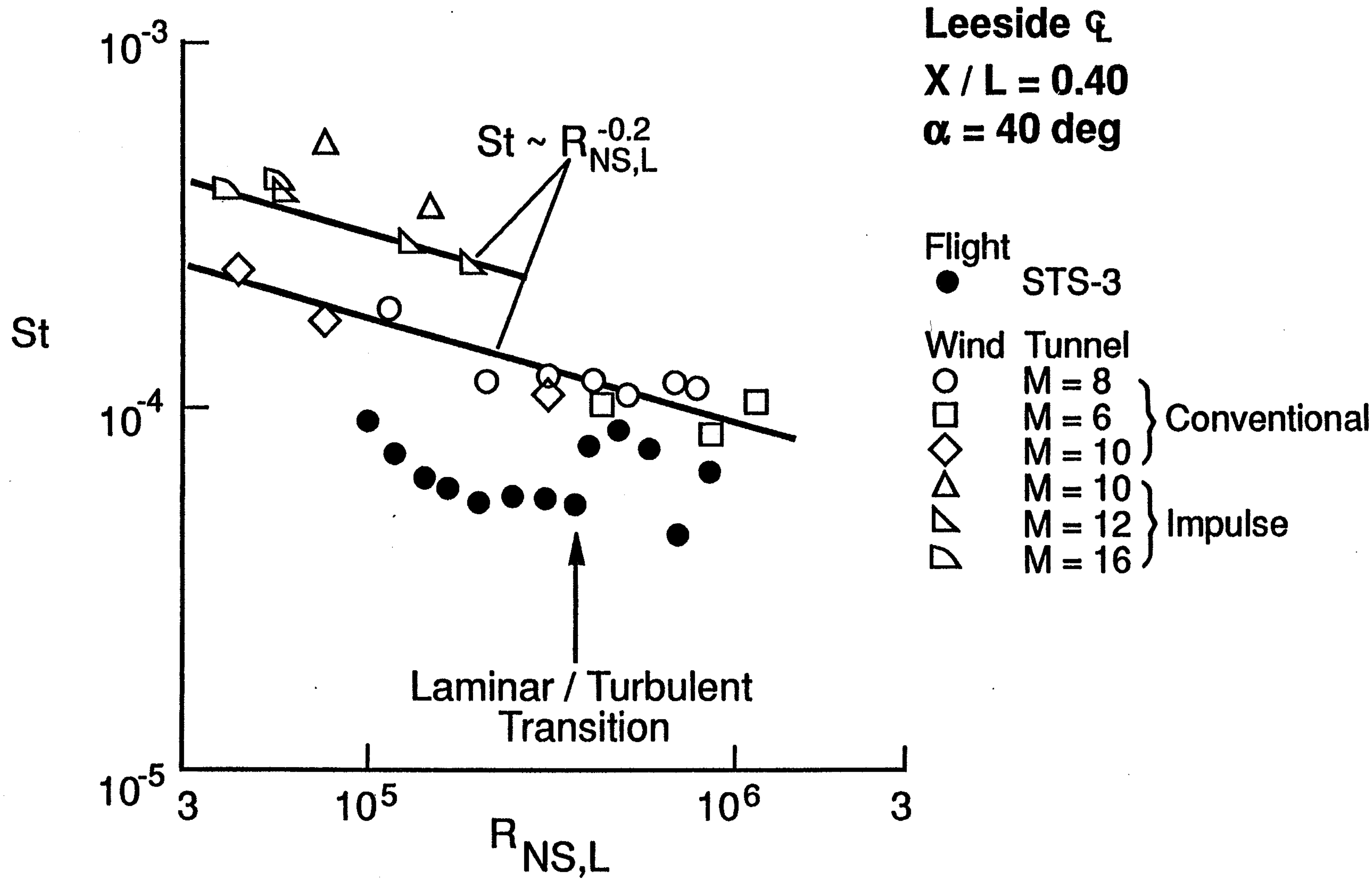
OARE SENSES PERIODIC ORBITAL DRAG VARIATION



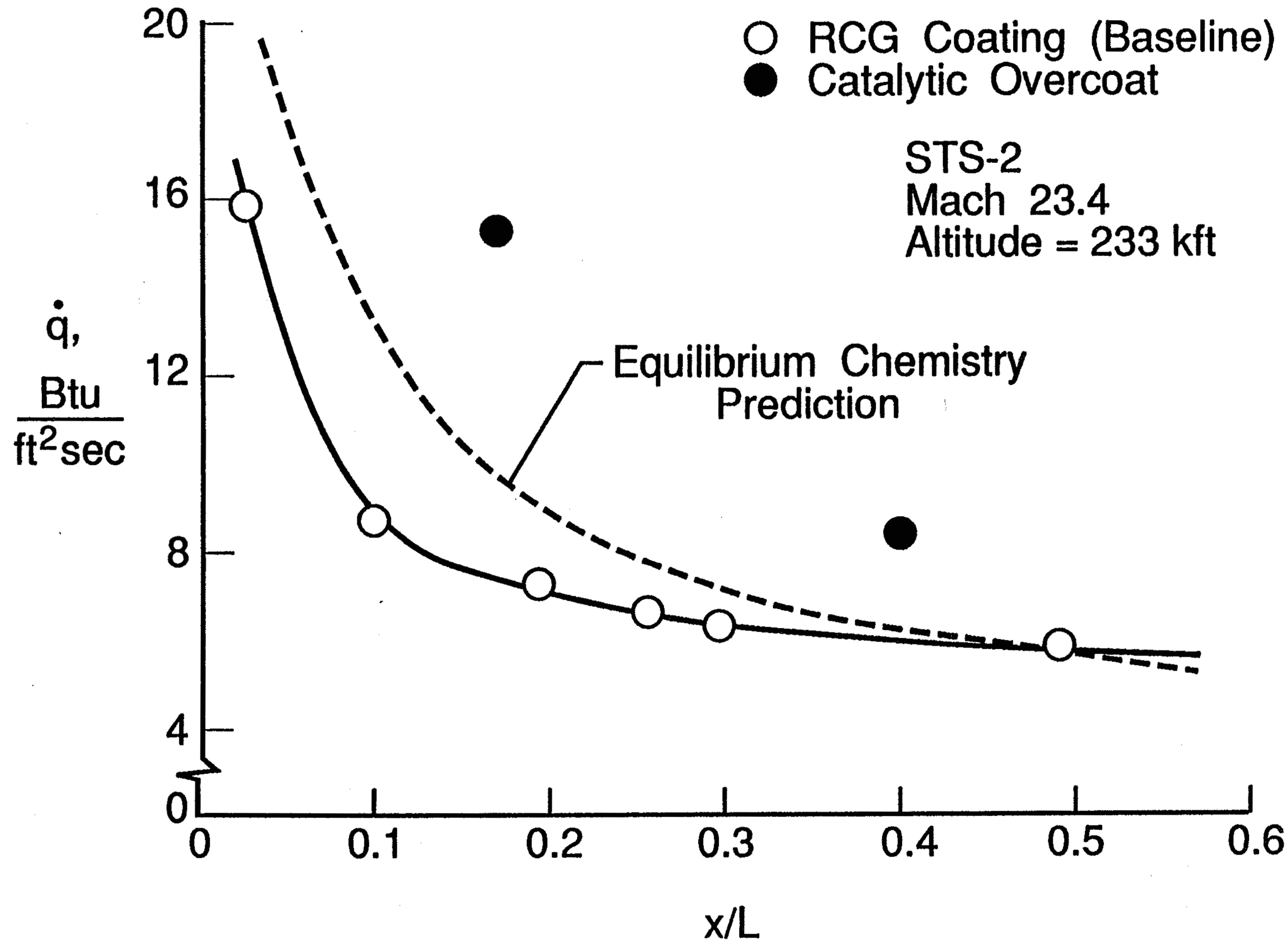
STS-3 CORRELATION OF PRESSURE CHANGES ON LEEWARD SURFACE OF WING WITH RCS FIRINGS



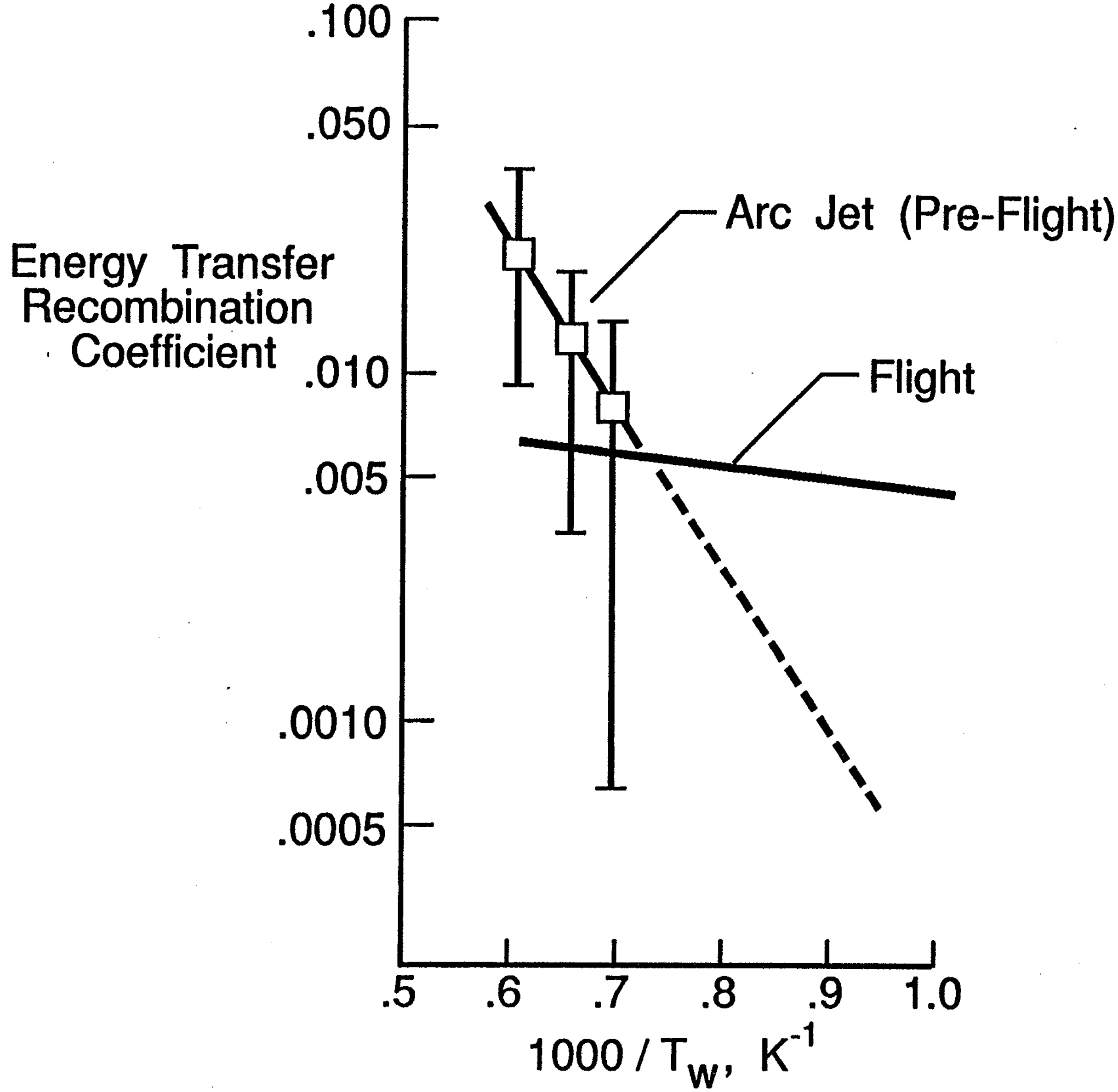
WIND TUNNEL DATA OVERPREDICT LEESIDE HEAT TRANSFER
(LAMINAR VS TURBULENT LEESIDE FLOWFIELD)



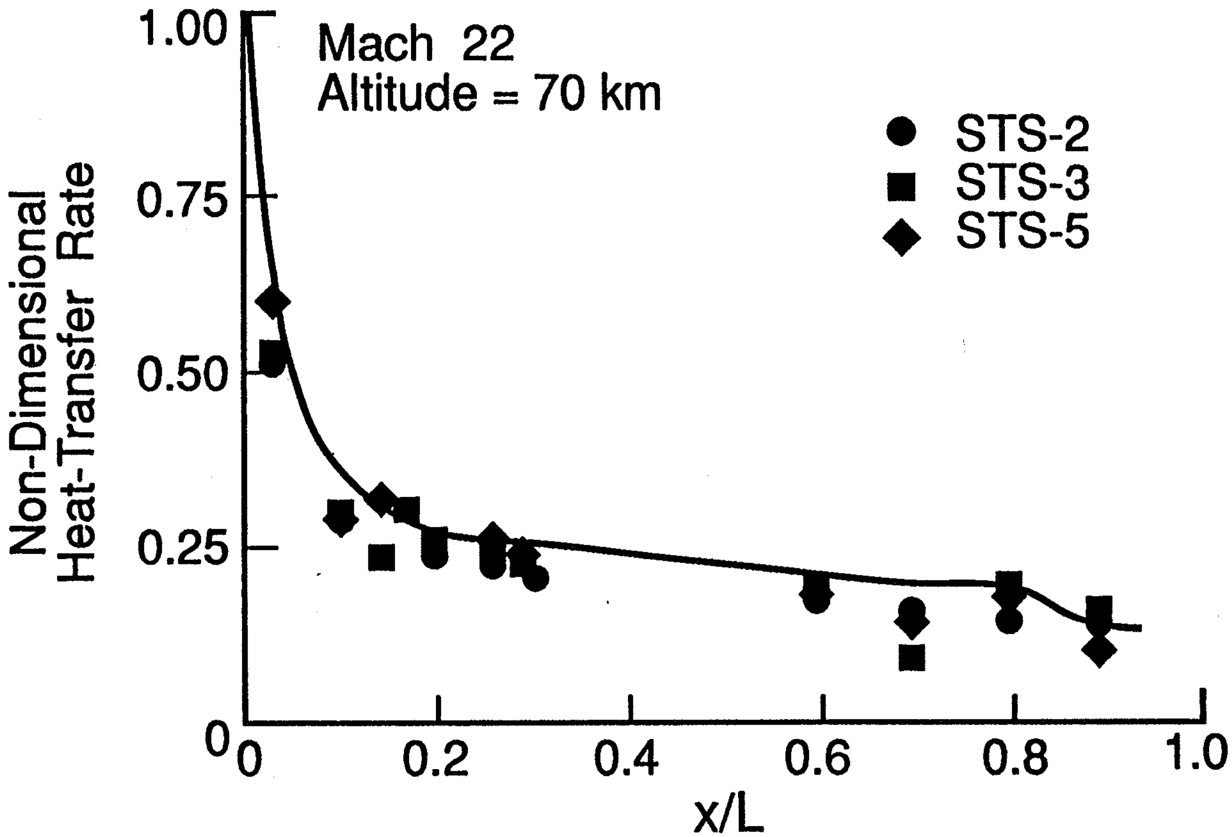
CSE EXPERIMENT CONFIRMS NON-CATALYTIC BENEFIT OF GLASS TILE COATING IN FLIGHT ENVIRONMENT



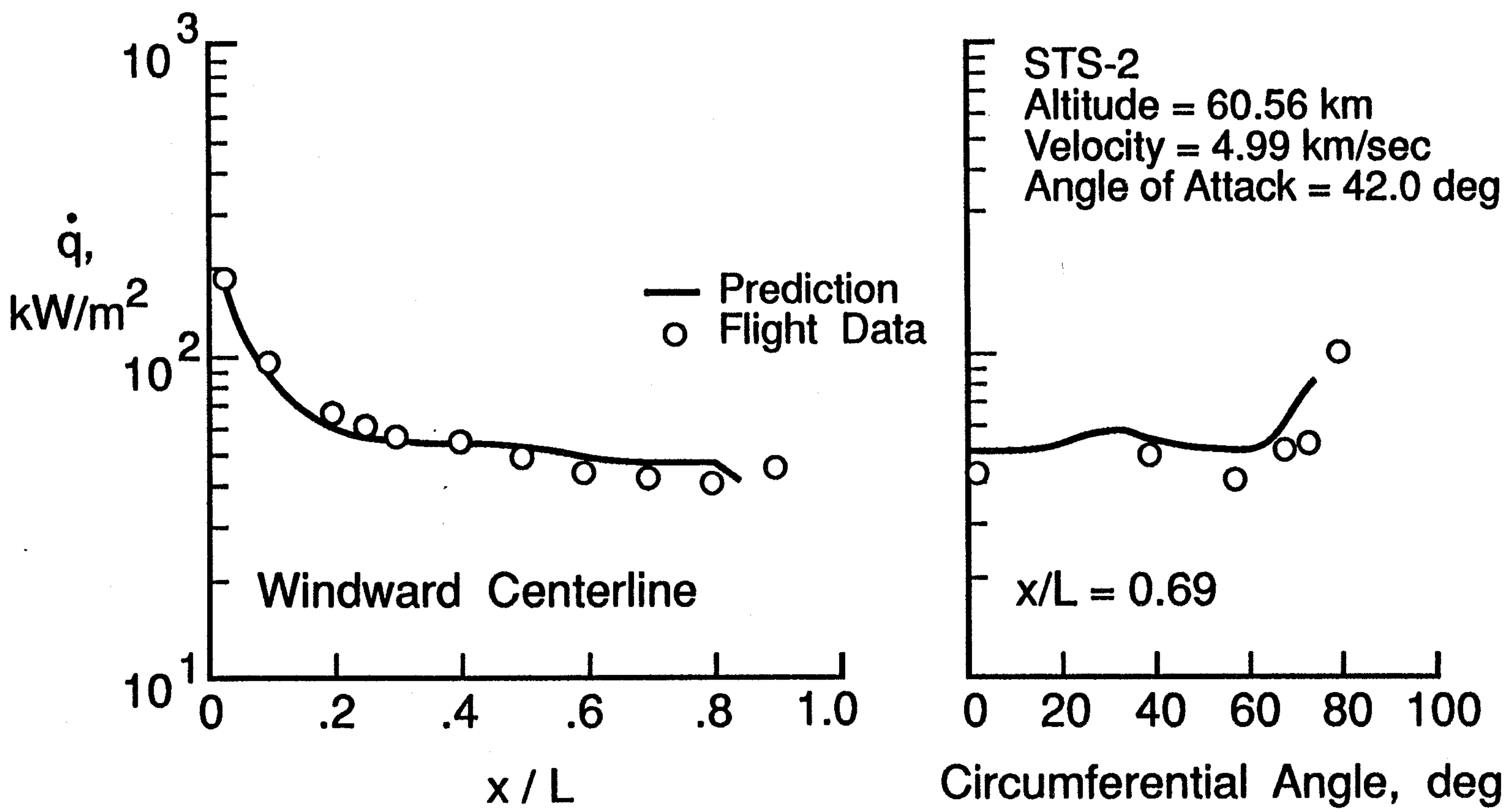
FLIGHT DATA ENABLE DETERMINATION OF TPS SURFACE CATALYTIC EFFICIENCY



COMPARISON OF FLIGHT DATA WITH 3-DIMENSIONAL NAVIER STOKES SOLUTION

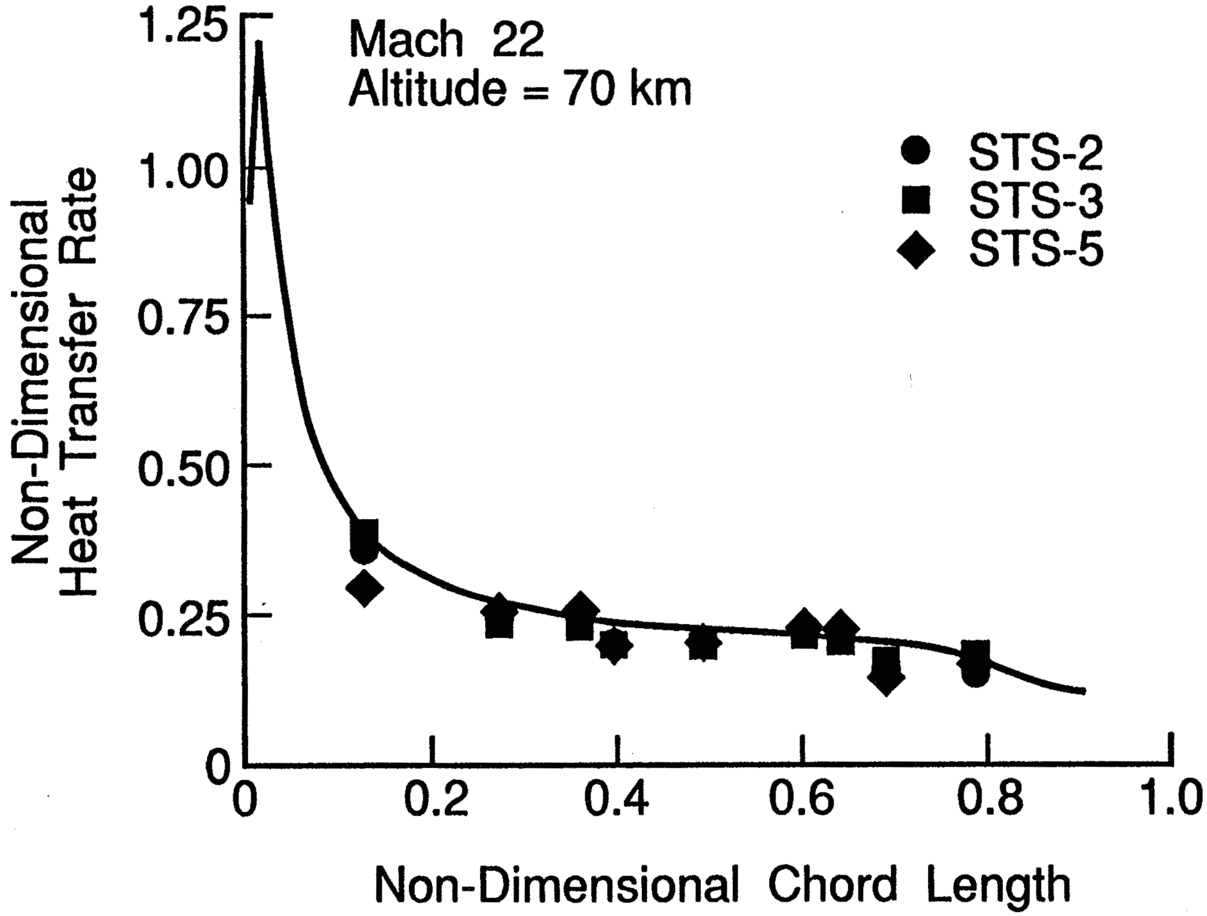


COMPARISON OF FLIGHT DATA WITH 3-DIMENSIONAL VISCOUS SHOCK LAYER SOLUTION

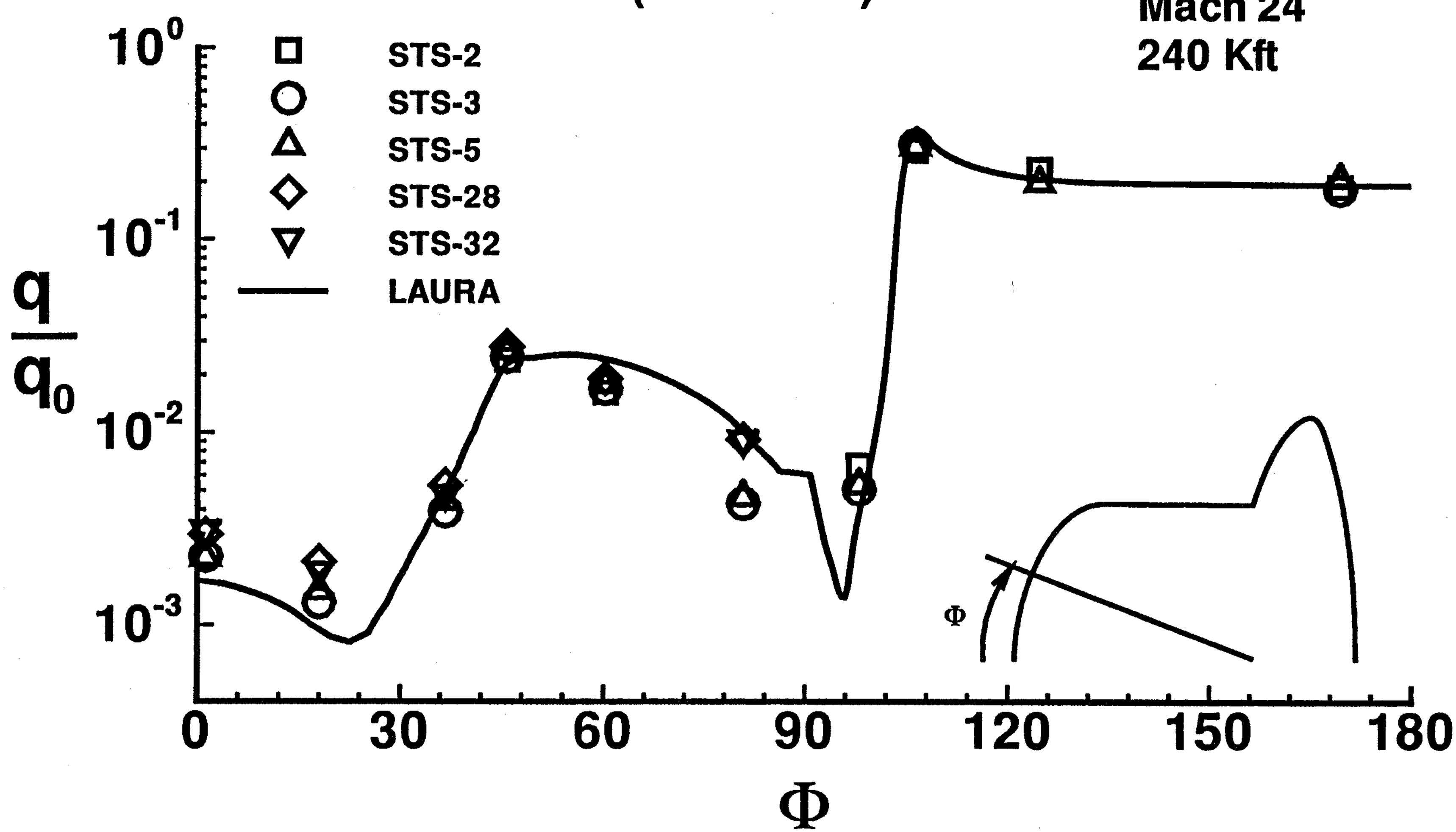


COMPARISON OF FLIGHT DATA WITH 3-DIMENSIONAL NAVIER STOKES SOLUTION

(50 PERCENT SEMISPAN)



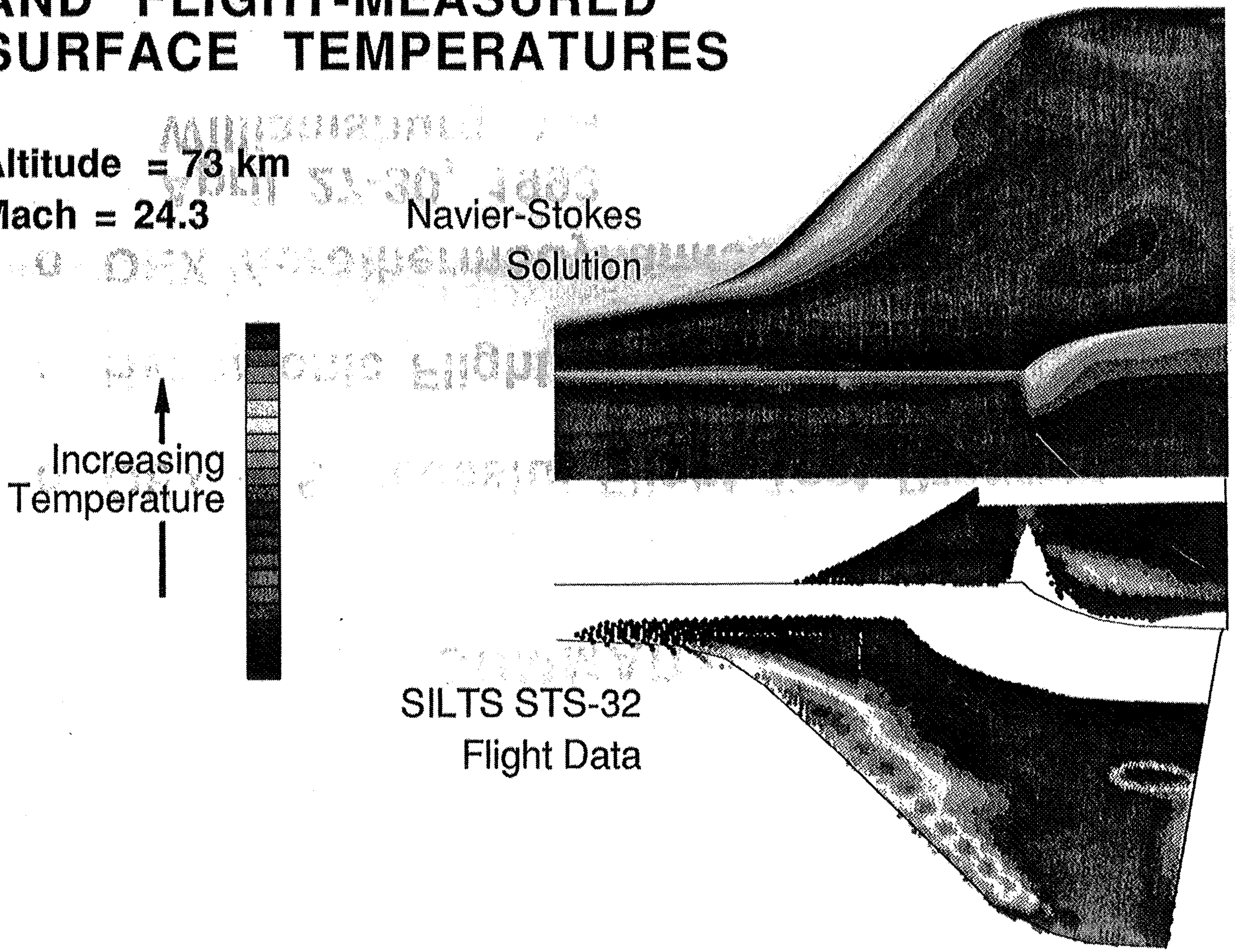
Comparison of Predicted and Flight-Measured Heat-Transfer Rates
(X/L=0.50)



COMPARISON OF PREDICTED AND FLIGHT-MEASURED SURFACE TEMPERATURES

Altitude = 73 km

Mach = 24.3



SUMMARY

- OEX -- Successful Flight Test Program
- Hypersonic Flight Test Lessons Learned
- OEX Aerothermodynamics Symposium
  April 27-30, 1993
  Williamsburg, VA

Space Active Modular Materials ExperimentS (SAMMES)

Low Earth Orbital Mission aboard the Space Test Experiments Platform (STEP-3)

David E. Brinza, Ph.D.
SAMMES/STEP-3 Principal Investigator
Jet Propulsion Laboratory
Pasadena, California

Prakash Joshi and Vic DiCristina
SAMMES Prime Contractor
Physical Sciences, Inc.
Andover, Massachusetts

NASA/DoD Flight Experiments
Technical Interchange Meeting
Monterey, California
October 7, 1992

SAMMES/STEP-3 Overview

- SAMMES Description
  - SAMMES/STEP-3 Team Members
  - System Architecture
  - System Control Module
  - Test Modules and Sensors

- SAMMES/STEP-3 Mission Overview
  - Mission Objectives
  - Mission Requirements
  - Orbital Operations
  - Data Analysis, Dissemination

- SAMMES Follow-on Efforts
  - SAMMES Enhancements
  - Health Monitor Applications
  - Potential Flight Opportunities

SAMMES/STEP-3 Team

Program Manager:	Lt. Col. Michael Obal, USAF (SDIO/TNI)
Principal Investigator:	David Brinza (Jet Propulsion Laboratory)
Experiment Support Group:	John Durrett, Leader (W.J. Schafer Associates) Graham Arnold (Aerospace Corp.) Michael Robyn (Aerospace Corp.) Robert Kraus (W.J. Schafer Associates)
Prime Contractor:	Physical Sciences, Inc.
Program Manager:	Vic DiCristina
Project Engineer:	Prakash Joshi
Major Sub-Contractors:	
Test Modules:	Research Support Instruments, Inc.
System Control Module:	Northeastern University
Environmental Test:	Fairchild Space Co.
STEP-3 Mission Manager:	Lt. Janet Mayer, USAF (SMC/CUL)
STEP-3 Experiment Integrator:	Douglas Wille (TRW)

SAMMES System Architecture

- Autonomous Modular System
  - System Control Module
  - Distributed Test Modules
  - Internal MIL-STD-1553 Communications Bus

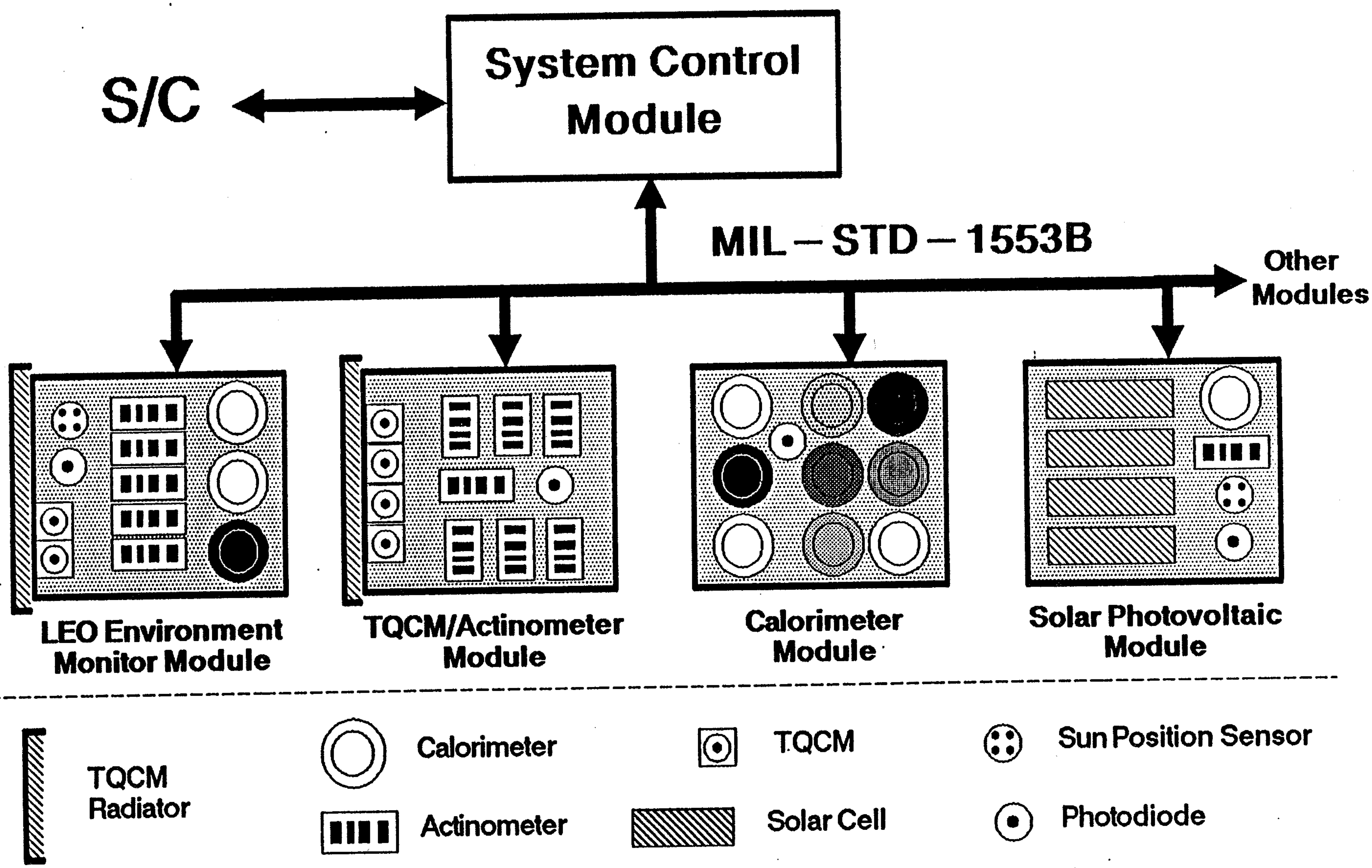
- Spacecraft Interface Adaptability
  - Host 1553, RS-232, RS-422 Standard Interfaces
  - TM Operations Controlled by SCM
  - Data Storage (8 Mbyte) Within SCM

- Flight Experiment Flexibility
  - Up to 8 Test Modules Controlled by SCM
  - Data Acquisition Asynchronous to Spacecraft Operations
  - On-board Data Processing Capability
  - Uplinkable Code for Operations and Data Processing

- STEP-3 Configuration
  - One System Control Module and Five Test Modules

    LEO Environment Monitor Module, Ram/Wake Calorimeter Modules, TQCM/Actinometer Module and Solar Photovoltaic Module

SAMMES System Architecture



SAMMES System Control Module

- Electronic Design
  - Host Microcontroller
    S/C Commands, Data Transfer
  - TM Microcontroller
    TM Operations, Data Acquisition
  - Program Memory
    128 kByte + 16 kByte Dual Port
  - Data Memory
    1 MByte EEPROM, 7 Mbyte DRAM (battery back-up)
  - Communications
    SCM/TM: MIL-STD-1553B
    Host/SCM: MIL-STD-1553B, RS-232, RS-422
  - Power Management
    Auto-quiescent Mode, Conditioning, Heaters
  - Health and Status
    Temperatures, Microcontroller Status

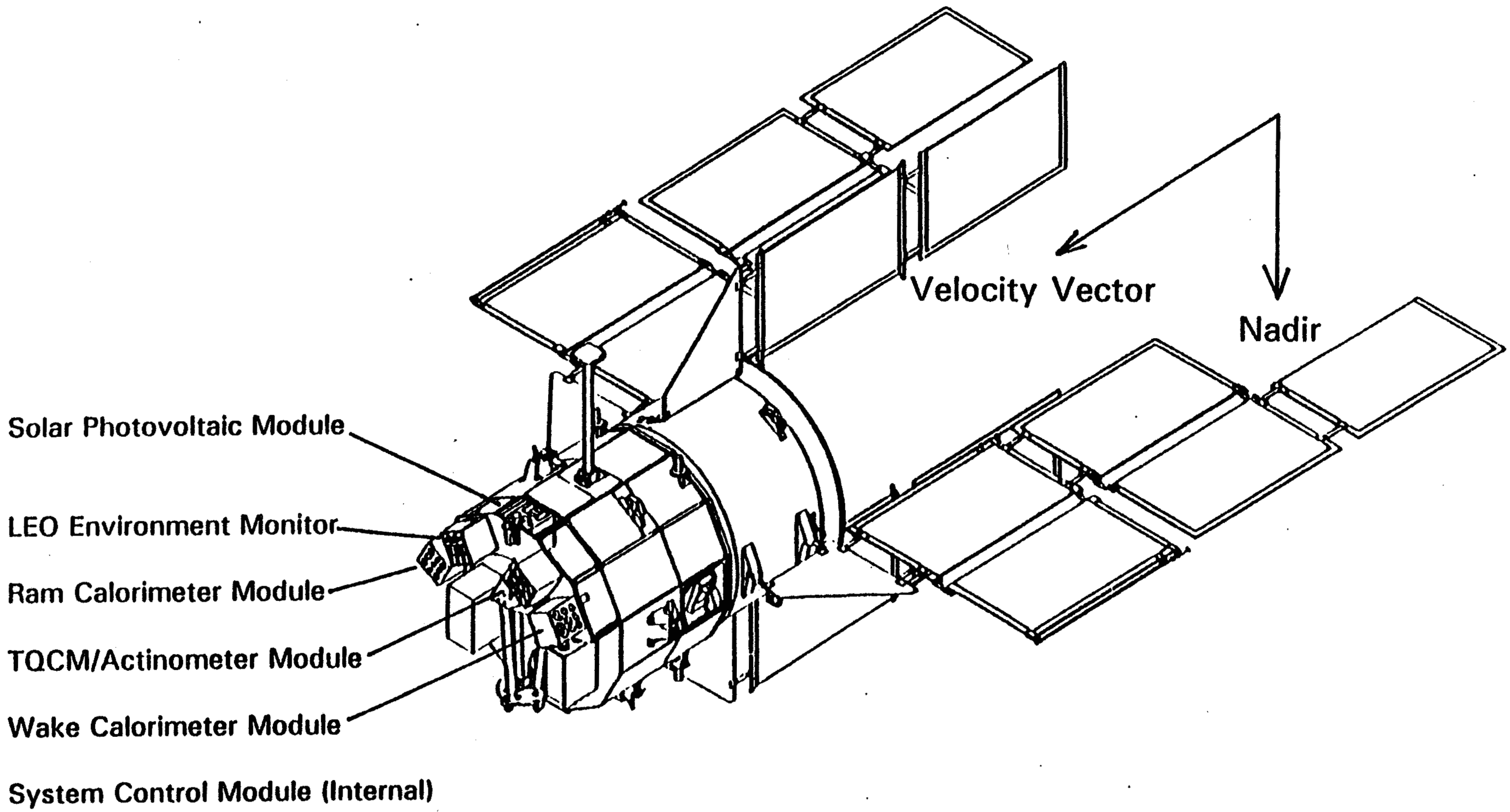
- Mechanical
  - Dimensions: 7.875" x 7.500" x 6.063"
  - Weight: 4.71 kg (Mg), 6.08 kg (Al)

SAMMES Test Module (Typical)

- Architecture
  - Microcontroller
    - SCM Commands, Experiment Control, Data Transfer
  - Analog Signal Conditioning & ADC
  - Sensor Temperature Measurement and Control
  - Sensors
    - Temperature-Controlled Quartz Crystal Microbalances
    - Temperature-Controllable Reichard-Triolo Calorimeters
    - Temperature-Controlled Atomic Oxygen Actinometers
    - RADFET Total Radiation Dose Monitors
    - Sun Position Sensors, Photodiodes
    - Solar Photovoltaic I-V Diagnostics
    - Temperature Sensors (PRT & AD590)
  - Operational Modes
    - Quiescent Mode: Maintain Specimen Temperatures
    - Acquisition Mode: Sensor Sampling, Temperature Control
- Mechanical
  - Dimensions: 6.500" x 6.000" x 5.500" (excluding radiators)
  - Weight: 2.5 - 3.2 kg

SAMMES on STEP-3

- **Test Module Configuration on STEP-3 Vehicle**



SAMMES/STEP-3 Mission Objectives

- Assess LEO Space Environmental Effects on SDIO Materials
  - Performance (a/e) of Thermal Control Materials (Ram/Wake)
  - Durability of Optical, Thermal Control, Protective Coatings
  - Performance of Advanced Solar Photovoltaics

- Quantify Orbital and Local Environments
  - Measure Atomic Oxygen Flux and Fluence
  - Assess Contaminant Accretion, Species ID, and Effects
  - Determine Sun Angle, Earth Albedo and Irradiance
  - Measure Total Radiation Dose

- Demonstrate Modular Experiment Concept
  - Autonomous Operations
  - Internal Power Management
  - Uplink Operational and Data Processing Code

SAMMES/STEP-3 Mission Requirements

- Orbit Parameters, Mission Duration
  - LEO Circular Orbit (~500km)
  - Ram and Wake Exposure Environments
  - 1-Year Minimum, 3-Year Goal

- Data Integrity and Validation
  - Material Pedigree
  - Contamination Control
  - Complete Environmental History (Early Mission Phase)
  - Test Material Temperature Control/Knowledge
  - Benchmark Material Performance

- SAMMES/STEP-3 System Requirements
  - System Mass: <25 kg
  - System Power:
    - Peak (Operating): <30 W
    - Average (Quiescent): <20 W
  - Data (average) <1 Mbyte/day

SAMMES/STEP-3 Orbital Operations

- Early Operations (Insertion --> Post-Checkout)
  - Power-up SAMMES, Early Operations Initiate Command
  - Verify SCM Status (if not operating, recycle power & initiate)
  - Activate Specimen Heaters
  - Autonomous SAMMES Operation:
    - Sample and Store Data from Selected Sensors (up to 8 Mbyte)
    - Power: ~28 W (Power-fault tolerant)
  - Downlink up to 8 Mbyte Data at end of Check-out Phase

- Nominal Operations
  - Initiate Normal Operations Command (once per day)
  - Autonomous Operation:
    - Deactivate Calorimeter Heaters, Stabilize (2 orbits @ ~15W)
    - Acquire Sensor Data (1.1 orbits @ ~28 W)
    - Transfer Data to S/C Onboard Storage
    - Re-activate Calorimeter Heaters
    - Return to Quiescent Mode (12 orbits @ ~18W)
    - Downlink ~1Mbyte Data

- Special Operations
  - Thermo-Gravometric Analysis (QCM's), Etc. (TBD)

SAMMES Data Analysis & Dissemination

- Time-Variant Sensor Data
  - Full Orbital Temperature Profiles for Calorimeters
  - Frequency/Temperature Data for TQCMs
  - Resistance Measurements for Actinometers and Radiation Monitor
  - I-V and Temperature Data for Solar Photovoltaics
  - Current Measurements for Sun Sensors, Photodiodes

- Data Conversion and Analyses
  - Conversion to Engineering Units
  - Calibration Factors
  - Analysis Algorithms
  - Contamination Effects Assessment

- Data Dissemination
  - SDIO SEE Database
  - Interim and Final Reports
  - Workshops, Conferences and Publications

SAMMES Enhancements

- Test Module Autonomy
  - Eliminate Need for System Control Module
  - Expanded TM Data and Program Memory
  - MIL-STD-1553 (Option for: RS-422, RS-232)

- Test Module Miniaturization and Hardening
  - ASIC, Hybrid Circuitry
  - Extensively Remoted Sensors
  - Radiation Hardening via Spot Shielding, Parts Selection

- Expanded Sensor Suite
  - Optical Properties Monitoring
  - Micrometeoroid and Debris Impact Sensing
  - Proton Spectrometer

SAMMES Health Monitor Applications

- General Spacecraft Engineering Data
  - Temperature Monitoring
  - Accelerations, Structural Deformations
  - Power System Monitoring
    - Solar Array Diagnosis
    - Battery Charge Rates

- Orbital Environment Monitoring
  - Atomic Oxygen Flux
  - Internal Radiation Dosage
  - Debris Cloud Detection

- Payload and Mission Specific Monitoring
  - Contamination Events and Effects
  - Optical System Diagnosis
  - Solar Exclusion Monitor (Safing)

SAMMES Potential Flights

- SDIO TECHSAT
  - Low Earth Orbital Mission
  - Mid-altitude Earth Orbital Mission

- SDIO Testbed and Demonstration Vehicles
  - Brilliant Pebbles Orbital Flight Test Vehicles
  - Brilliant Eyes Dem/Val Spacecraft

- SDIO Operational Spacecraft
  - Brilliant Eyes
  - Brilliant Pebbles

- Other Satellites and Platforms
  - Space Station Freedom and Free-Flyers
  - DoD Spacecraft
  - Civil Spacecraft (NASA, NOAA, Commercial)

NPB CESIUM SPACE EXPERIMENT (U)

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ABSTRACT (U)

(U) Neutral Particle Beam (NPB) Weapons systems are planned to perform the ballistic missile defense functions of nuclear weapon/decoy discrimination and warhead kill at appropriate energy levels and ion currents. Negatively charged ions are produced in a specialized ion source and focused into a high quality particle beam. NPB linear accelerators accelerate and steer the negatively charged ions using electric and magnetic fields. After acceleration and steering the neutralizer system strips away extra electrons from ions to form the electrically neutral particle beam. The neutral beam then travel through space to the target unaffected by the Earth's magnetic fields. Continuing technological advances have greatly reduced the size and weight of NPB accelerator systems. Ion current production has been enhanced by over 100 per cent with the intermittent addition of cesium at the NPB ion source device. This increase in current is essential to attain the most light weight, compact NPB platforms and minimize expensive launch costs. Addition of cesium into the ion source has been identified by the NPB community as the highest priority risk reduction space experiment necessary prior to planned NPB accelerator experiments and later weapons systems.

(U) The NPB Cesium Space Experiment is planned to successfully demonstrate controlled cesium introduction and vaporization into a simulated ion source chamber. Microgravity effects on the cesium deposition will be studied as will the effects of small amounts of cesium on high voltage accelerator components that might be susceptible to electrical insulator break downs. The experiment design will simulate as closely as possible the environmental, physical and operational characteristics of the actual NPB ion source.

Introduction (U)

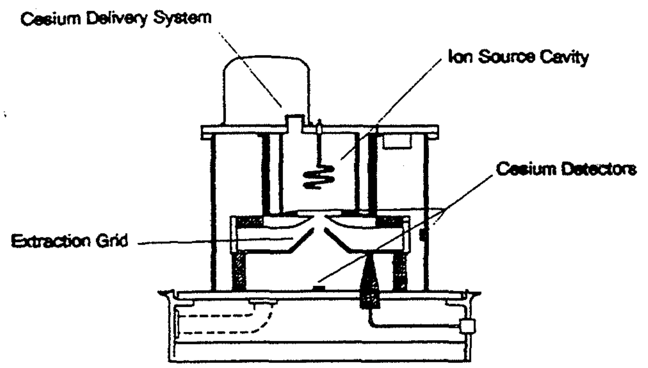
(U) The NPB systems planned for experiments and as future weapons systems all depend upon a reliable high current ion source to produce negative ions which can be accelerated to energies appropriate to the required mission. Any application of a particle beam system in a spacecraft requires that the greatest efficiency of output be attained at the lightest weight possible to minimize associated launch vehicle costs. Design developments which increase beam current output with little additional weight are sought after for system enhancement. The addition of cesium to the ion source as a catalyst for ion production has been shown to increase output by over 100% depending upon the ion source design. Cesium is added to ground based ion sources by heating a supply of cesium in an oven type device to vaporize minute amounts of the metal and supply it into the ion source cavity. This method cannot be utilized in an orbital system due to the open nature of the system. Only small amounts of cesium are required to provide the current enhancement and its intermittent introduction occurs only when current output falls after long periods of operation. Introduction of cesium must be limited to prevent the arcing of high voltage accelerator components installed downstream of the ion source.

(U) Previous experience with cesiated ion engines on spacecraft have shown that careful design of the delivery system is essential. In 1974 the Advanced Technology Satellite, ATS-6, tested station keeping ion thrusters utilizing a cesium delivery system to enhance current output. The test engines were carefully designed, built, and successfully tested on the ground and in KC-135 aircraft. On orbit the two ion engines each operated one time and then failed in a mode that suggested the cesium flow had not been properly controlled and short circuited the engines. The use of cesium for ion thruster enhancement was apparently discontinued after this experience.

(U) NPB ion source developers at Culham Laboratory, (Abingdon, UK) and Grumman Aerospace Corporation (Bethpage, NY) have baselined a cesium delivery system design for the NPBSE that overcomes the postulated failure modes present in the ATS-6 satellite. A special study by Grumman Aerospace Corporation was authorized by the US Army Space and Strategic Defense Command to identify risk reduction experiments that would increase the confidence fo the successful operation of the NPB Space Experiment (NPBSE). The contractor team identified several tests that should be performed tc reduce risk for the NPBSE. The cesium delivery system experiment was identified as the most valuable space experiment. (See Grumman Repor " Accelerator Component Experiment (U), Special Study NPBSE-01", prepared under US Government contract DASG60-90-C-0103)

(U) EXPERIMENT DESCRIPTION

(U) The experiment will include a simulated ion source identical to that baselined for the NPBSE (see diagram 1). The cesium delivery system heats and supplies cesium to the ion source cavity from a metal bellows assembly under a gas pressure actuation, through a small bore (0.005 inch) capillary tube, and a heater/vaporizer assembly. Grid systems which simulate the negative ion extraction system and their associated insulators will be raised to 35 Kv to investigate their behavior with the minute cesium that will migrate to the insulators' location. Instrumentation will measure cesium delivery, chamber pressures, delivery component temperatures and extraction voltage characteristics. Data collection and experiment control will be performed by the STS GRID computer system. A mission specialist will initiate the experiment by connecting a power cord to the orbiter power system and starting the



Cesium Space Experiment Simulated Ion Source
Figure 1.

controlling computer program to begin the experiment. Periodic checks will be made to assure that the experiment parameters are normal. If experiment setpoints are exceeded, alarm signals will alert the mission specialist to check the conditions and alter or terminate the experiment as is appropriate.

STATUS (U)

(U) The NPBCSE hardware experiment container will utilize a standard previously flown NASA approved design. Some additional requalification will be done to meet current safety specifications. The container is presently being fabricated and is expected to be complete in October 1992. Early completion of the enclosure will allow the experimental apparatus developer to fit check and test the equipment within the actual size container.

(U) The experimental cesium delivery device preliminary design is completed and a functionally equivalent test device has been prepared to verify proper operation on an operational accelerator system. Documentation activities have begun that fit a typical timeline for a launch ready experiment in mid year 1994.

SUMMARY (U)

(U) The Cesium Space Experiment will perform a necessary risk reduction investigation of cesium delivery in orbit. This experiment will demonstrate that a device can deliver intermittent supplies of cesium into an ion source in the microgravity environment. Simulated accelerator system voltage will characterize the migration of the cesium within the simulated extraction grids and demonstrate that cesium breakdowns can be either avoided or controlled in a linear accelerator. The cesium delivery system will be the first flight hardware supporting the NPBSE and other NPB accelerator systems. This equipment could be useful for other similar space flight applications.

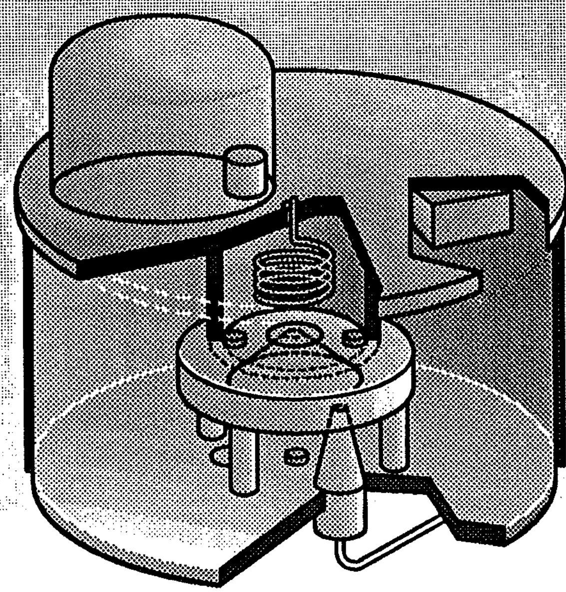
REFERENCES (U)

[1] R. Worlock, *et al.*, "ATS-6 Cesium Bombardment Engine North-South Stationkeeping Experiment," IEEE Transactions on Aerospace and Electronic Systems, Vol. AES-11 no. 6, November 1975

[2] Grumman Aerospace Corporation, "Accelerator Component Experiment (U) - Final Report prepared under USASDC Contract DASG60-90-C-0103, Technical Directive Number -001

M-920211-67U-A (C) (2273)

NEUTRAL PARTICLE BEAM ION SOURCE CESIUM SPACE EXPERIMENT (NPBCSE)



PRESENTED BY

G. M. PARSONS

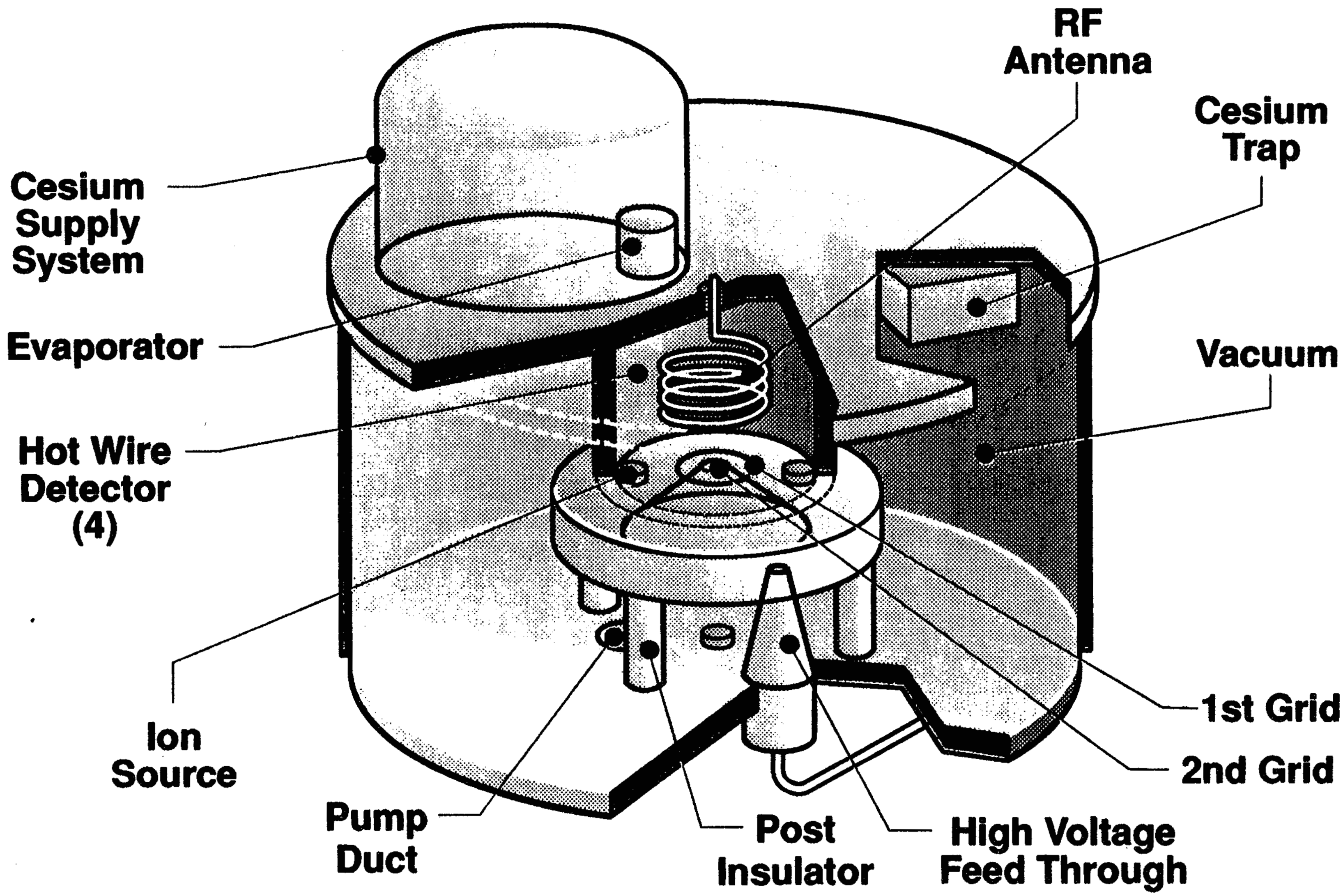
U.S. ARMY STRATEGIC DEFENSE COMMAND

DEW DIRECTORATE

UNCLASSIFIED

NEUTRAL PARTICLE BEAM ION SOURCE CESIUM SPACE EXPERRIMENT (NPBCSE) (U)

M-920419-08U-D (C) (2274)



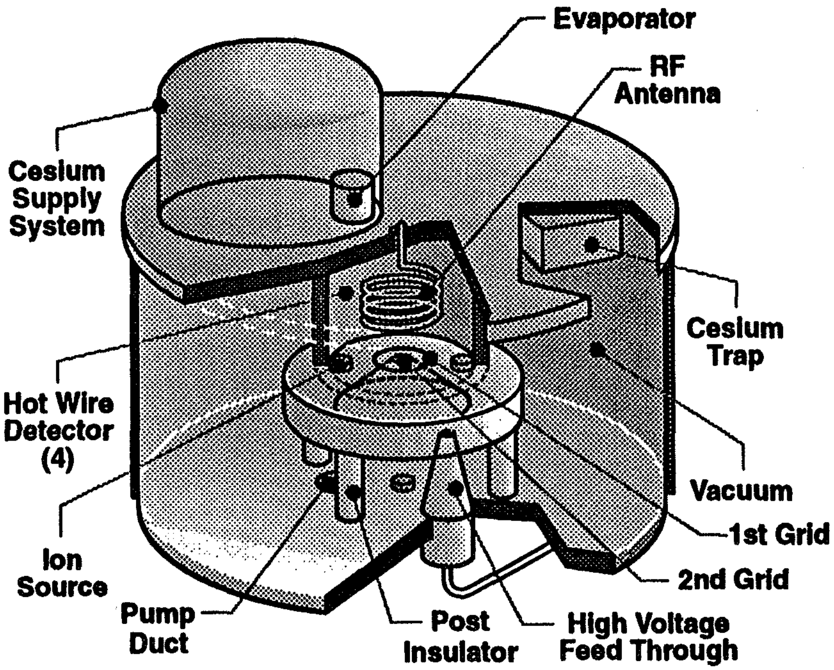
NEUTRAL PARTICLE BEAM ION SOURCE CESIUM SPACE EXPERIMENT (NPBCSE) (U)

M-920419-08U-B (C) (2274)

CONCEPTS AND OBJECTIVES

REALISTIC MICROGRAVITY SIMULATION OF NPB ION SOURCE, INJECTOR AND ACCELERATOR TO DEMONSTRATE RELIABLE CESIUM DELIVERY AND CONTROL FOR NPBSE RISK REDUCTION

- TWO CHAMBERS AND ELECTRONICS WILL DEMONSTRATE ESSENTIAL ION SOURCE FUNCTIONS
  - CESIUM SUPPLY PROOF-OF-CONCEPT
    - REGULATED VAPORIZATION INTO ION SOURCE
    - CONTROLLED MIGRATION AND DEPOSITION WITHIN ACCELERATOR
  - SIMULATE NPB ACCELERATOR HIGH VOLTAGE (HV) CHARACTERISTICS WITH CESIUM IN MICROGRAVITY ENVIRONMENT
- SEMI-AUTONOMOUS AFTER CREW POWER UP WITH PERIODIC MONITORING BY CREW MEMBER
  - INDEPENDENT OF GROUND CONTROL
  - REAL-TIME DIAGNOSTICS



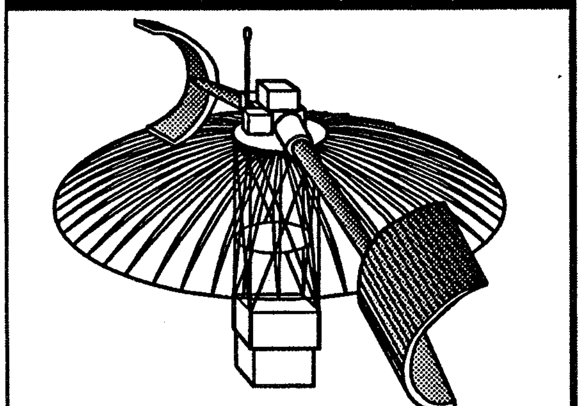
- MEASURES CESIUM MIGRATION CHARACTERISTICS
  - SURFACE DEPOSITION RATE
  - VAPOR PRESSURE
  - EVAPORATOR AND CAPILLARY TEMPERATURES
- INSTRUMENTATION RANGES:
  - HOTWIRE CESIUM VAPOR DETECTOR (10^{-3} TO 1 TORR)
  - TEMPERATURE SENSORS (0 TO 600° C AND 0 TO 100° C)
  - PIRANI GAUGE (10^{-6} TO 100 TORR)
- CHAMBERS AND INSTRUMENTATION HOUSED IN NASA-APPROVED CONTAINER REPLACING TWO MIDDECK LOCKER SPACES
  - COMPLIES WITH FLIGHT SAFETY REQUIREMENTS
- ADJOINING ELECTRONICS PACKAGE REPLACES ONE MIDDECK LOCKER SPACE

NEUTRAL PARTICLE BEAM ION SOURCE CESIUM SPACE EXPERIMENT (NPBCSE) JUSTIFICATION (U)

M-920419-07U-B (C) (2274)

- **MILITARY RELEVANCE**
  - HIGH CURRENT/ENERGY NPB WEAPONS WILL PROVIDE ESSENTIAL DISCRIMINATION AND RV KILL CAPABILITIES TO BALLISTIC MISSILE DEFENSE SYSTEMS
    - CESIUM IS ESSENTIAL TO ACHIEVE THE 100%-200% INCREASE IN ION CURRENT REQUIRED FOR OPTIMUM NPB WEAPONS SYSTEM PERFORMANCE
    - SUCCESS DEPENDENT ON RELIABLE PERFORMANCE OF CESIATED ION SOURCE
      - ACHIEVING CORRECT CONCENTRATION OF CESIUM IN THE ION SOURCE
      - KEEPING HV INSULATORS AND GRIDS CESIUM FREE
- **COMPARISON OF ALTERNATIVES**
  - GROUND SIMULATION OF ON-ORBIT CONDITIONS NOT POSSIBLE - LONG TERM MICROGRAVITY REQUIRED

APPLICATIONS TECHNOLOGY SATELLITE (ATS-6)



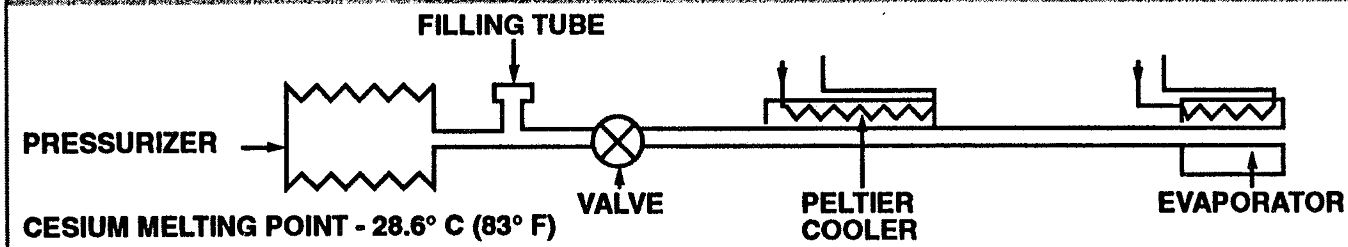
ATS-6 WAS EXTENSIVELY TESTED ON GROUND AND ON THE KC-135

- **NEED FOR SPACEFLIGHT**
  - EARLY MICROGRAVITY TESTING OF THE REVISED CESIUM SUPPLY SYSTEM REQUIRED TO SUPPORT NPB PROGRAM
  - NECESSARY FOLLOW-ON TO SUCCESSFUL NPBCSE BREADBOARD TESTING
  - NPBCSE DESIGN ADDRESSES ATS-6 ION THRUSTER FAILURE MODES
    - INSUFFICIENT ATS-6 SUPPLY CHAMBER WETTING
    - OUTGASSING DURING ATS-6 OPERATION

NEUTRAL PARTICAL BEAM CESIUM SPACE EXPERIMENT (NPBCSE) DETAILED OVERVIEW (U)

M-920419-09U-C (C) (2274)

CESIUM SUPPLY SYSTEM



CESIUM MELTING POINT - 28.6° C (83° F)

- **FLIGHT DATA**
  - HARDWARE IDENTICAL TO PLANNED NPBSE INJECTOR
  - STANDARD NASA INTEGRATION, LAUNCH, AND FLIGHT SERVICES REQUIRED
  - COMPATIBLE WITH EXISTING MIDDECK HARDWARE
  - 8 CU.FT., LESS THAN 240 LB (INCLUDING CONTAINER AND ATTACHMENT HARDWARE)
  - 28 VDC ONLY, STANDARD VIDEO (PAYLOAD REQUIRES CONDITIONING OF ORBITER DC POWER)
  - PASSIVE HEAT REJECTION FROM CSE CONTAINER AND ELECTRONICS PACKAGE BY RADIATION AND CONDUCTION INTO MIDDECK
  - ALL FLIGHT DATA WILL BE STORED IN PAYLOAD FOR POST MISSION ANALYSIS
  - MISSION SPECIALIST INITIATES, OPERATES, AND MONITORS EXPERIMENT VIA EXPERIMENT CONTROL AND DISPLAY PANEL
  - SECONDARY PAYLOAD, ANY SHUTTLE DURATION OR INCLINATION

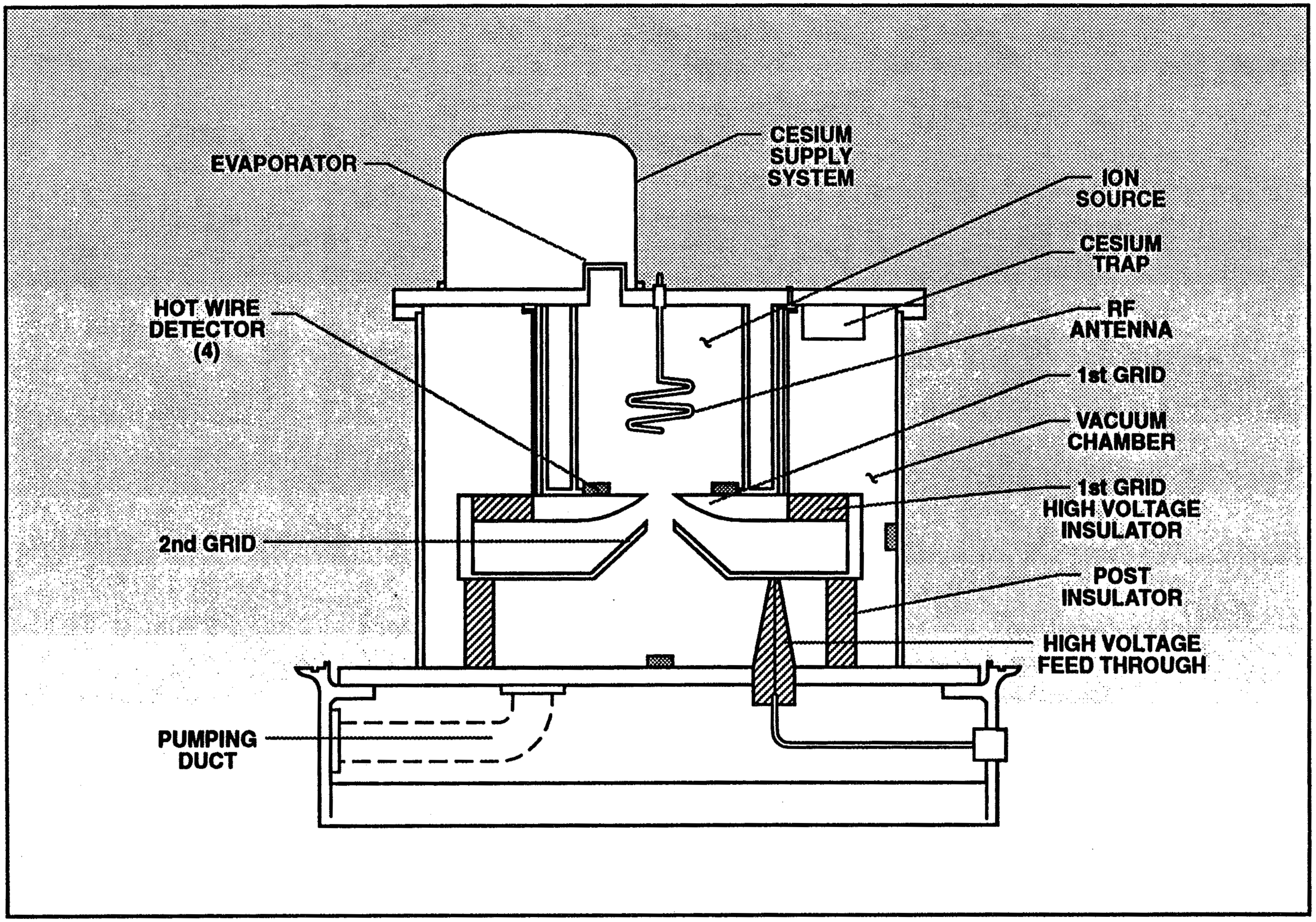
NEUTRAL PARTICLE BEAM ION SOURCE CESIUM SPACE EXPERIMENT (NPBCSE) SUMMARY (U)

M-920924-02U-A (C) (2274)

- HARDWARE READY FOR FLIGHT MID-1994
  - PROPER RESOURCES APPLIED TO ENSURE EFFICIENT DESIGN, DEVELOPMENT AND INTEGRATION OF NPBCSE HARDWARE
  - ON-ORBIT OPERATION WILL PROVIDE PROOF-OF-CONCEPT AND FLIGHT HARDWARE TO NPB PROGRAM

EXPERIMENT CONFIGURATION CESIUM SPACE EXPERIMENT (U)

M-920915-02U (C) (2275)



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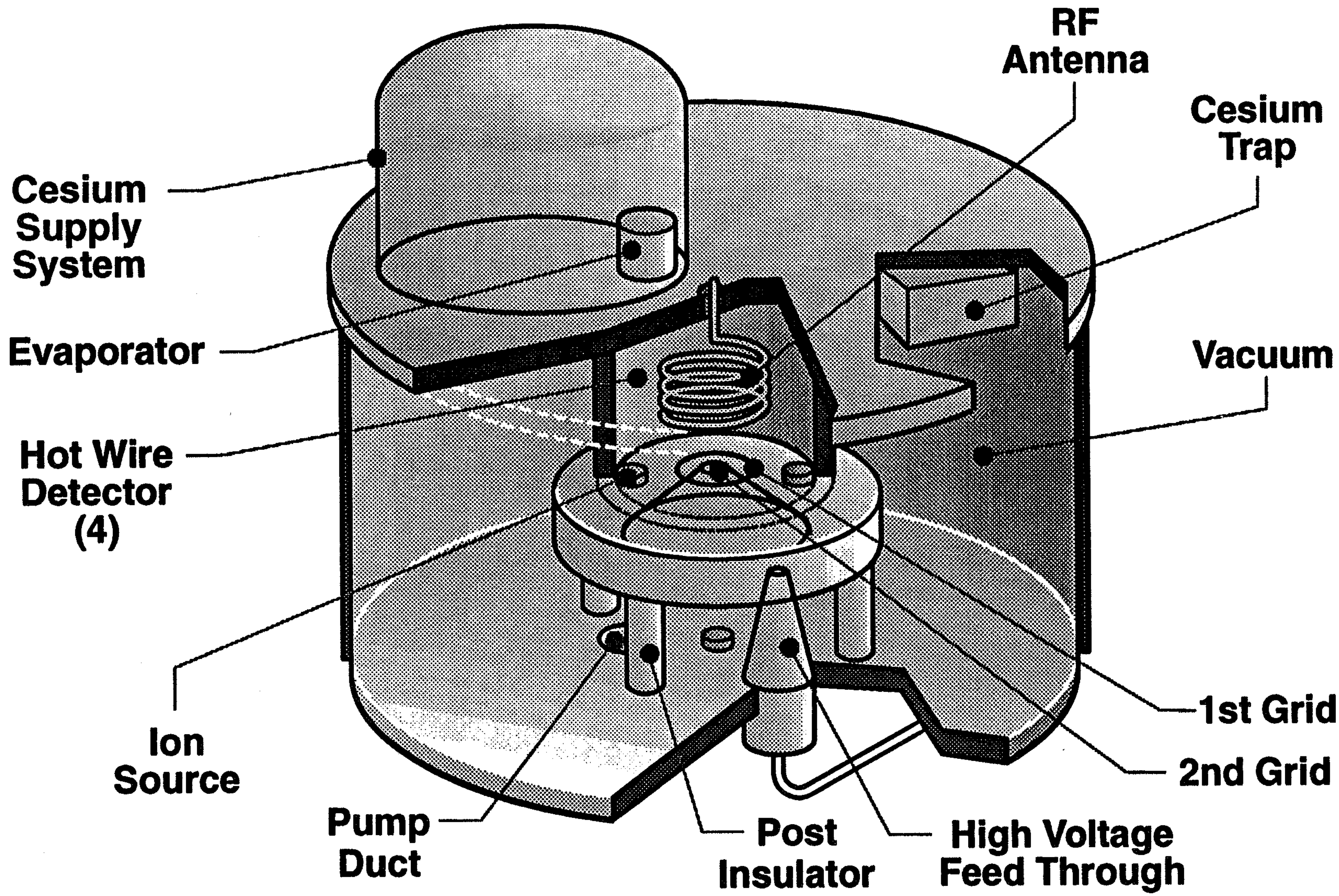
NEUTRAL PARTICLE BEAM ION SOURCE CESIUM SPACE EXPERRIMENT (NPBCSE) (U)

M-920419-08U-D (C) (2274)



NEP Space Test Program Objective

ТОПАЗ TOPAZ

The Objective Of The NEP Space Test Program Is To Launch A NEP Satellite Powered By A Russian Topaz II Reactor By December 1995

DUPLICATE – 33
OF
153229

NEP Space Test Program Goals

Primary Goals

- Demonstrate The Feasibility Of Launching A Space Nuclear Power System In The United States
- Demonstrate An Orbit Adjust Capability Using Nuclear Electric Propulsion
- Evaluate The In-Orbit Performance Of The Topaz II Reactor And Selected Electric Thrusters
- Measure, Analyze, And Model The NEP Self-Induced Environment

Secondary Goal

- Conduct A Space Science/Engineering Mission Compatible With The Primary Mission Requirements

Topaz II Description Fact Sheet

ТОПАЗ

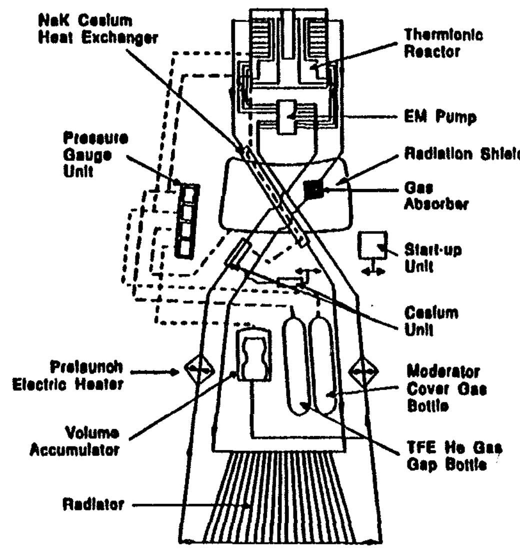
TOPAZ

Summary

The Topaz II is a Russian built thermionic space nuclear reactor. It is a single-cell design that has several advantages over multi-cell designs including: it permits full system qualification testing using electric heaters in place of nuclear fuel, it doesn't require early commitment of expensive enriched uranium (96% U-235), it allows removal of fuel for shipping and storage which improves the safety and safeguards environment, and the open cavity of the single-cell design facilitates fission gas escape (mitigating fuel swelling). The reactor is an epithermal system. The fuel loading is low (<27 kg). The output of the reactor is 28 VDC when operating at 135 kWth and 6 kWe. The reactor is cooled by flowing liquid NaK. The coolant loop is susceptible to single-point failures.

Key Terms and Acronyms

Thermionic: The physical process whereby heat energy is converted directly into electric energy via the mechanism of electron emission from an emitter to a collector.
TFE: Thermionic Fuel Element. The structure which contains the components required to produce and utilize the thermionic conversion process.
TSET: Thermionic System Evaluation Test. The facility used to perform full scale Topaz II system tests using electric heaters in place of nuclear fuel in the TFEs.
Single-cell/Multi-cell: Differing designs of the thermionic fuel elements where the TFE consists of a single energy converter versus several energy converters connected in series much as batteries stacked in a flashlight.
NaK: Eutectic composition of sodium and potassium metals which is used as the primary coolant in the Topaz II reactor.
Moderator: ZrH blocks that surround the TFEs and reduce neutron energies down to thermal equilibrium levels.
Reflector: Beryllium blocks above and below the moderator, and segments and drums which surround the reactor vessel that "reflect" neutrons back into the core region.
Core: The central region of the reactor consisting of the fuel, TFEs and moderator where the peak neutron flux exists.
Radiator: A skirt consisting of small pipes, welded to copper plates, carrying the NaK coolant which radiates the excess heat from the coolant to space.
Radiation Shield: Lithium hydride filled stainless steel casing that is used to reduce the level of neutron and gamma ray radiation intensity in the direction along the boom.



Topaz II System Diagram.

Key Facts

The Topaz II system consists of three principal components: the reactor, the radiation shield and the radiator/frame structure. Additionally, there are four fluidic systems used to support the Topaz operation: NaK coolant, cesium in the TFE interelectrode gap, and two bottled gas systems.

TFE emitter temperature is ~1600 C during operation and the collector temperature is ~900 C. There are ~11 liters of NaK coolant in the Topaz system at temperatures ranging between 500 C at the reactor inlet and 600 C at the outlet. The cesium system is designed to provide 2 torr of pressure within the TFE interelectrode gap, thus enhancing the thermionic conversion process.

The reactor mass is 290 kg, that of the radiation shield is 390 kg, and the radiator mass is 50 kg (filled with coolant). The frame structure has a mass of 45 kg.

NEP Space Test Program Fact Sheet

Summary

The NEP Space Test Program is sponsored by SDIO. The objective of the program is to launch an NEP satellite powered by a Russian Topaz II reactor by December 1995 for a cost of $150M, excluding the cost of the launch vehicle costs. The cost distribution is $80M for the satellite and $70M for the power system. The Applied Physics Lab (APL) is responsible for the satellite and the Phillips Lab (PL) is responsible for the power system.

Key Terms and Acronyms

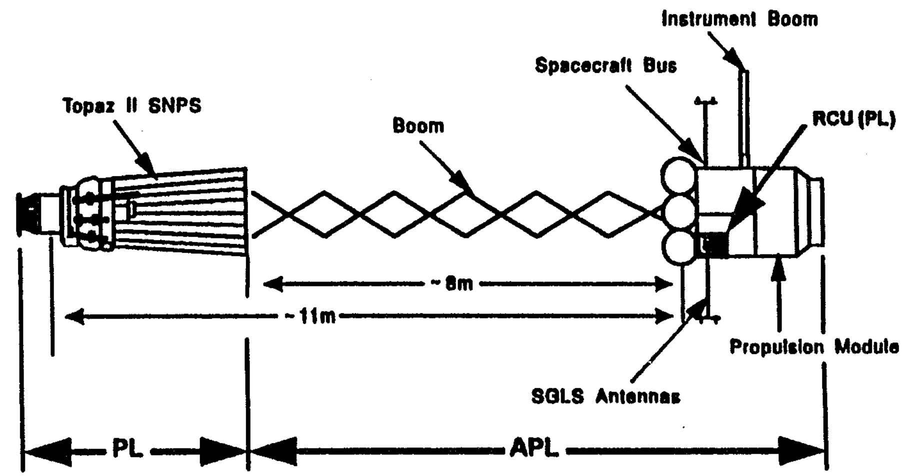
Topaz II: 6 kWe, Russian SNPS based on thermionic conversion
NEP: Nuclear Electric Propulsion
SNPS: Space Nuclear Power System
SDIO: Strategic Defense Initiative Organization
PL: Phillips Lab
APL: Applied Physics Lab
SNL: Sandia National Laboratories
LANL: Los Alamos National Laboratories
CDBMB: Central Design Bureau for Machine Building (St. Petersburg,)
KIAE: Kurchatov Institute of Atomic Energy (Moscow)
CoDR: Conceptual Design Review
PDR: Preliminary Design Review
CDR: Critical Design Review
PSA: Preliminary Safety Assessment (of the Topaz II)

Status

The Topaz Program was begun in April 1992. To date, the following milestones have been reached:

- Satellite CoDR (APL)
- Reactor CoDR (PL)
- Draft Topaz PSA (PL)

A program plan has been established for reaching the objective of launching the NEP satellite by December 1995.



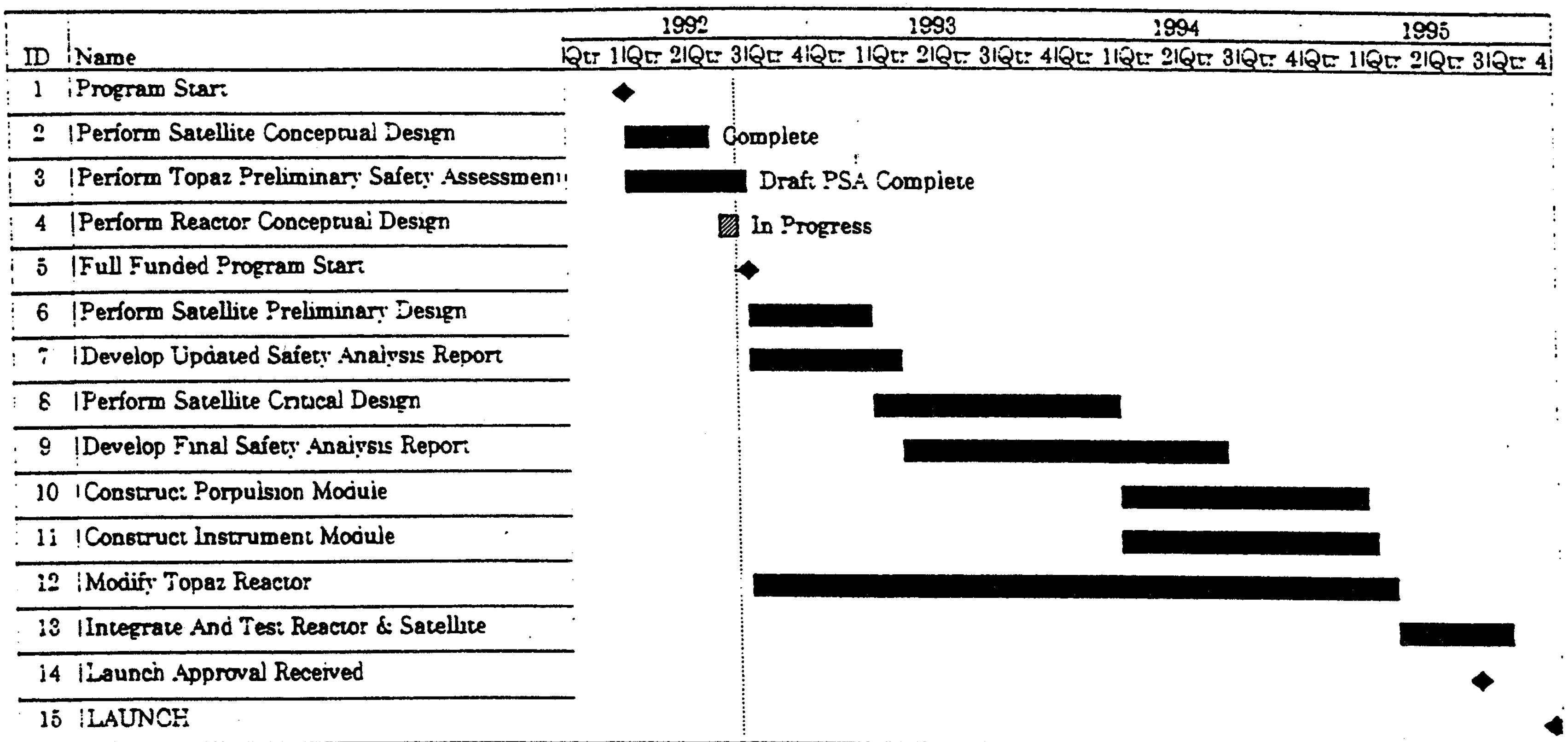
Topaz Satellite Configuration

Topaz Modifications

Four modifications will have to be made to the Topaz II power system:

- A reentry shield will be added to ensure the system meets the safety requirement of an essentially intact reentry (SNL),
- A neutron poison may be added to the core to prevent criticality in accident scenarios involving water immersion and flooding (LANL),
- A new reactor control system will be added because the Russian system is not space qualified and does not meet US safety standards (PL, SNL), and
- US fuel may be used because of difficulties associated with obtaining special nuclear material from Russia (LANL).

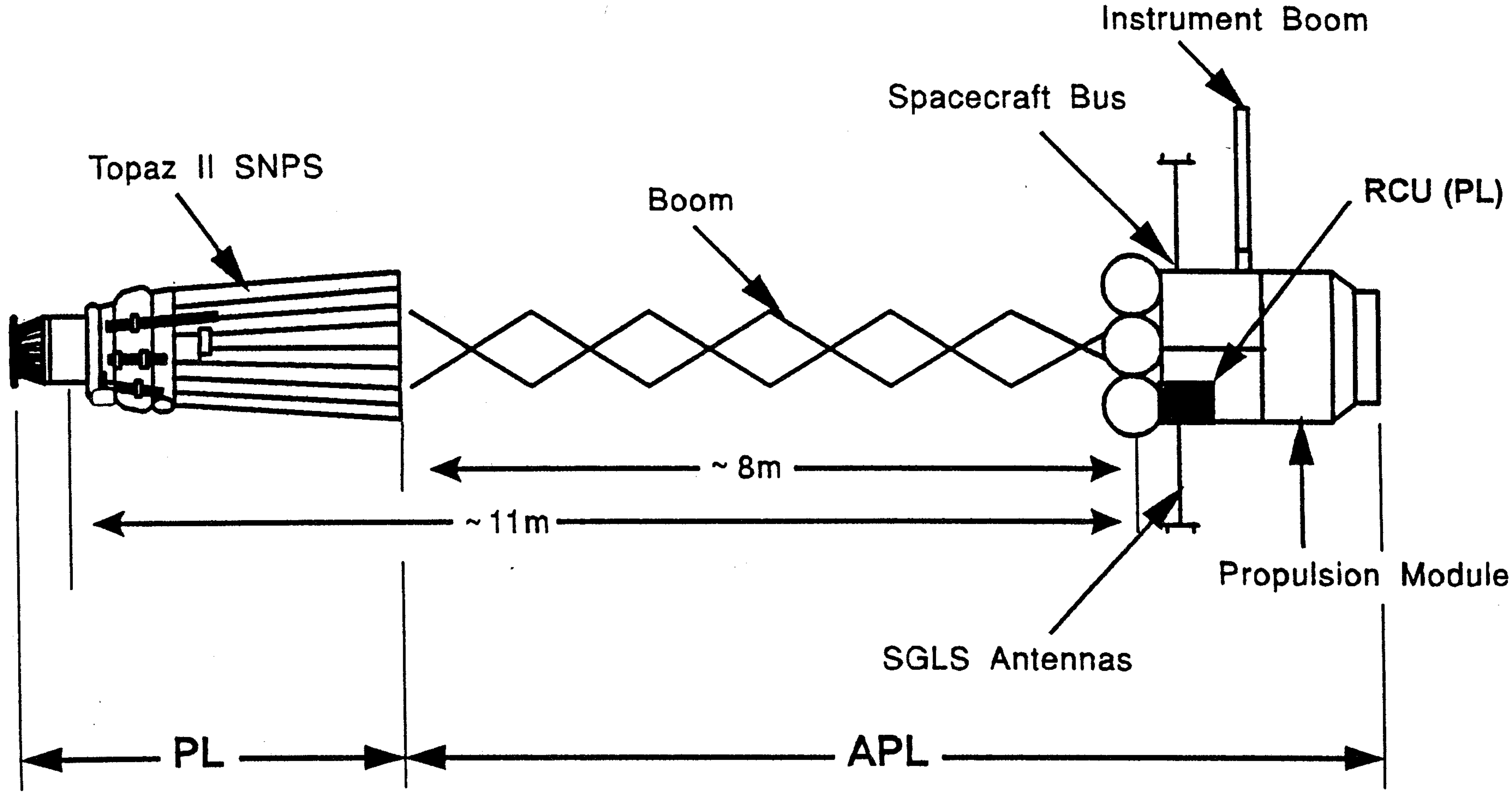
NEP Space Test Program Schedule



Major Milestones For The NEP Space Test Program

✓August 92	Satellite Conceptual Design Review
✓September 92	Reactor Conceptual Design Review
✓September 92	Topaz II Preliminary Safety Assessment Complete
October 92	Full Funded Start
April 93	Satellite Preliminary Design Review
May 93?	Updated Safety Analysis Report Complete
April 94	Satellite Critical Design Review
September 94	Final Safety Analysis Report Complete
April 95	Propulsion Module Complete
May 95	Instrument Module Complete
May 95	Reactor Modifications Complete
September 95	Topaz II Modifications Complete
September 95?	Launch Approval Received
December 95	Launch

NEP Satellite Configuration



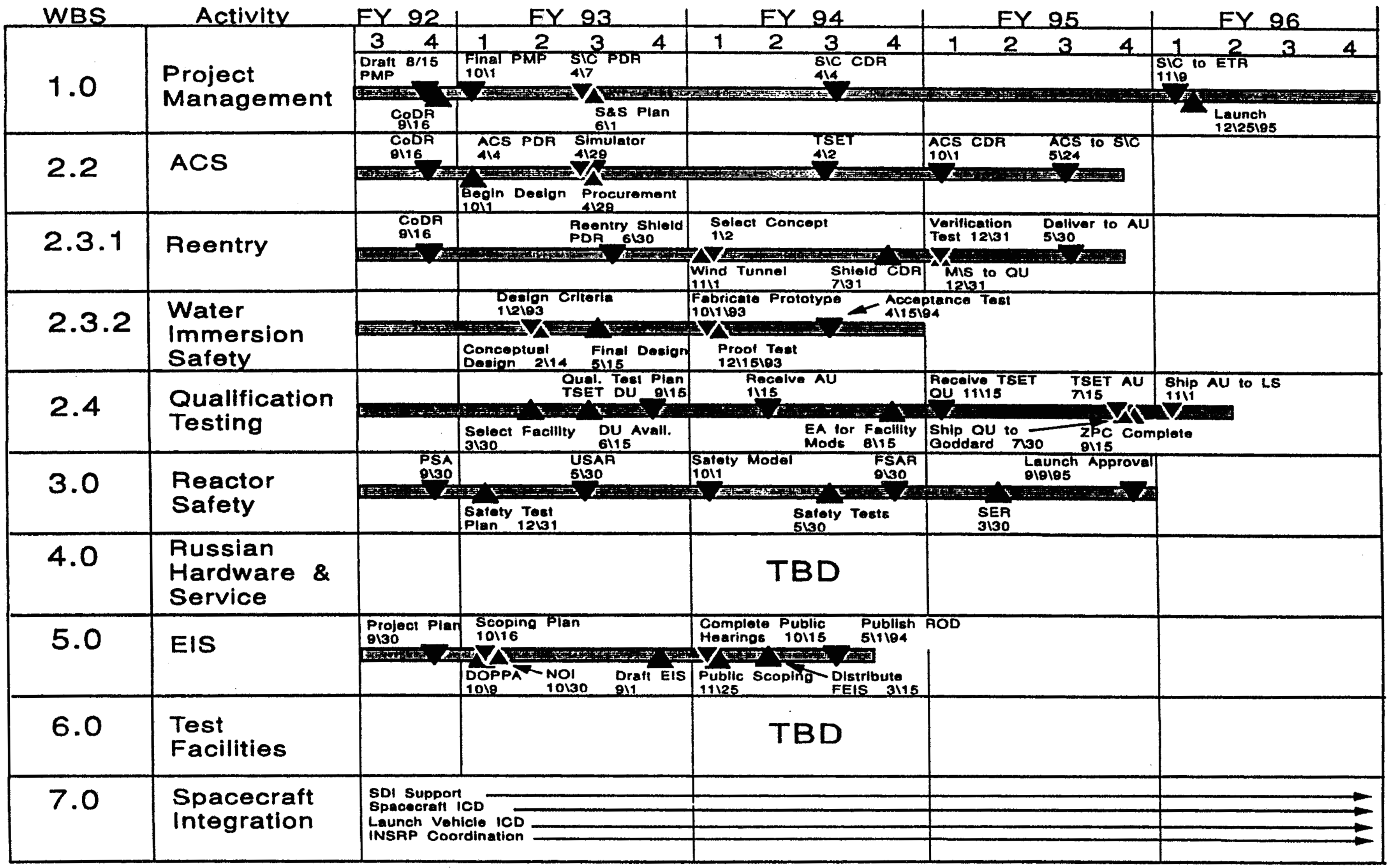
ТОПАЗ

TOPAZ

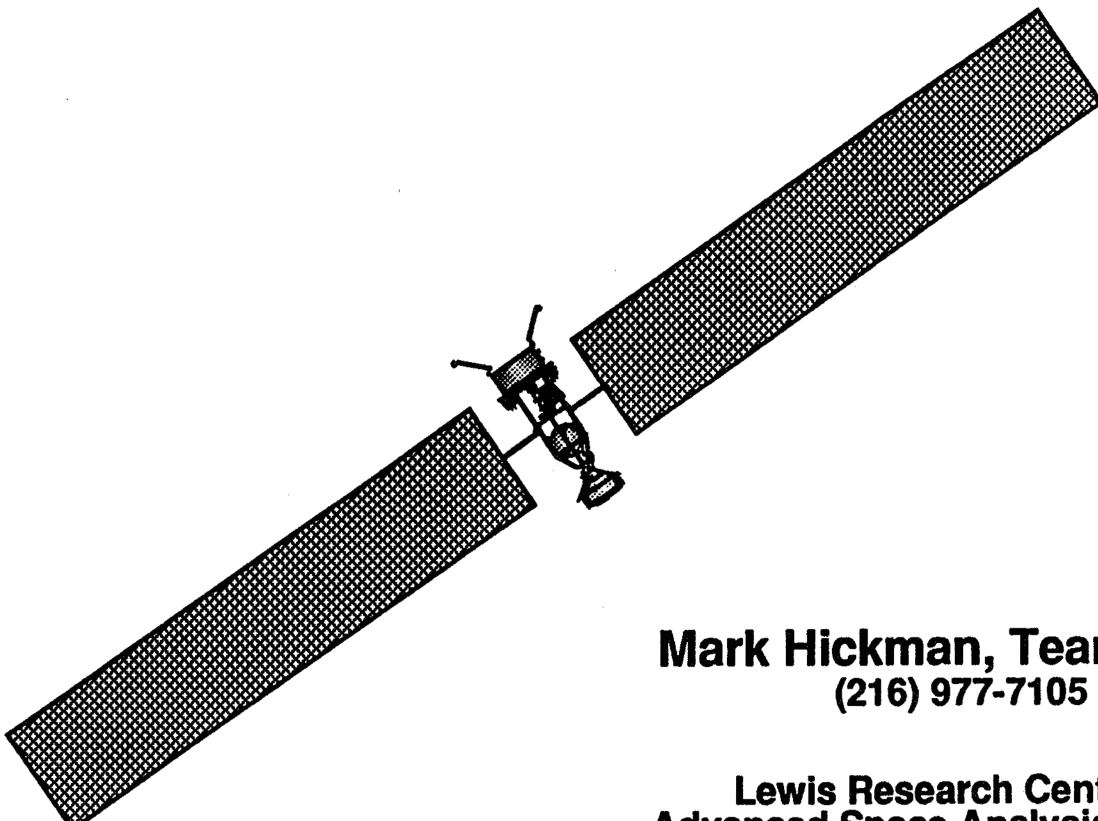
Mission Profile

- Launch into Low Earth Orbit (LEO) (nominal orbit 1,600 km @ 28.5°)
- Confirm nuclear safe orbit (by ground skin track radar)
- Start up TOPAZ 2 reactor (by ground command)
- Evaluate performance of reactor and spacecraft bus
- Begin electric propulsion orbit raising; spiral out with thrust vector parallel to velocity vector
- Test each type of engine for 1,000 hours
- Duty-cycle thrusters to separate reactor measurements from thruster measurements; ~ 97% duty cycle
- Perform life testing on selected engine types; use high thrust engines first, low thrust engines later
- Above 6.6 R_E, begin twice per orbit yaw maneuver to change inclination, simultaneously raising orbit

Preliminary Topaz II Reactor Program Schedule



Transfer Orbit Plasma Interaction Experiment (TROPIX)



Mark Hickman, Team Lead
(216) 977-7105

Lewis Research Center
Advanced Space Analysis Office

N93-28734

TROPIX Science Mission Description

- Map the charged particles in Earth's magnetosphere from LEO to GEO at high inclinations
- Measure plasma current collection and resulting shifts in vehicle electrical potential from LEO to GEO across range of orbital inclinations
- Study spacecraft interaction with plasma environment using SEP thrusters as plasma contactors
- Measure array degradation over mission duration
- Evaluate the potential of various microelectronics, spacecraft components, and instruments for future space missions
- Demonstrate SEP technology applied to a flight vehicle

TROPIX Builds On Previous Investigations

- **CRRES (Combined Release and Radiation Effects Satellite)**
  - Joint program of MSFC and DOD/Air Force Space Test Program designed to understand problems of highly sensitive electronics and sensors in hostile radiation environment
  - Collected data on dynamics, structure, and chemistry of near-earthspace environment, and studied the survivability of electronics
  - Atlas/Centaur launch, July 25, 1990
  - Elliptical GTO: 400 km by 35,800 km

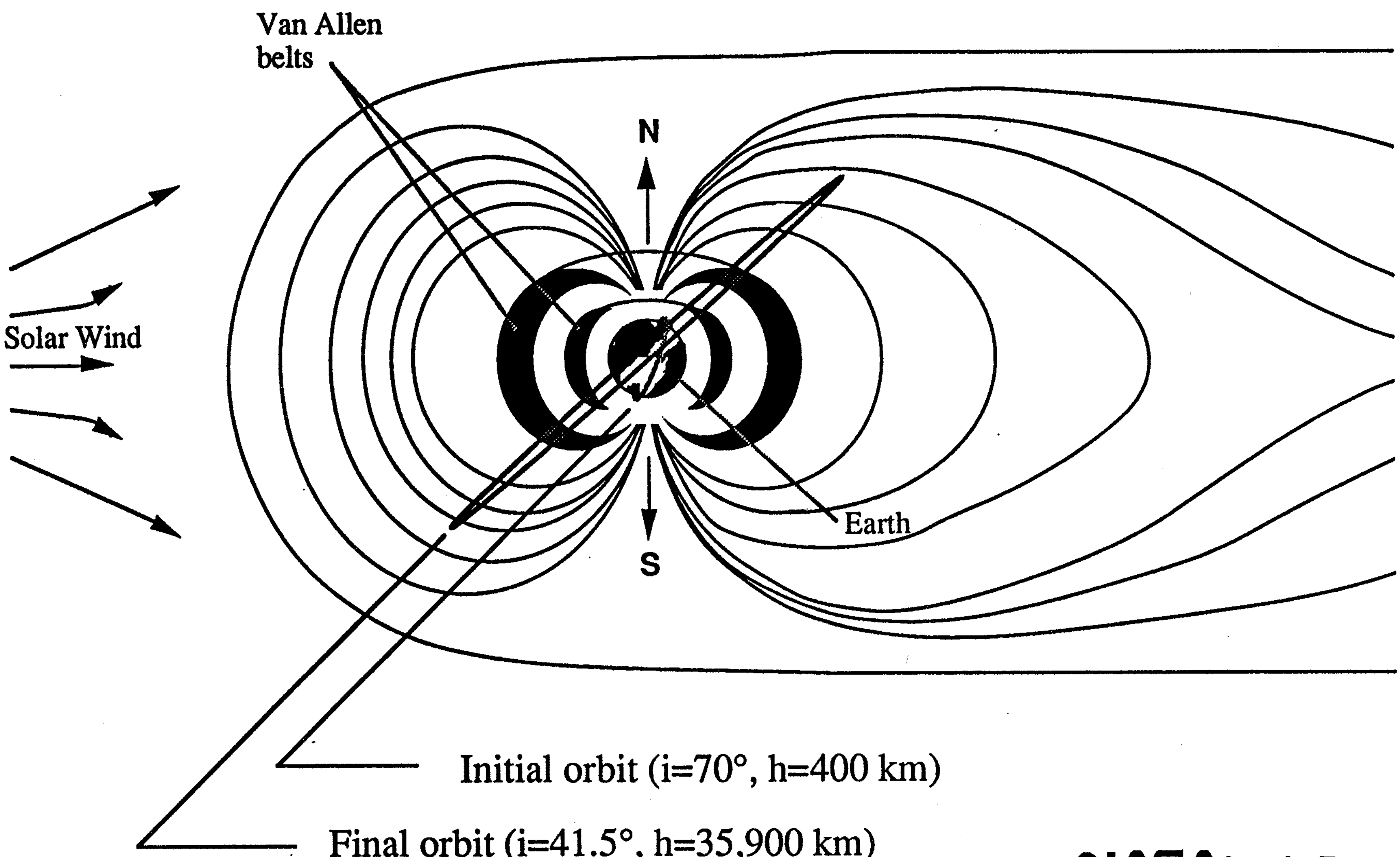
- **PASP+ (Photovoltaic Array for Space Power plus diagnostics)**
  - Pegasus launch, May 1993
  - Elliptical orbit: 190 nm (352 km) by 1050 nm (1945 km) at 70° inclination

- **SAMPIE (Solar Array Module Plasma Interaction Experiment)**
  - Investigate the arcing and current collection behavior of materials and geometries likely to be exposed to the LEO plasma on high voltage space power systems in order to minimize adverse environmental interactions
  - To be flown aboard Shuttle, January 1994

TROPIX Will Augment Previous Missions

- TROPIX will provide a clearer understanding of the effects induced by energetic particles and solar radiation
  - Damage caused to satellite electronics
  - Degradation to solar arrays
  - Hazards to people in space
- TROPIX altitudes and inclinations are unmeasured
- Mission length of approximately 1 year sufficient to take time dependent data
- Spacecraft power over 3 kWe
  - Enhances current collection measurements
  - Conditions more applicable to real future spacecraft than at lower power

Initial and Final TROPIX Orbits Within Earth's Magnetosphere



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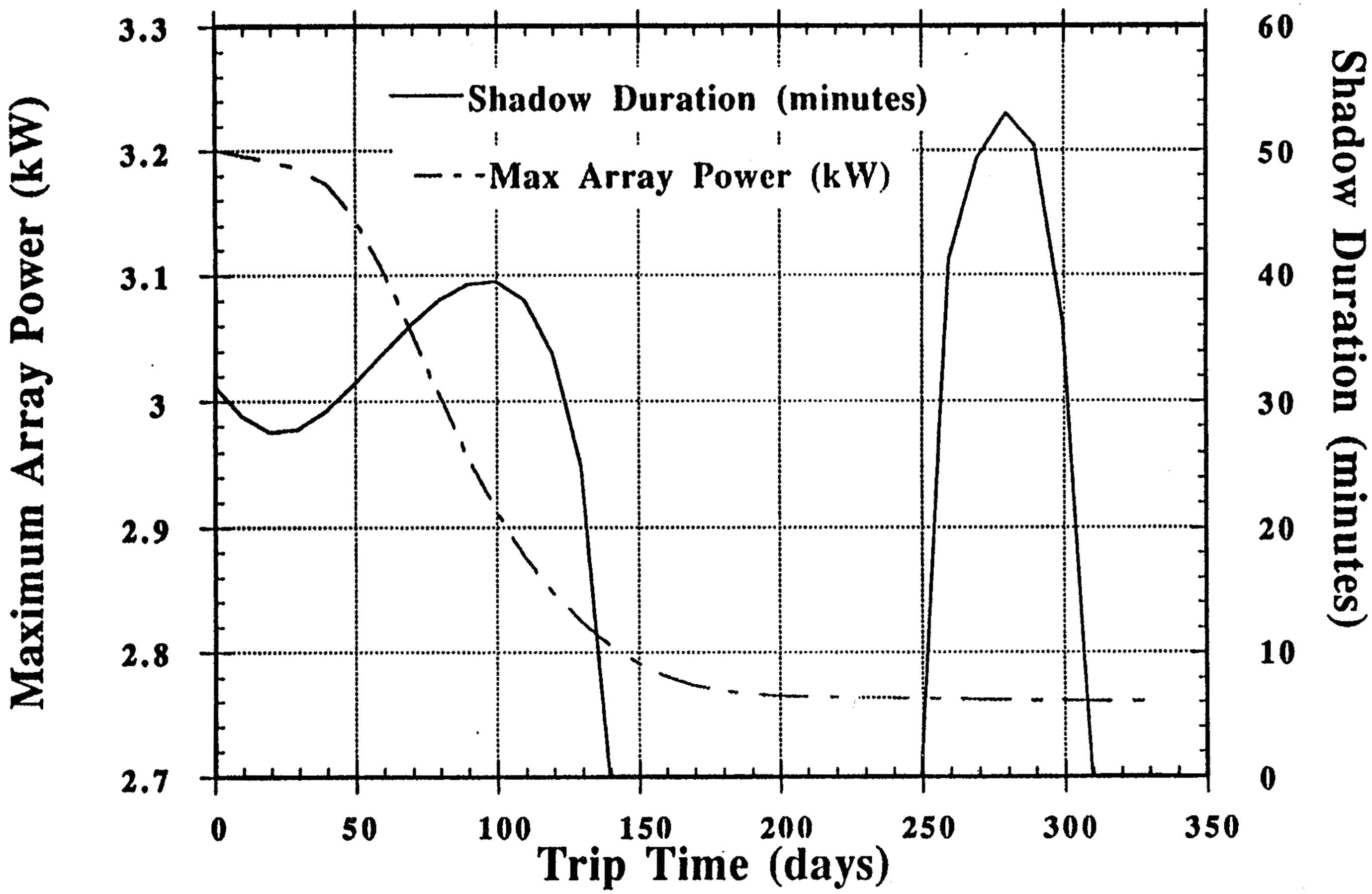
Advanced Space Analysis Office

Solar Array Degradation and Shadow Duration vs. Trip Time

- **Mission Trajectory Optimized for Science Requirements**



Potential Customers

- **Applied Science**
  - **NASA & DOD** — SEP has application to highly maneuverable observation satellites
  - **Mid and high orbit spacecraft designers** — environmental interactions impact spacecraft and platform design
  - **Solar cell designers and array manufacturers** — measurements of different solar cell types as they degrade through the Van Allen radiation belts may lead to improved design
- **Theoretical Science**
  - **Plasma physicists** — characterization of plasma field at altitudes and inclinations not previously measured
  - **Geophysicists** — real time data on magnetospheric field line vibration and charged particle population

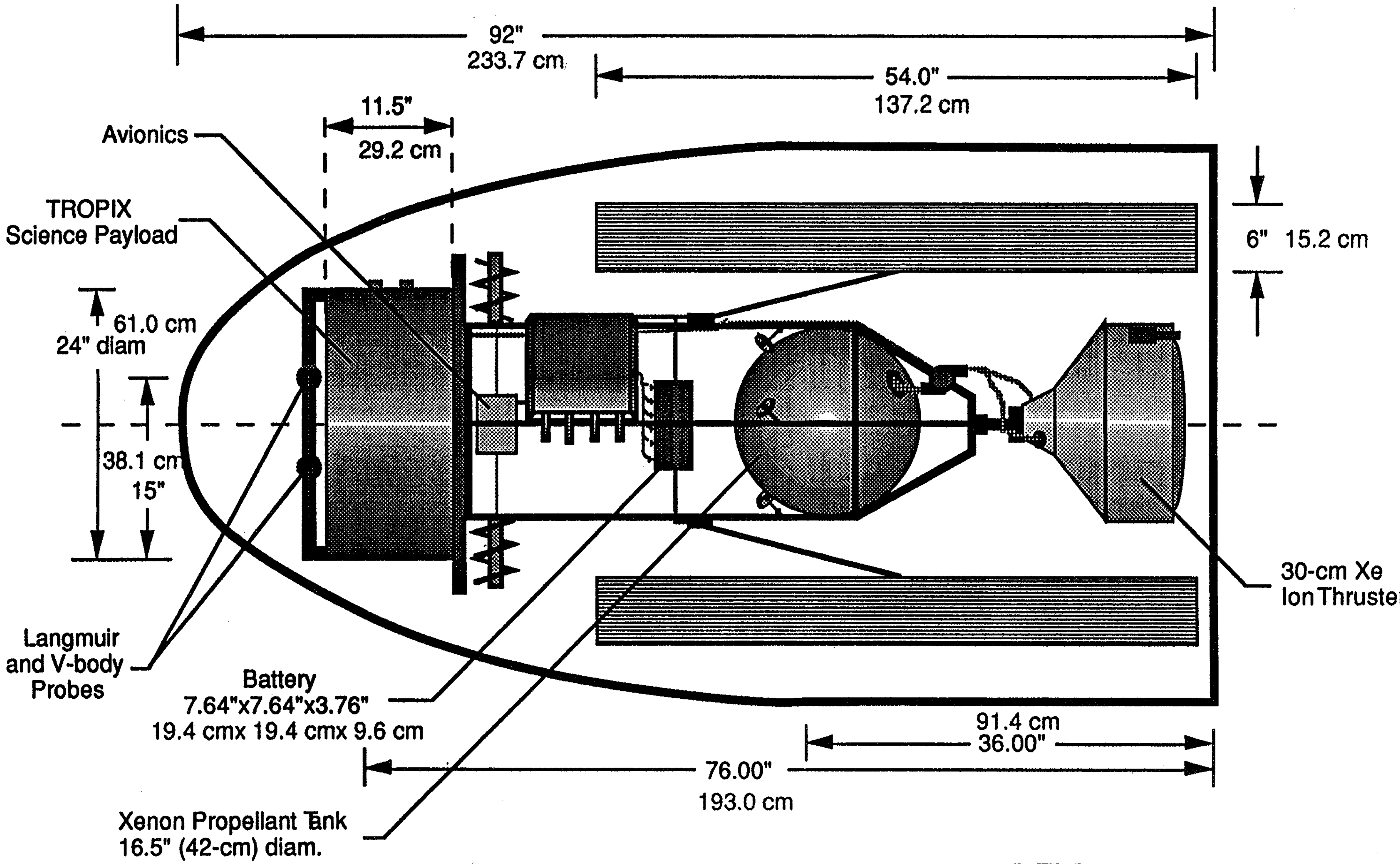
TROPIX Vehicle
To Use Solar Electric Propulsion (SEP)

- Maneuverability of SEP vehicle to different orbital planes enables large variation in data collection location
- Chemical mission designed to approximate SEP mission requires over 4 times initial mass in LEO
- Ion thrusters act as plasma contactors enhancing environmental interaction study
- Arrays
    - Act as solar cell degradation experiment
    - Large electrical power capacity for payload satisfies high power requirements
- Power availability allow mission operation design flexibility
- Slow transit time of electric propulsion permits many months of data collection
- Evolvable to other near earth, and possibly, lunar missions

TROPIX Instrumentation

- **Science instruments based on SAMPIE, CRRES, and PASP+**
  - **V-body probe:** measure the potential of spacecraft relative to the surrounding plasma environment
  - **Langmuir probe:** measure plasma density and temperature
  - **Particle detectors:** characterize total plasma environment local to the vehicle to understand thruster efflux interactions with the plasma environment
  - **Four dosimeters:** determine the high energy electron and proton dose rates through the radiation belts
  - **Electrometers:** read the electric currents of test samples
  - **Test samples**
    - **Indium Phosphide solar cells without cover slides**
    - **"Plasma-proof" array test section**
    - **Array test section with overhanging coverslides**
- **Gallium arsenide arrays used as degradation test sample**
- **Xenon ion thrusters as plasma contactors**

TROPIX Spacecraft Stowed Within Pegasus XL Fairing



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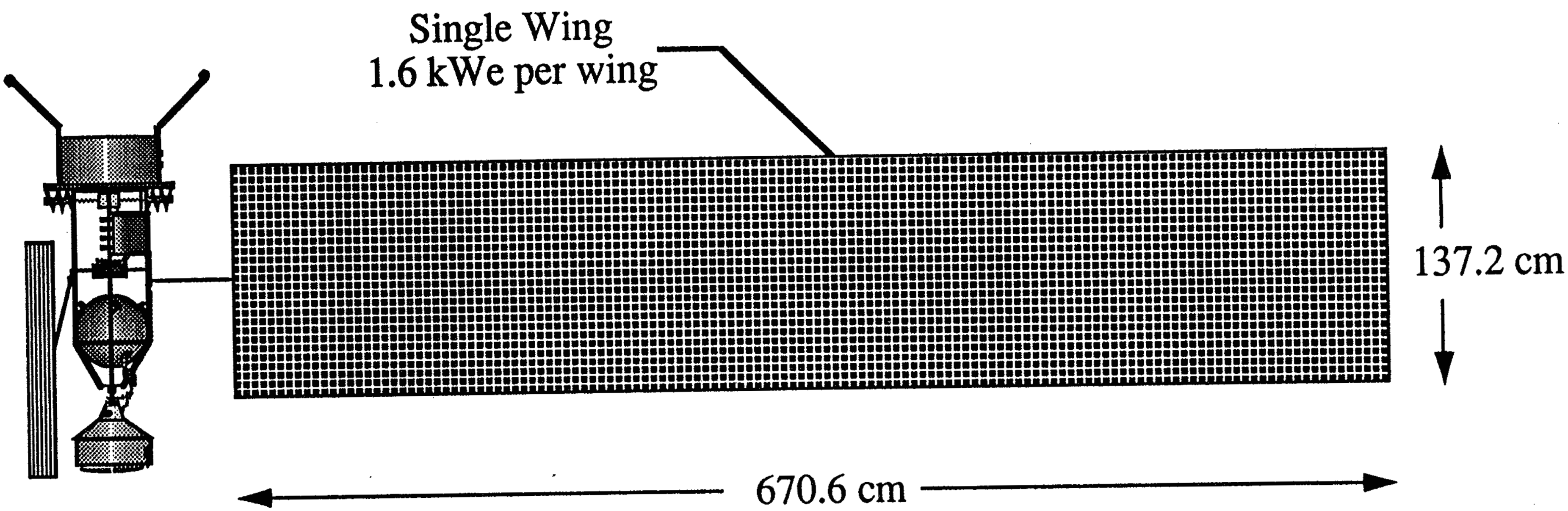
TROPIX With One Wing Deployed



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Advanced Space Analysis Office

Next Steps

- Complete Team/Form Collaborations
- Characterize TROPIX experiment hardware interfaces and requirements
- Prepare for Phase A
  - Complete preliminary trades and analyses
  - Complete preliminary program plan
- Refine TROPIX concept
  - Determine and design components
  - Complete packaging scheme
- Prepare for Concept Design Review

ORBITAL DEBRIS ENVIRONMENT MONITOR
ODEM

Dr. John P. Oliver
Institute for Space Science and Technology,
Gainesville, Florida

Presented at the Flight Experiments
Technical Interchange Meeting, Monterey, California
5-9 October, 1992

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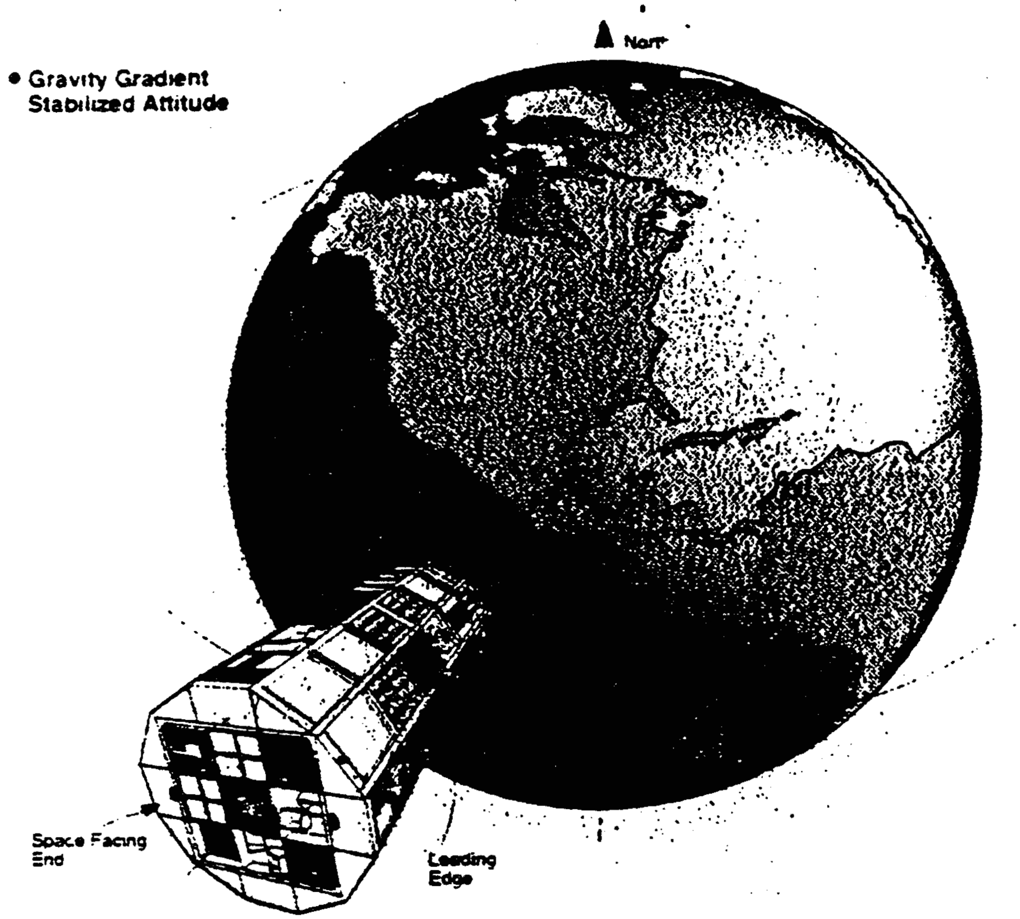
LDEF Orbital Flight Orientation



LDEF
Long Duration Exposure Facility

Orbital Attitude: Gravity Gradient Stabilized
(Schematic)

Deployed: April 7, 1984, at 250 nmi (463 km)

Retrieved: January 12, 1990, at 178 nmi (330 km)

Total Duration in Orbit: 70 months (5.8 yr)

Atomic Oxygen Fluence (atoms/sq cm): $\approx 10^{22}$

Orbital Ram Velocity: ≈ 8 km/s

LDEF Structure:

- Material: 6061-T6, Aluminum
- Coating: (Interior) Chemglaze Z306 (Flat Black)
- Size: (12-Sided Cylinder)
  - Length: 30 ft (≈ 9 m)
  - Diameter: 14 ft (≈ 4 m)
- Weight. (With Experiments) 21,393 lb (9,704 kg)

Number of Experiments: 57

Number of Investigators: >200

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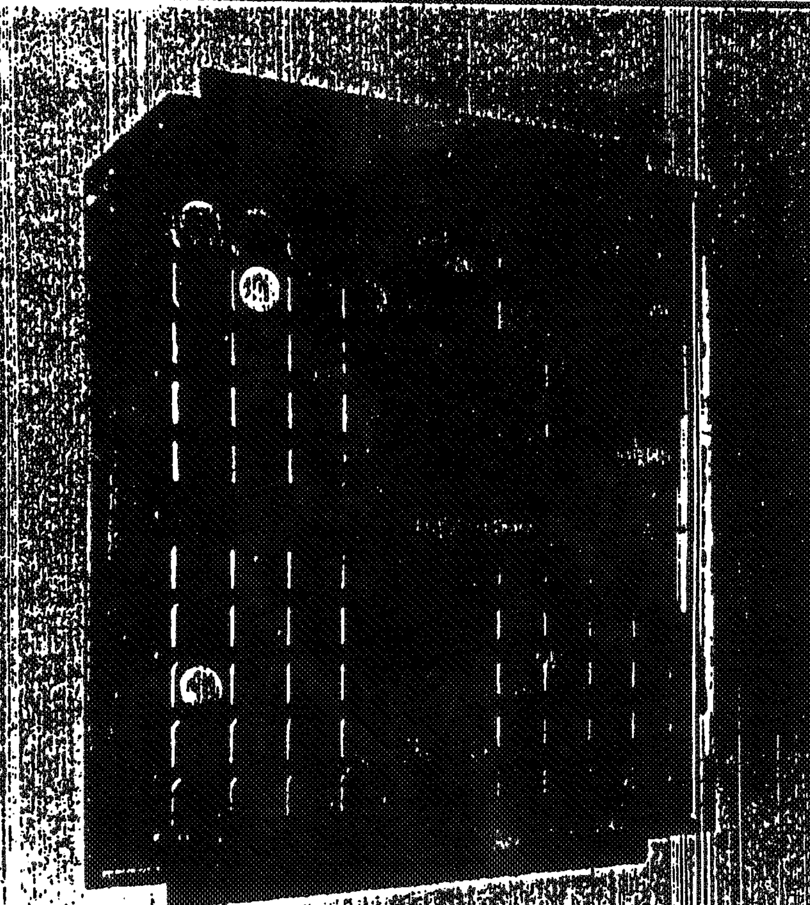
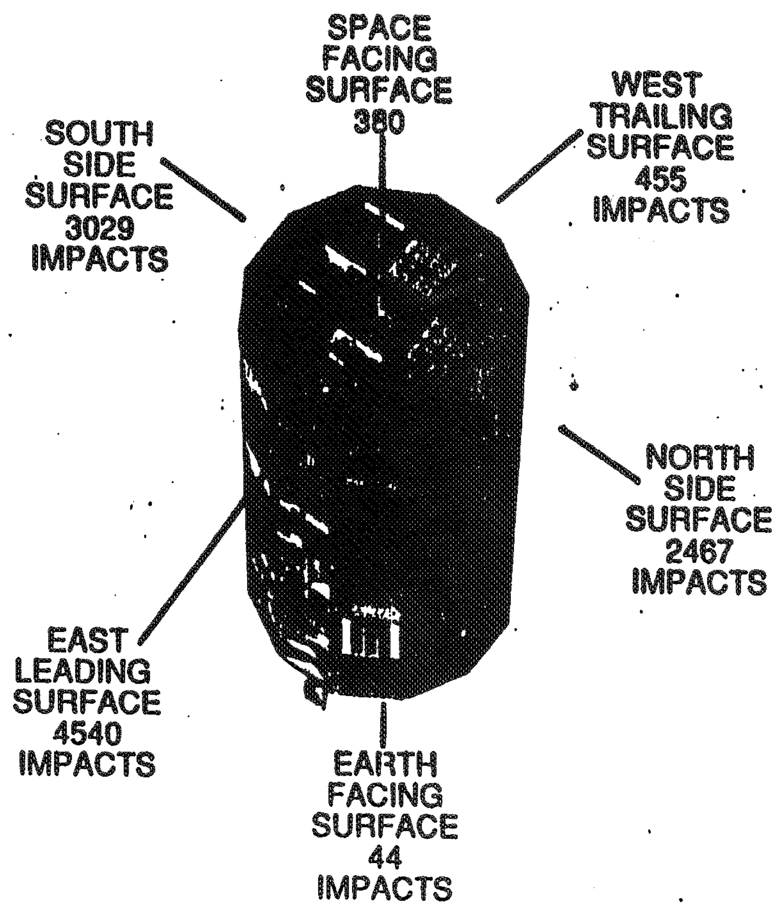
INTERPLANETARY DUST EXPERIMENT

• Detector Arrays Mounted on 6 Sides of LDEF



DETECTOR ARRAY MOUNTED ON EARTH SIDE OF LDEF



FLUX COUNTS FROM 4,000 ANGSTROM THICK MOS DETECTORS

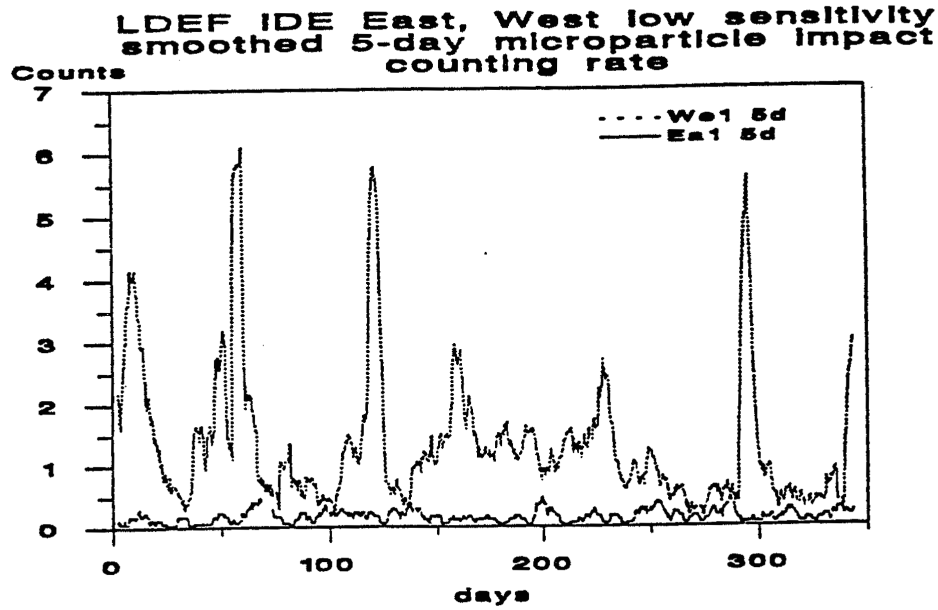


Figure 2a. Five day smoothed flux on the East and West low sensitivity IDE sensors as a function of time.

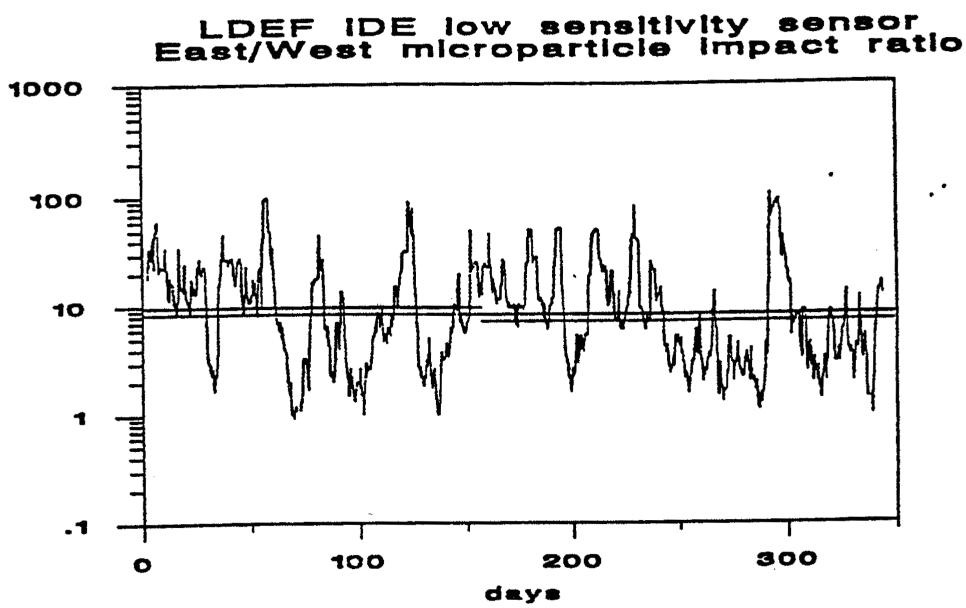
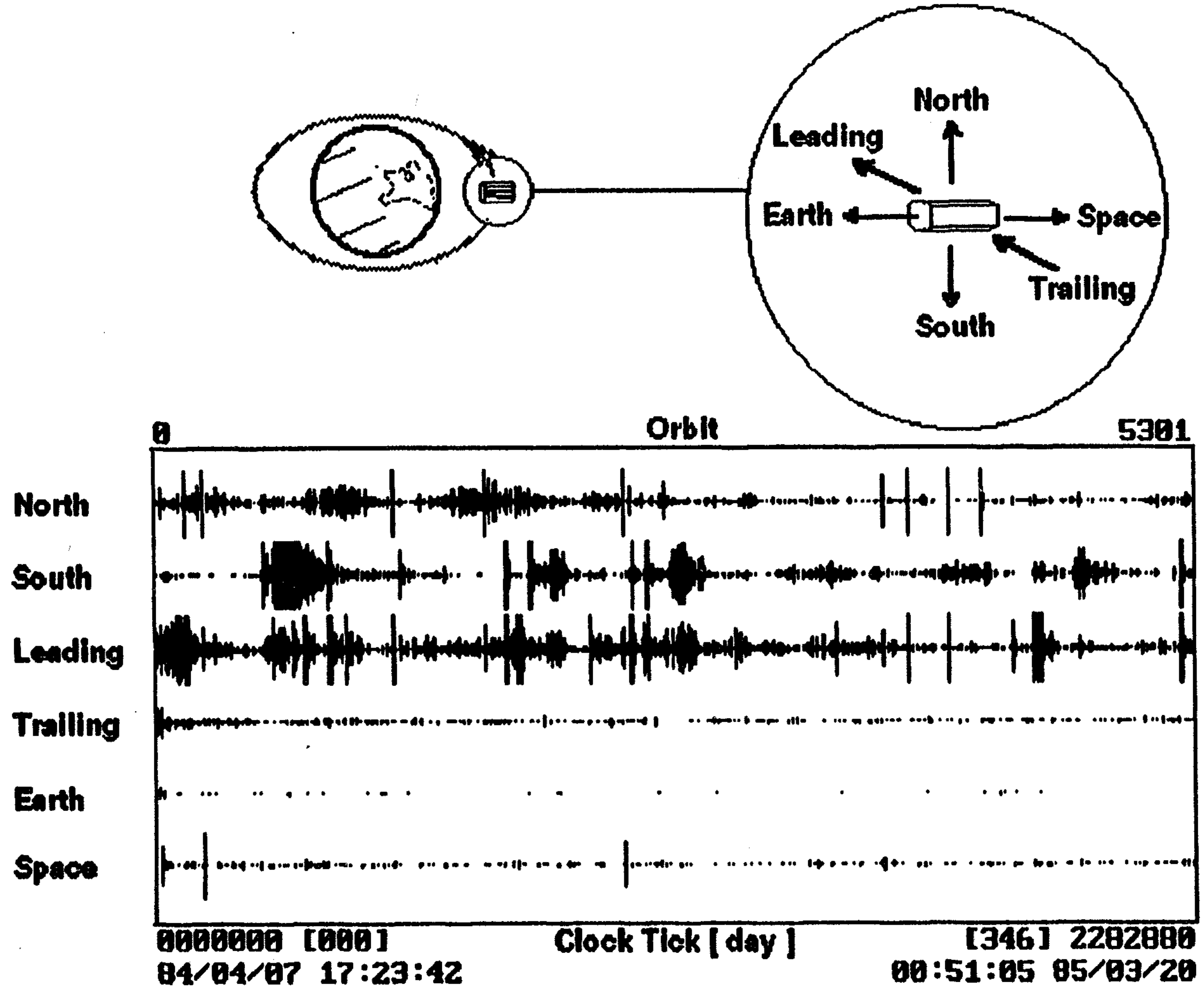


Figure 2b. Five day smoothed ratio of East and West low sensitivity sensor microparticle impact fluxes as a function of time. Horizontal lines mark mean values of full data set and the first and second halves of the data set..

"Seismograph" Plot



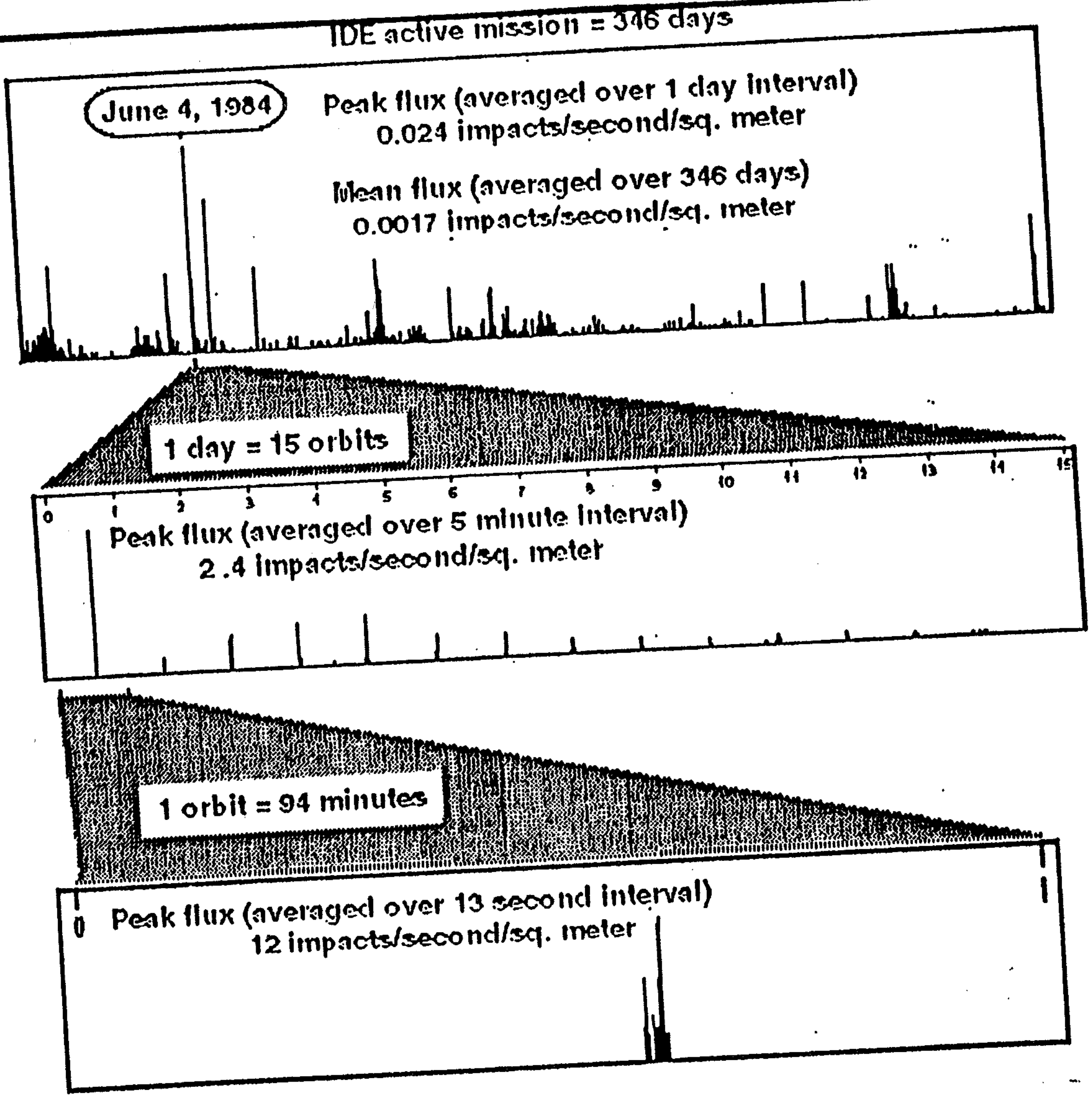


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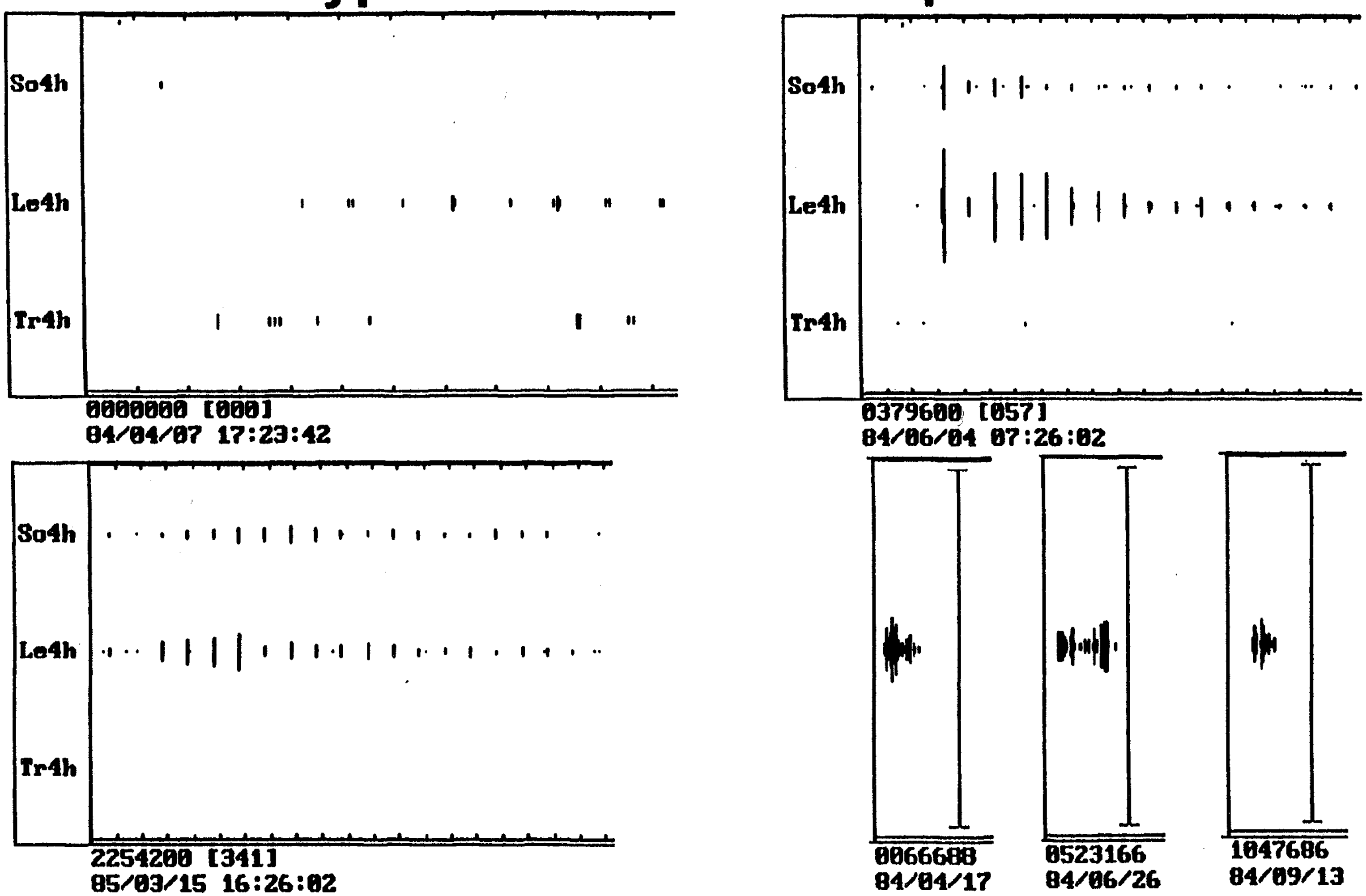
Figure 2. Observed activity on the leading (ram) edge of LDEF as recorded by the 0.4 µm thickness detectors of IDE.

Typical "MOEs" and "Spikes"



Time Full Scale:MOE ≈ 1 day; Spike ≈ 10 minutes

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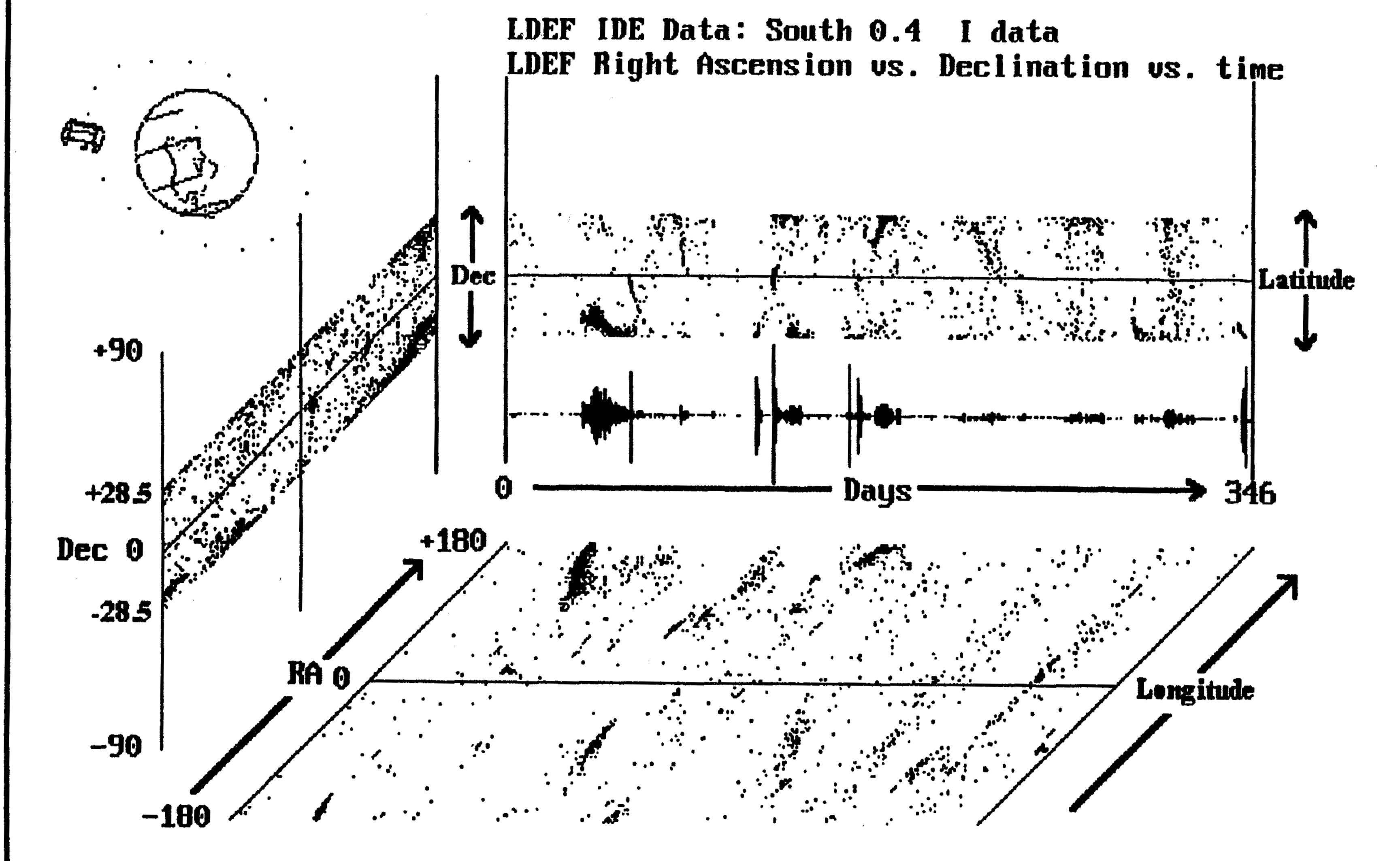
LDEF IDE Data: South 0.4 I data
LDEF Right Ascension vs. Declination vs. time
Dec
Latitude
+90
+28.5
Dec 0
-28.5
-90
0
Days
346
+180
RA 0
-180
Longitude

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Orbital Debris Clouds

- Greatly Increased Impact Rates Localized In Time and Space
  - Events Occur Every 94.1 Minutes
  - Typical Event Duration; 3 to 5 Minutes (1500 to 2500 Km)
- Events Occur in Same Place Each Orbit
  Relative Activity on Differing Surfaces
  May Yield Apparent Source Direction
- Precession Allows Mapping In Space
- May 13th Swarm . . . ≈30° Orbital Inclination
- June 4th B Event . . . ≈65° Orbital Inclination

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Mapping and Modeling of Orbital Debris Clouds

- Exploit the Unique IDE Spatio-Temporal Dataset
- Identify and Categorize Cloud Events
- Relate Cloud Events to Sources
    - Launches to LEO
    - LEO to GEO Insertion
    - Accidental Disruption
    - Deliberate Disruption
- Analyze Cloud Evolution and Dispersion
- Predictive Modeling of Clouds - Space Weather Prediction
- Statistical Prediction of Total LEO Debris Environment

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Solar Maximum Mission Spacecraft

- In same orbital plane, 15 km "above" LDEF at LDEF release
- HAO Coronagraph images showed near-field particles and particle clouds
- Although originally attributed to local contamination, statistical analysis suggests some were orbital debris particles
- Correlation of LDEF IDE data with SMM Coronagraph data taken during 1984-1985 can confirm orbital debris origin
- SMM Coronagraph data available for period from 1980 through 1989; allowing assessment of changes in

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SynMOD (precursor to ODEM)

- *Syn*optic *M*onitoring of *O*rbital *D*ebris
- High time-resolution monitoring of near-Earth small particle impacts
- Uses proven MTS/Explorer 46/IDE sensors
- Selected for Eureca-2

ODEM

- *O*rbital *D*ebris *E*nvironment *M*onitor
- Adds impact energy/size/mass discrimination to SynMOD
- Standardized, Modular system
- Can easily mount on any Bus

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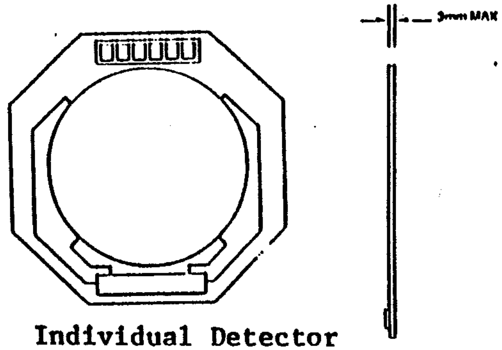
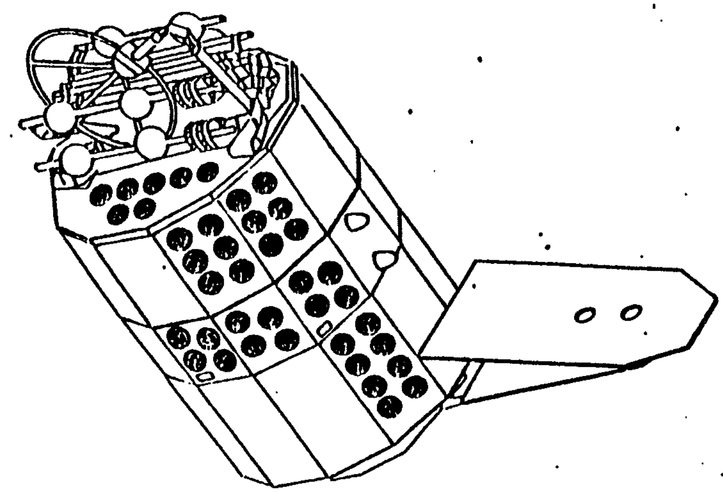
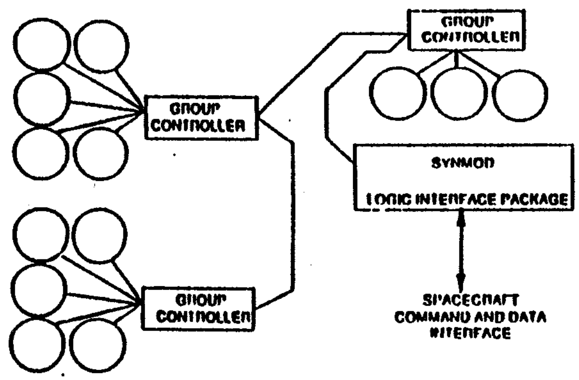
22. PICTORIAL



Individual Detector Element



Possible Configuration Options



System Architecture

DD FORM 1721 AUGUST 1990
PREVIOUS EDITIONS ARE OBSOLETE

Security Classification *(When data entered)*

RESOURCE REQUIREMENTS

	SynMOD	ODEM
Command and Data Uplink:	none	none
Telemetry Downlink:	< 250 Kbits/day	same
Power:	nominal 2 watts	same
Mass (per module):	< 0.75 kg	same
Mass (electronics):	< 1.0 kg	same
Volume (per module):	.3 x .3 x .003 m	same
Stabilization Required:	none	none
Desired Configuration:	12 modules	same

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omit

SESSION 8:

SPACE OPERATIONS

Co-Chaired by:
Dr. Kumar Krishen, NASA Johnson Space Center
Dr. W. Carter Alexander, USAF Armstrong Laboratory

NASA In-STEP

Permeable Membrane Experiment

Presented at
NASA/DoD
Flight Experiment Technical Interchange Meeting
Monterey, CA
October 8, 1992

Boeing Defense and Space Group
Kent Washington

Agenda

BOEING

- Experiment Overview
- Membrane Phase Separation Experiment
- Membrane Diffusion Interference Experiment
- Membrane Wetting Experiment
- Summary and Conclusions

Experiment Background

BOEING

History

Announcement of opportunity	11/89
Submittal of proposal	12/89
Notification of award	5/90
Contract negotiations begun	3/91
Contract start date (Phase B)	9/91
Phase B completion	5/92
Phase B extension start date	9/92
Phase B extension expected completion date	11/92

Experiment Design
Experiment Project

BOEING

Problem Statement

- There is a need for compact, reliable, and efficient technologies.
  - Advanced life support
  - Life sciences facilities
- Membrane technology meets this need in the following areas:
  - Phase separation
  - Fluid degassing
  - Particulate removal (including micro-organisms)
  - Ion transfer
- Membrane performance may be compromised by multiple phases.
  - Gas/liquid/membrane interface
  - Effect on phase separation and ion transfer efficiency
  - Area of greatest influence by presence of gravity

Experiment Design
Experiment Project

BOEING

Project Objectives

- Primary
  - Determine influence of different phases at membrane surface
  - Provide information on performance and possible problems
  - Study three areas of critical membrane design concern:
    - – Phase separation.
    - – Diffusion.
    - – Wetting.
  - Use these data and provide data to other design engineers
- Secondary
  - Provide a reusable membrane experiment package

9-B34280pk34-130

Experiment Design
Experiment Project

BOEING

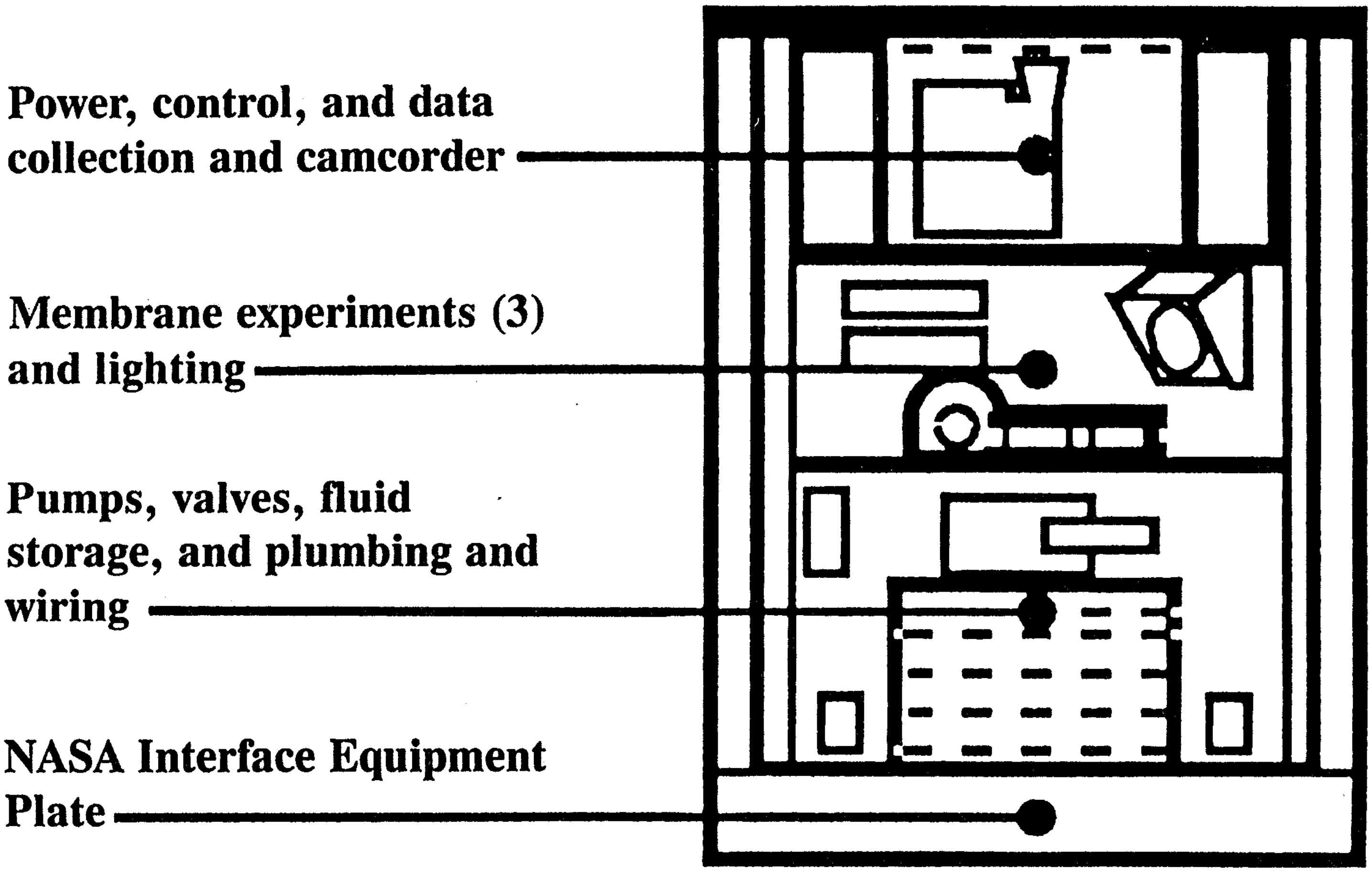
Experiment Description

- Three experiments packaged within a single shuttle CAP canister:
  - Dual-membrane gas/liquid phase separator
  - Membrane diffusion interference by gas bubbles
  - Membrane fluid wetting behavior
- Standalone Complex Autonomous Payload (CAP) carrier
  - Battery power
  - Passive thermal control
  - Embedded data acquisition and control
  - 8-mm video camcorder for visual record
  - Experiment package initialized from aft flight deck

Experiment Design
Experiment Project

BOEING

Experiment Package—CAP Canister Section



Experiment Description

No. 1 Dual-Membrane Gas/Liquid Phase Separator

BOEING

Problem Statement

- Free-gas contamination of liquid systems
- Gas interference with transport processes
- Difficulty of gas elimination in microgravity
- Drawbacks of existing approaches
  - EMU gas trap
  - Shuttle fan/separator

Experiment Description

No. 1 Dual-Membrane Gas/Liquid Phase Separator

BOEING

Objectives

- Evaluate ability to completely separate gas and liquid.
- Evaluate separation over a range of free-gas conditions.
- Eliminate the effects of gravity.

Experiment Design

No. 1 Dual-Membrane Gas/Liquid Phase Separator

BOEING

Experiment Description

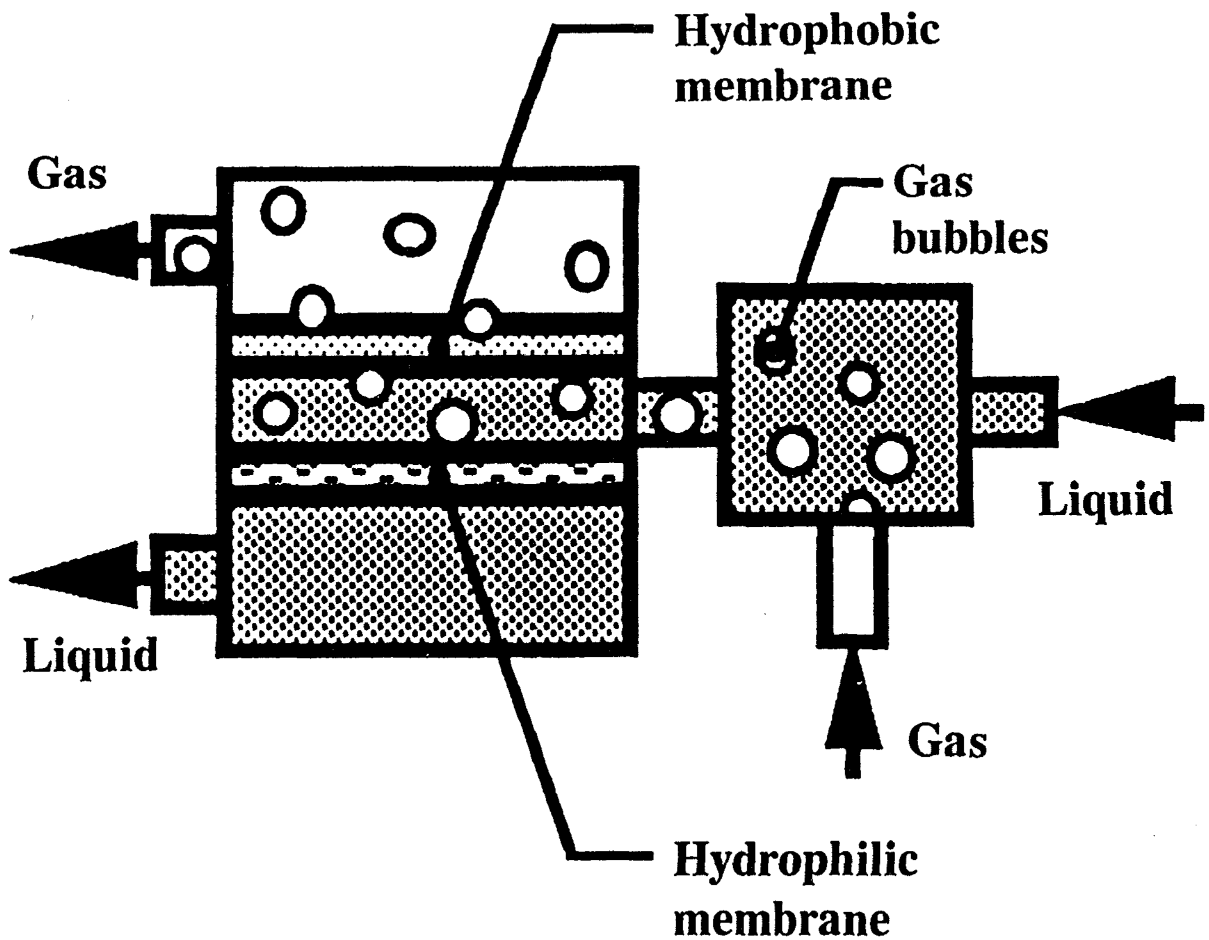
- Three-chamber test cell with two membranes
  - Hydrophilic for water passage
  - Hydrophobic for gas passage
- Fixed liquid flow with varying gas flow - mixed
- Video recording of tubing and test cell chambers
- Record of --
  - Flow rates (fluid and gas)
  - Separation effectiveness (visual)
  - Inter-chamber gas bubble behavior (visual)
  - Time, pressure and temperature
  - Shuttle acceleration environment

Experiment Description

No. 1 Dual-Membrane Gas/Liquid Phase Separator

BOEING

Test Configuration



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Experiment Description

No. 1 Dual-Membrane Gas/Liquid Phase Separator

BOEING

Parameters To Be Tested

- Complete separation of gas from gas/liquid stream
- Performance envelope for dual-membrane separator
  - Gas loading
  - Liquid flow rate
  - Pressure

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Experiment Description

No. 1 Dual-Membrane Gas/Liquid Phase Separator

BOEING

Microgravity Testing Requirement

- Performance depends on gas-to-membrane contact.
- Gravity strongly influences contact based on orientation.
- There is an unknown attraction of hydrophilic membrane for bubbles.
- Time periods greater than 50 sec are required.

Experiment Description

No. 1 Dual-Membrane Gas/Liquid Phase Separator

BOEING

Benefits

- Definition of operating parameters
  - Pressure
  - Flow rate
  - Gas loading
- Improvements in microgravity phase separation
  - Reduced complexity, mass, volume, and power
  - Increased reliability
- Applications
  - Humidity condensate removal
  - Urine collection
  - Hand wash and shower water recovery
  - Fluid (liquid) system degassing

Experiment Design

No. 2 Membrane Diffusion Interference by Gas Bubbles

BOEING

Problem Statement

- **Entrained gas bubbles potentially adhere to hydrophilic membranes in microgravity.**
- **Adhered gas bubbles reduce effective transfer surfaces for material diffusion.**

Experiment Design

No. 2 Membrane Diffusion Interference by Gas Bubbles

Objectives

- Determine to what degree entrained gas bubbles adhere to hydrophilic membranes.
- Determine the interference of adhered gas bubbles to diffusion.

Experiment Design

No. 2 Membrane Diffusion Interference by Gas Bubbles

BOEING

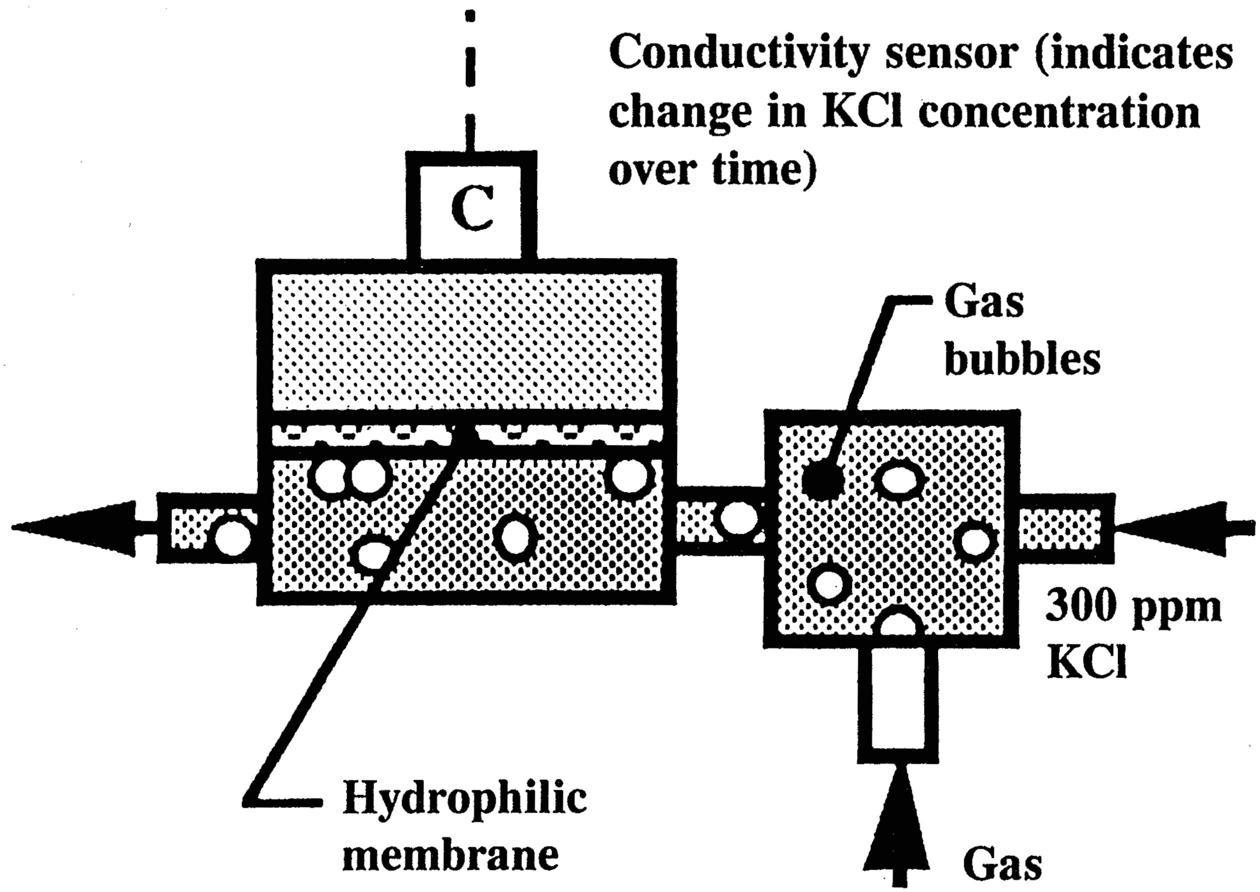
Experiment Description

- Two test cells used—control and induced-gas entrainment.
- Each cell is composed of two compartments separated by a hydrophilic membrane.
- Each cell contains test fluid, which is pumped through one compartment (feed), and deionized water, which is stagnant in the other compartment (permeate).
- A variable gas flow rate is added to the feed of the induced-gas test cell.
- The adhesion of entrained gas bubbles on the membrane surface is video-recorded for later analysis.
- The difference in diffusion between the two cells is demonstrated by the difference in the rate of change in measured conductivity of the permeates of both test cells.

Experiment Description

No. 2 Membrane Diffusion Interference by Gas Bubbles

Test Configuration



Experiment Design

No. 2 Membrane Diffusion Interference by Gas Bubbles

BOEING

Parameters To Be Tested

- The adhesion of entrained gas bubbles to a hydrophilic membrane surface
- The interference of adhered gas bubbles to the material diffusion through membranes

Microgravity Testing Requirements

- Buoyancy of gas bubbles in 1g dominates bubble behavior in liquid.
- KC-135 cannot provide stable low gravity for the required 20 min.

Experiment Design

No. 2 Membrane Diffusion Interference by Gas Bubbles

BOEING

Benefits

- Design effective plant nutrient delivery systems.
- Provide information to predict gas-bubble adhesion on hydrophilic surfaces such as metal pipes and tubes.
- Provide information to determine whether gas bubbles adhere to the hydrophilic membrane of the phase separator under low-flow conditions.

Experiment Design

No. 3 Membrane Wetting Experiment

BOEING

Problem Statement

- Certain membranes are sensitive to wetting (conditioning) for proper operation.
- Preconditioning membranes –
  - Add weight.
  - Create waste water for flushing.
  - Require special packaging.
- Wetting dried membranes in microgravity may not be feasible depending on fluid behavior.

Experiment Design

No. 3 Membrane Wetting Experiment

BOEING

Objective

- Investigate fluid behavior on a dried membrane surface as the fluid permeates the membrane.

Experiment Design

No. 3 Membrane Wetting Experiment

Experiment Description

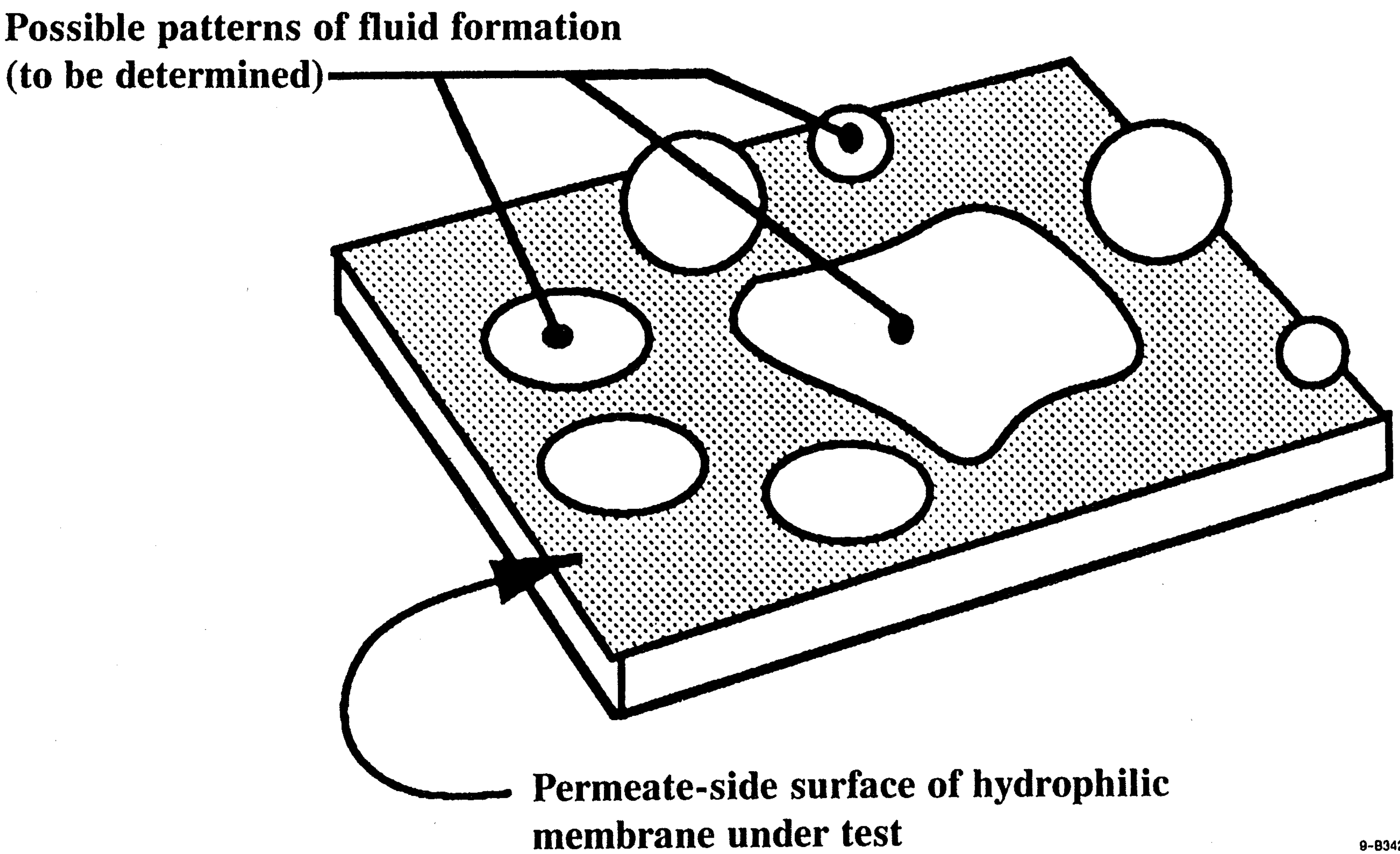
- Two-chamber test cell is separated by a hydrophilic membrane.
- Liquid flows through one chamber and permeates the membrane.
- Droplet or film formation on the permeate side of the membrane surface is recorded on video.

Experiment Description

No. 3 Membrane Fluid Wetting Behavior

BOEING

Test Configuration



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Experiment Design
No. 3 Membrane Wetting Experiment

BOEING

Parameters To Be Tested

- Fluid behavior on permeate side of membrane surface is observed.

Experiment Design
No. 3 Membrane Wetting Experiment

BOEING

Microgravity Testing Requirements

- Gravity dominates fluid behavior in 1g.
- Surface tension forces dominate in microgravity.
- Testing requires 20 min of stable microgravity.

Experiment Design

No. 3 Membrane Wetting Experiment

BOEING

Benefits

- Visual data is obtained to determine whether membranes can be conditioned in microgravity.
- Droplet formation data on membrane surfaces can be applied to condensate recovery on cold surfaces.

Summary and Conclusions

- Phase separation is an important issue for microgravity life support systems
  - Improvements could be made over the existing rotary separation technology
  - Membranes over a compact, passive and highly efficient means for gas/liquid separation
  - Membrane separation in microgravity is highly dependent upon surface tension forces and therefore requires testing in microgravity where these forces predominate.

Summary and Conclusions

BOEING

- Many life support processes depend upon transport (heat or material) across boundaries, such as for heat exchange, filtration, sensing, and water purification.
  - Membrane technology can be applied especially well for filtration, sensing and purification
  - Laboratory testing has shown that bubble adhesion on a membrane surface impedes the rate of transport across the membrane
  - The predominance of surface forces in microgravity requires testing for the susceptibility of membranes to bubble adhesion and the affects of that adhesion on transport

Summary and Conclusions

BOEING

- Some membrane applications (especially for water purification) require the membrane to be "wetted"
  - Wetting replacement membranes on-orbit as opposed to shipping them pre-wetted can result in weight, and labor savings
  - Information on how a wetting fluid forms across a membrane surface is needed to give an indication if dry membranes can be "wetted" after replacement
  - The predominance of surface forces in microgravity requires testing for membrane wetting in a microgravity environment

NASA/DOD FLIGHT EXPERIMENTS TECHNICAL INTERCHANGE MEETING

ELECTROLYSIS PERFORMANCE IMPROVEMENT AND VALIDATION EXPERIMENT

Life Systems, Inc.

Franz H. Schubert
Principal Investigator

Gene M. Lee, Ph.D.
Systems Engineer

NASA/JSC

Robert Cusick
Technical Monitor

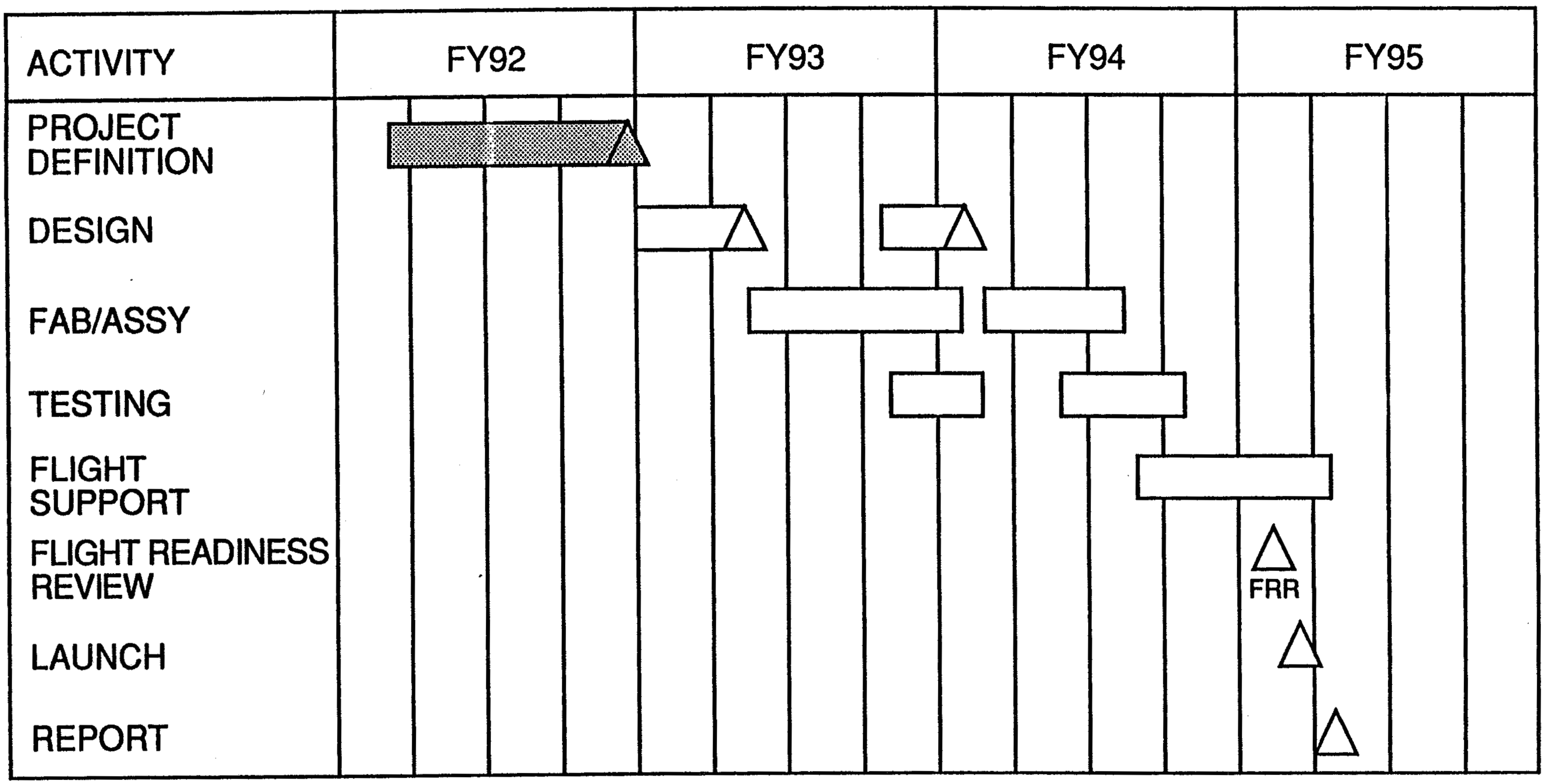
October 5-9, 1992
Monterey, California

OVERALL SUMMARY

- Phase B (Project Definition) Study has shown:
  - Experiment objectives are achievable
  - Safety requirements are satisfied, no "showstoppers" identified
  - Phase C/D schedule satisfactory for November 1994 launch

- Ready to begin Phase C/D

EPICS EXPERIMENT PROGRAM SCHEDULE



Life Systems
CLEVELAND, OHIO

4517

EPICS SCHEDULE

- Authorization to Proceed with Phase C/D Anticipated for November 1992 Start
- Unit Available for Launch in Late CY 1994 or Earlier

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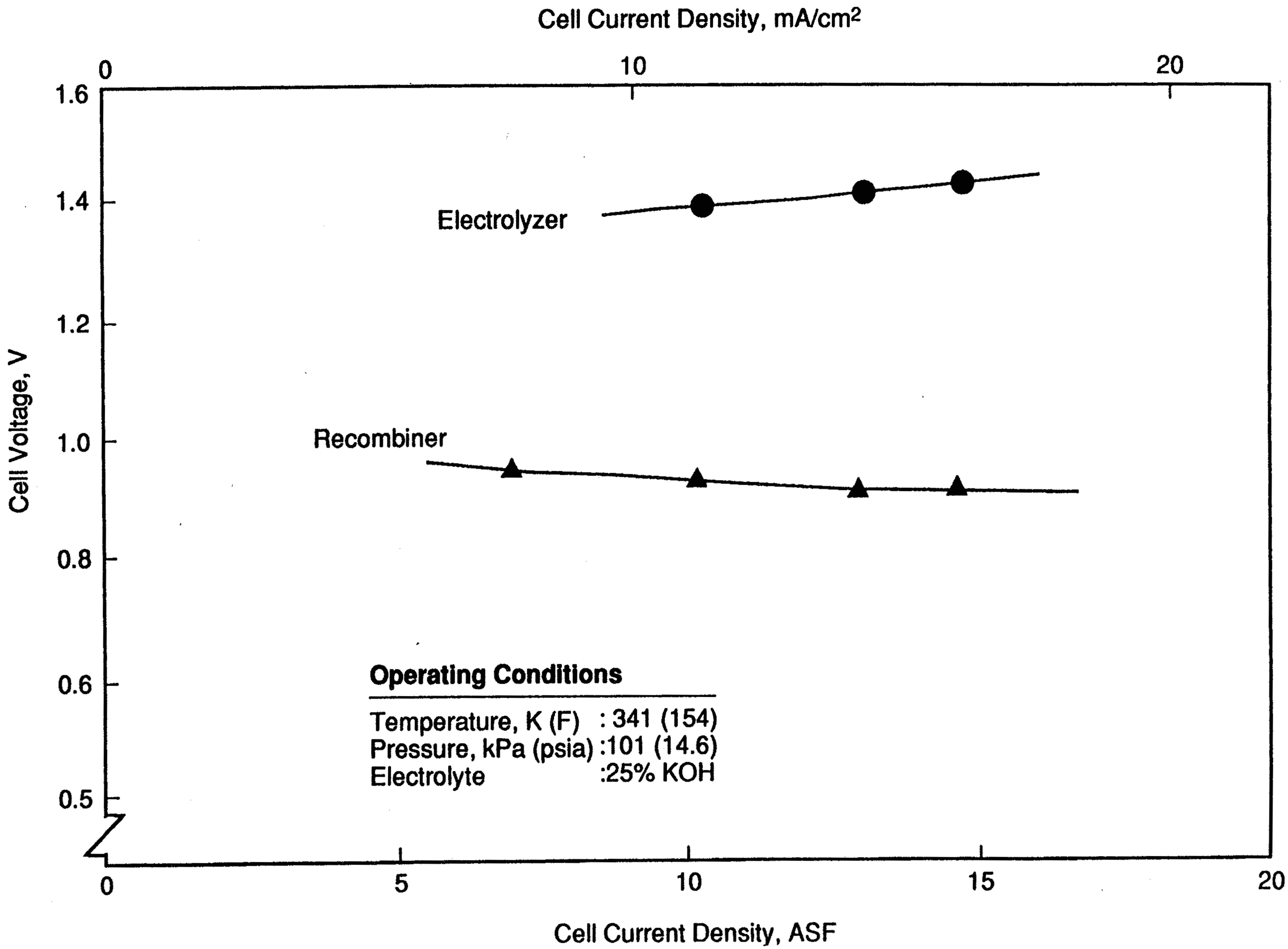
ELECTROLYZER/RECOMBINER CONCEPT TEST



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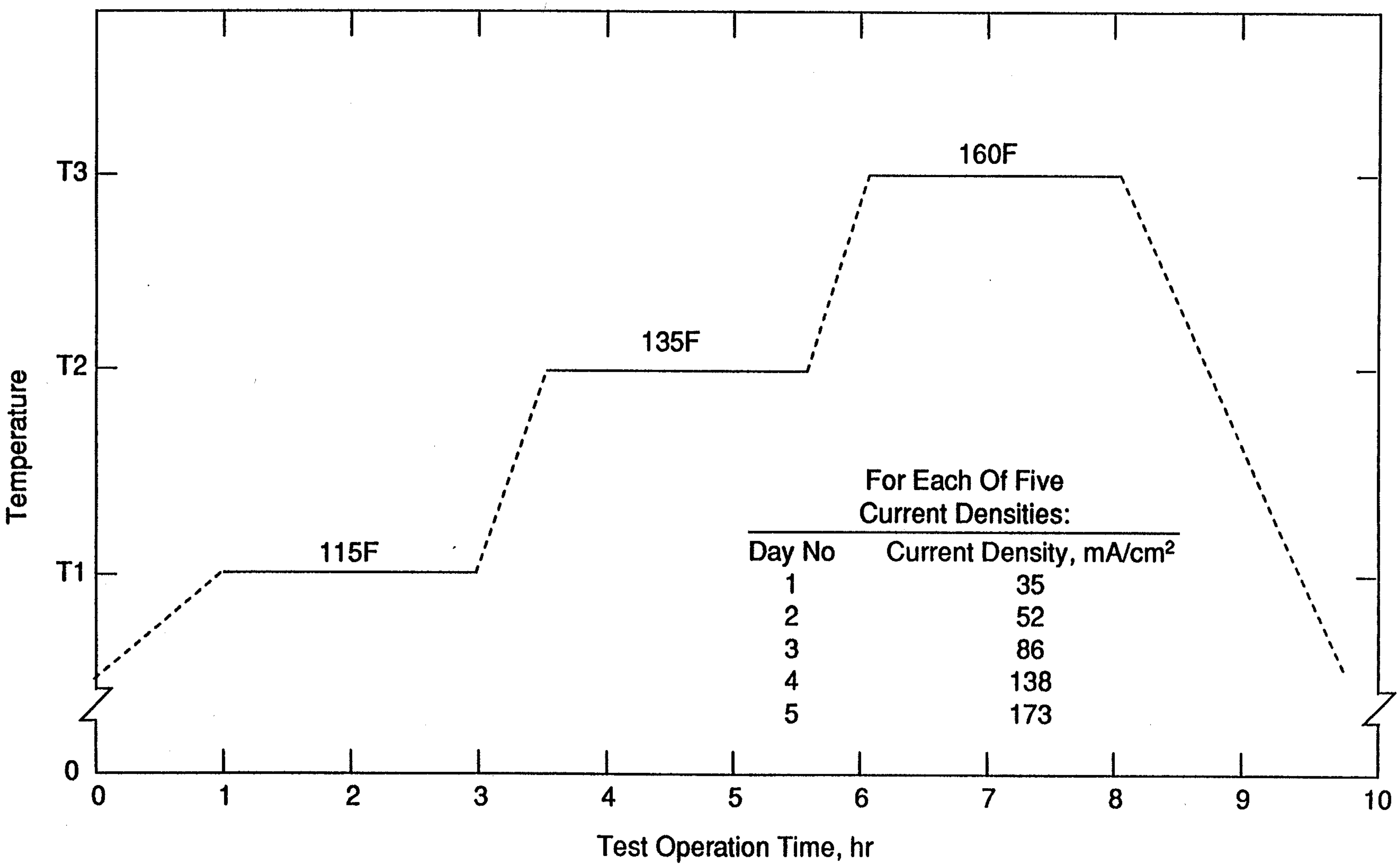
EPICS EXPERIMENT TEST SEQUENCE



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TEST SEQUENCE

Each Electrolysis Unit is Tested at a Combination of
Three Temperatures and Five Current Densities

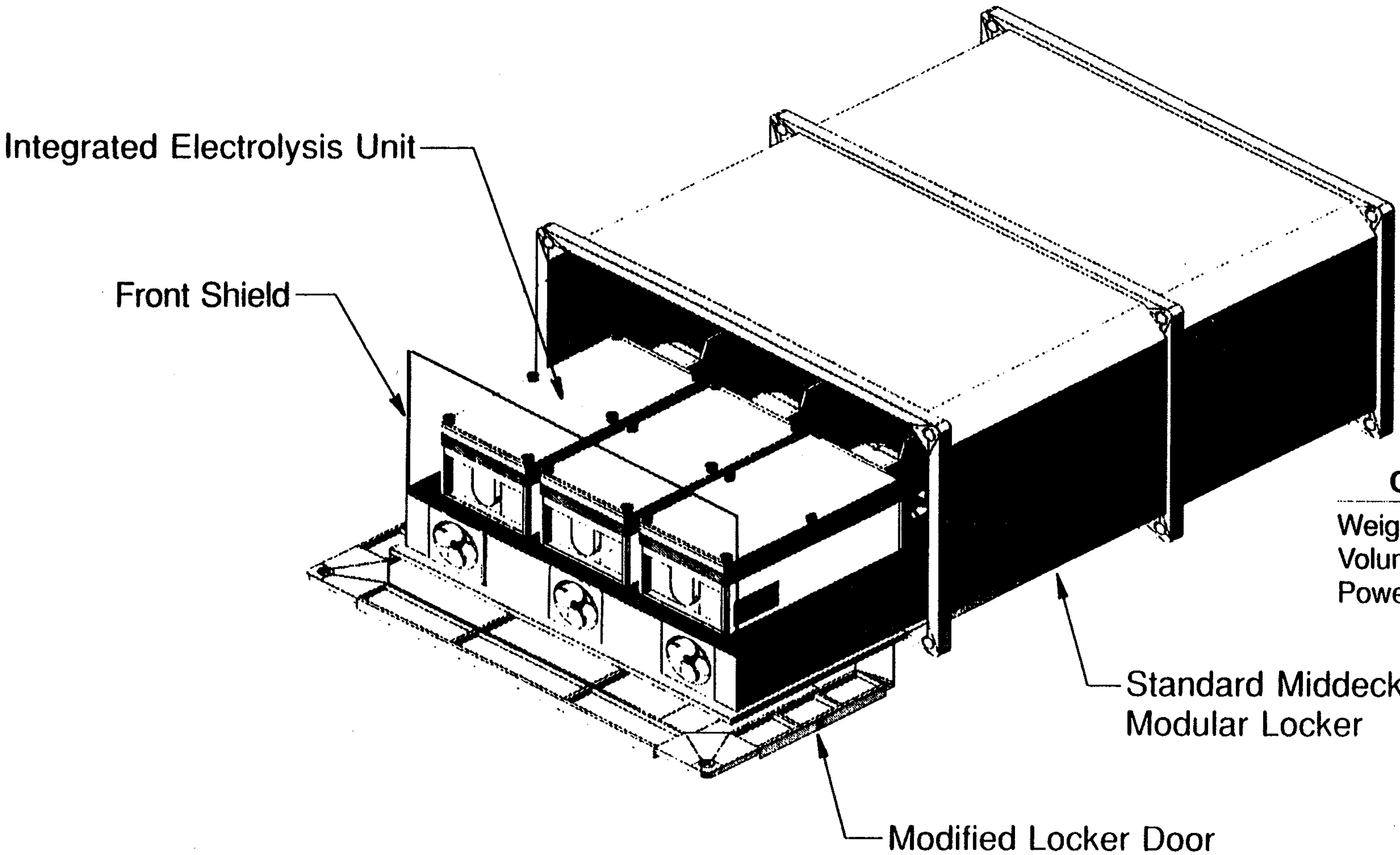
EPICS INTERFACES

Interface Requirements	Source
Water Supply	Self-contained in experiment
Coolant (air cooled)	Space Shuttle Cabin
Electrical Power	Space Shuttle
Data Acquisition/Storage	Self-contained in experiment
Crew Involvement	Single activation of experiment by operator.
Tools	No tools required

EPICS OPERATING CONDITIONS

Vehicle Conditions	
Middeck Total Pressure, kPa (psia)	101.3 ± 1.4 (14.7 ± 0.2)
Middeck Temperature, K (F)	292 to 300 (65 to 80)
Nominal Operating Conditions	
Number of Units	3
Current Density mA/cm^2 (ASF)	34 to 171 (32 to 160)
Operating Pressure, kPa (psia)	108.3 ± 1.4 (15.7 ± 0.2)
Operating Temperatures, Nominal, K (F)	319, 331 and 344 (115,135 and 160)

EPICS[(a)] EXPERIMENT PACKAGING CONCEPT (IN LOCKER)



Characteristics

Weight, kg (lb)	: 20 (45)
Volume, dm^3 (ft^3)	: 45 (1.6)
Power, W	: 111 Max.

(a) Electrolysis Performance Improvement Concept Studies

EXPERIMENT HARDWARE

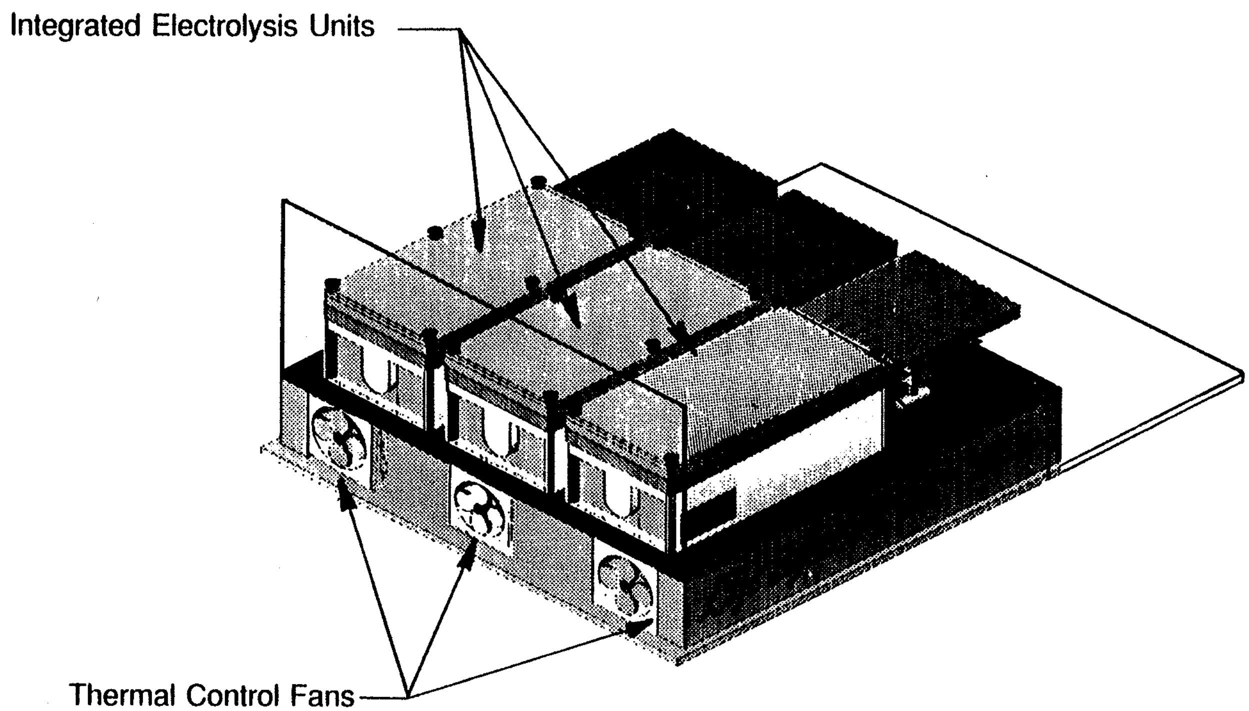
- Packaged in One Shuttle Orbiter Middeck Locker
- Simple Operational Requirements
- Minimum Interfaces

INTEGRATED ELECTROLYSIS UNIT (IEU) CELL CONFIGURATIONS

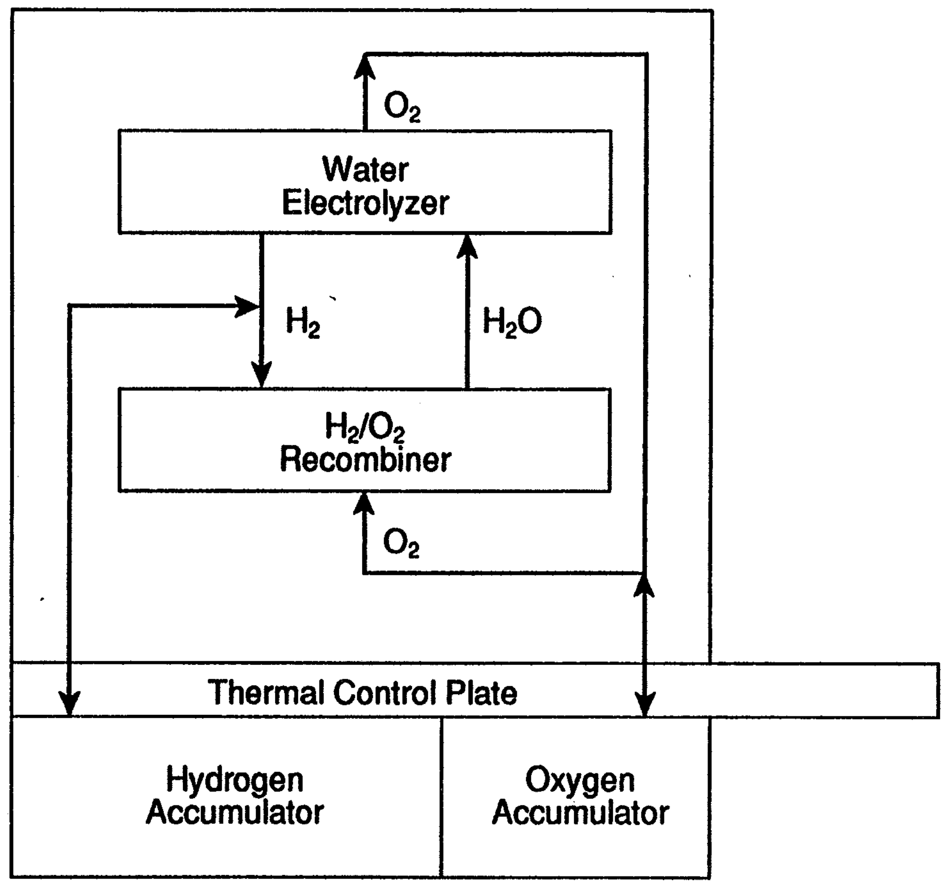
	IEU No.		
Parameter	**1**	**2**	**3**
Matrix Thickness[a]	Baseline	0.010 inch	Baseline
Electrode Thickness[b]	Baseline	Baseline	Baseline
Pore Size[c]	Baseline	Baseline	15 to 20 micron[e]
Porosity[d]	Baseline	Baseline	85-90[e]

(a) Baseline matrix thickness is 0.015 inch
(b) Baseline electrode thickness is 0.030 ±0.002 inch
(c) Baseline pore size is 10 to 12 micron
(d) Baseline porosity is 80 to 85%
(e) Actual pore size to be determined by manufacturer's capability

EPICS MECHANICAL/ELECTROCHEMICAL ASSEMBLY



FUNCTIONAL SCHEMATIC OF IEU



EPICS EXPERIMENT APPROACH

- Safety is Key
  - Use fuel cell based recombiner to consume H_2 and O_2 immediately after generation
- Enhance Experiment Success
  - Design EPICS with Three Independent Integrated Electrolysis Units (IEU) For Redundancy
  - Build and test engineering model before finalizing flight unit
- Increase Technology Base in Key Microgravity Impacted Areas
  - Include multiple combinations of electrode/separator (matrix) configurations

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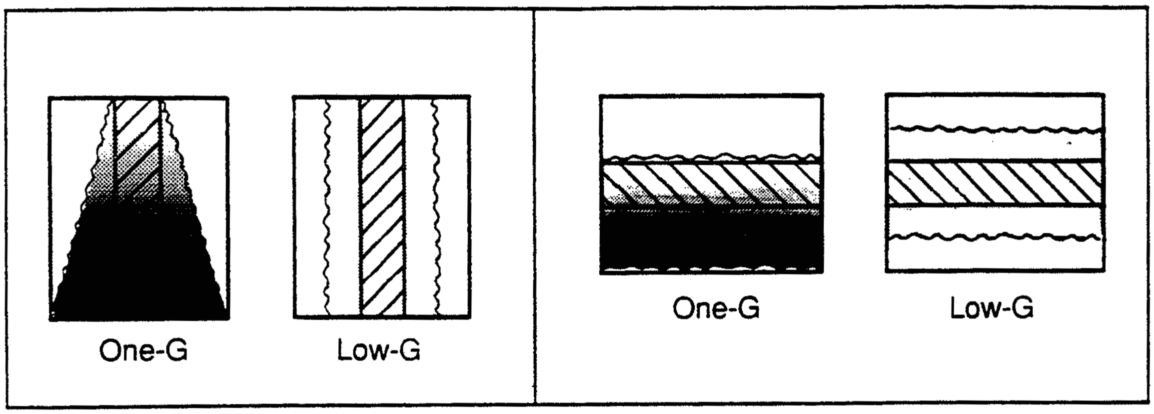
EPICS EXPERIMENT DESCRIPTION

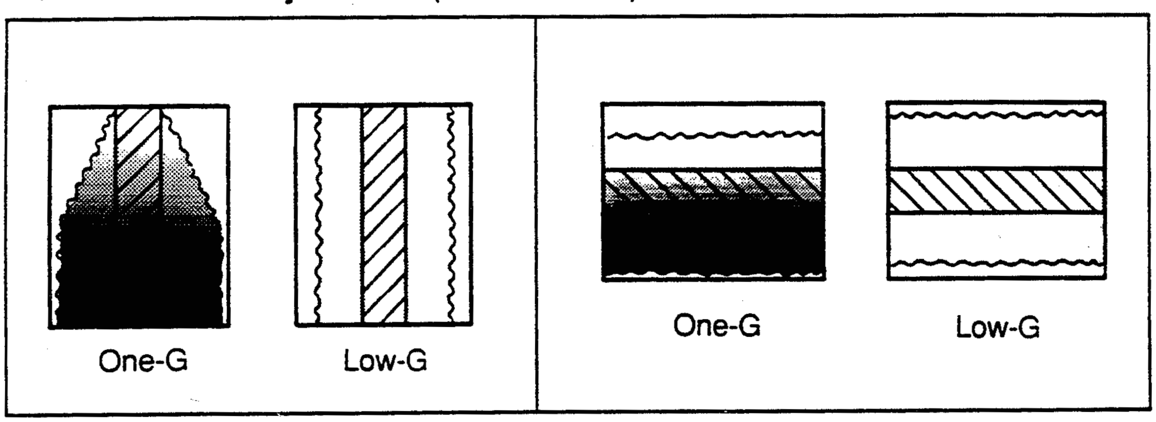
- Approach
- Hardware
- Test Sequence
- Schedule

MAGNIFICATION OF GRAVITY EFFECTS ON ELECTROLYTE DENSITY DISTRIBUTION

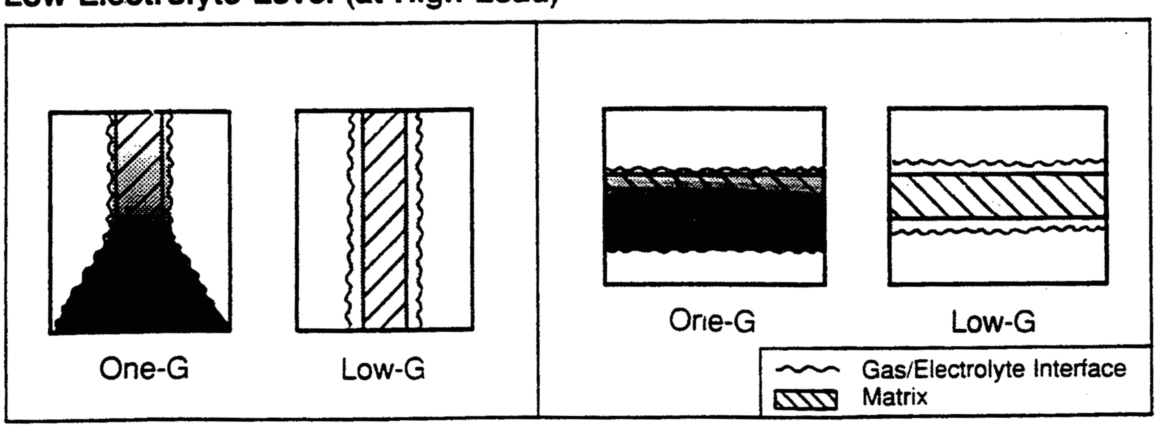
Nominal Electrolyte Level (at Nominal Load)



Increased Electrolyte Level (at Low Load)



Low Electrolyte Level (at High Load)



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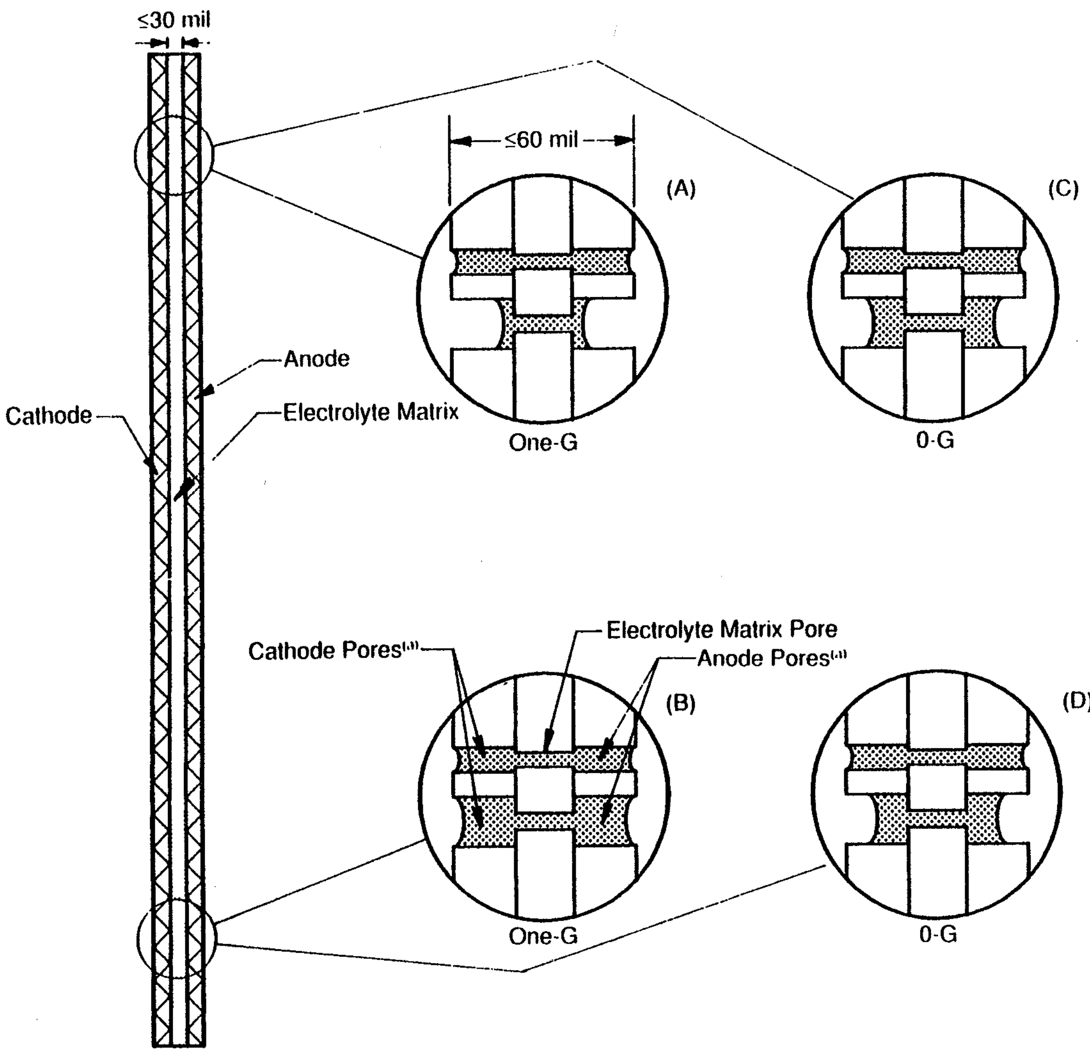
MAGNIFICATION OF GRAVITY EFFECTS ON GAS-LIQUID INTERFACE



(a) All pores will not be the same size.

NOTE:

1. Capillary effects in one-G are not strong enough to completely counteract gravity effects.
2. The smaller pores will have stronger capillary force, which may be able to completely counteract gravity.
3. There is a gravity force gradient, from top to bottom in one-G. Resulting in a liquid distribution gradient.
4. In 0-G the gravity gradient is eliminated and therefore the liquid will be distributed evenly.
5. The gravity force gradient will also result in a density difference.

EPICS EXPERIMENT OBJECTIVES

- Demonstrate and validate the Static Feed Electrolyzer (SFE) concept in microgravity
- Investigate ways a microgravity environment may improve SFE process efficiency since "absence" of gravity will result in:
  - A more uniform electrolyte volume distribution governed primarily by capillary forces and surface tension
  - A more uniform electrolyte density distribution

The Electrolysis Performance Improvement Concepts Study (EPICS) Has Two Objectives

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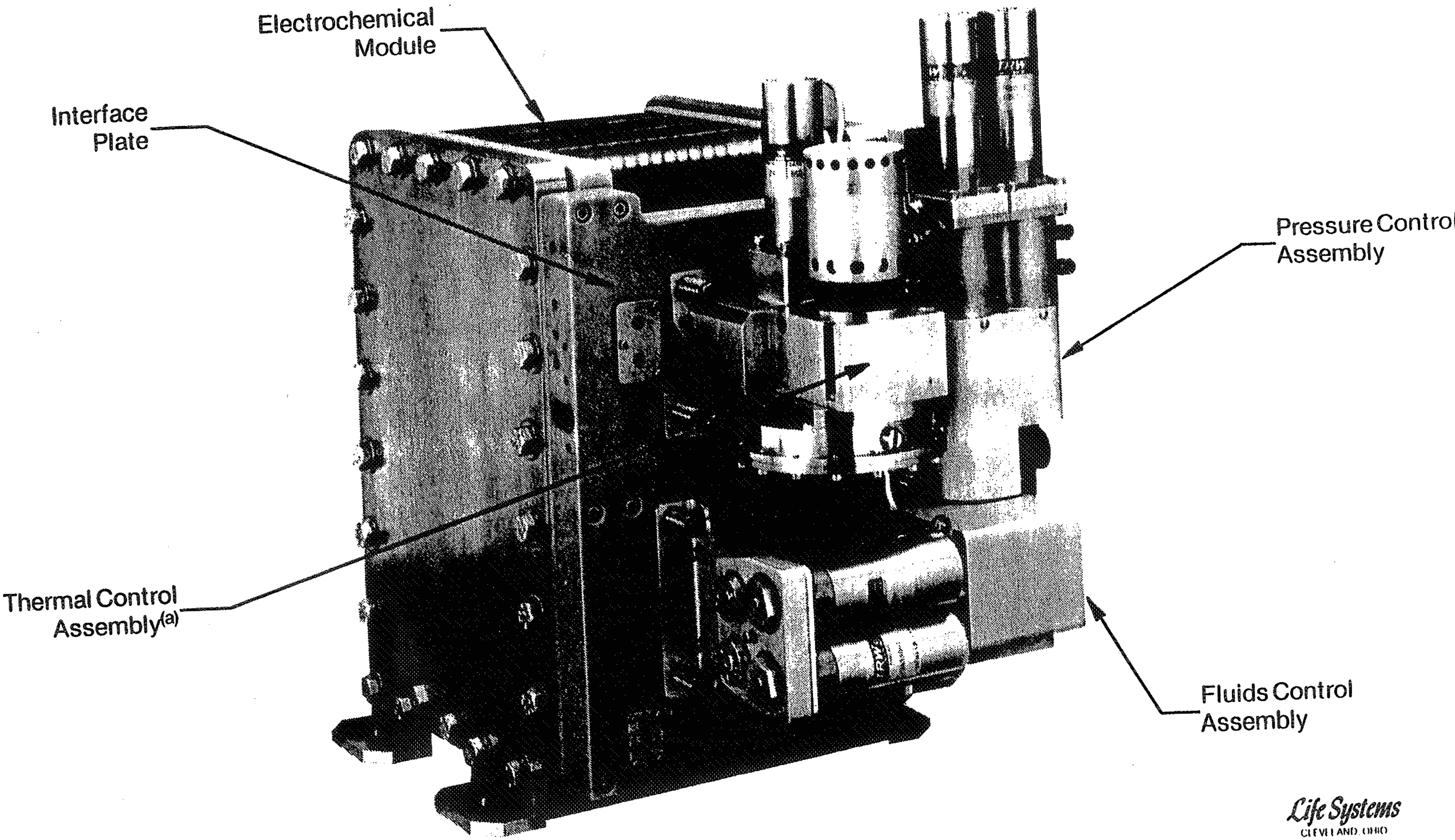
SFE-1 ILLUSTRATION OF SFE SUBSYSTEM



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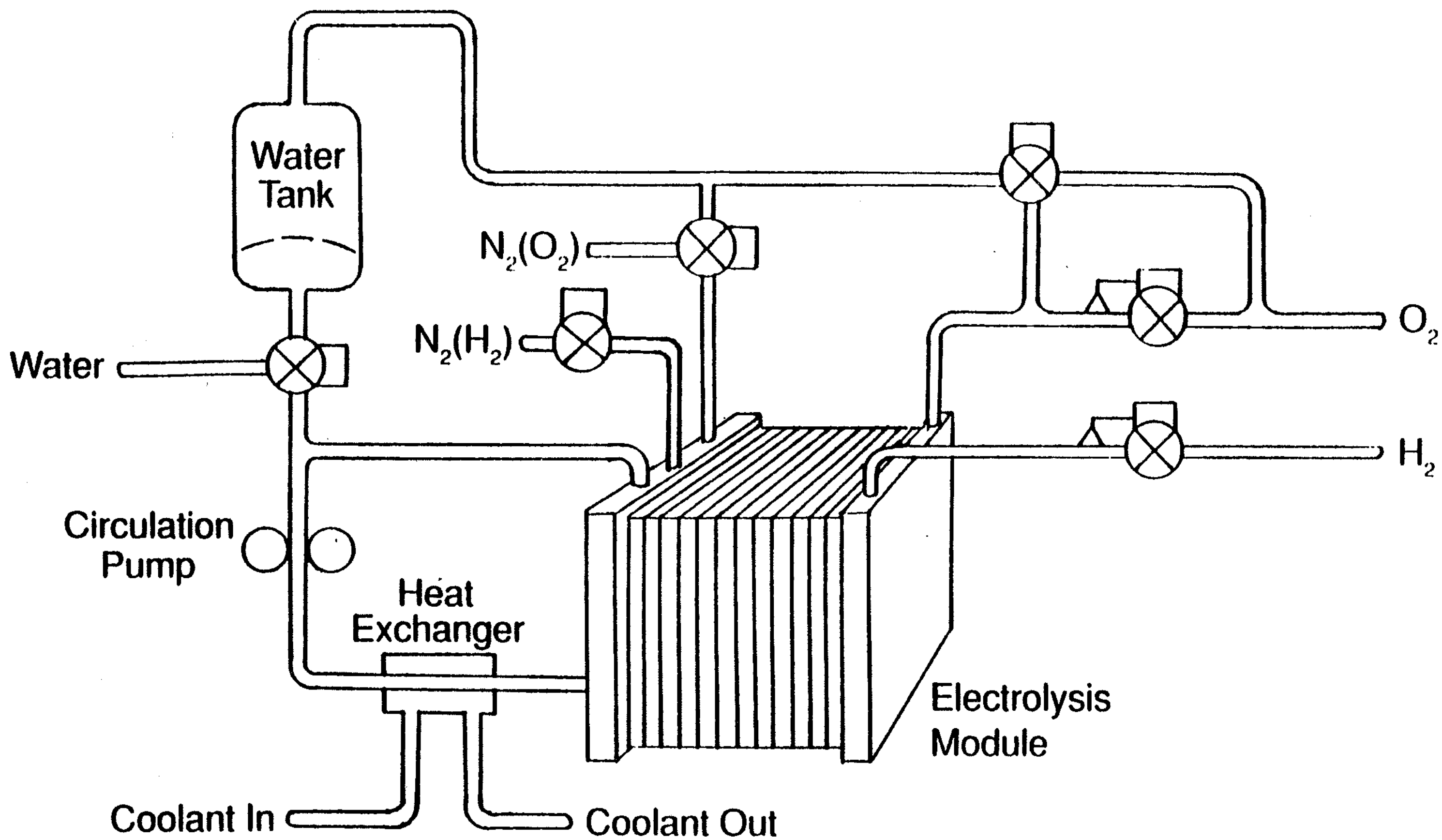
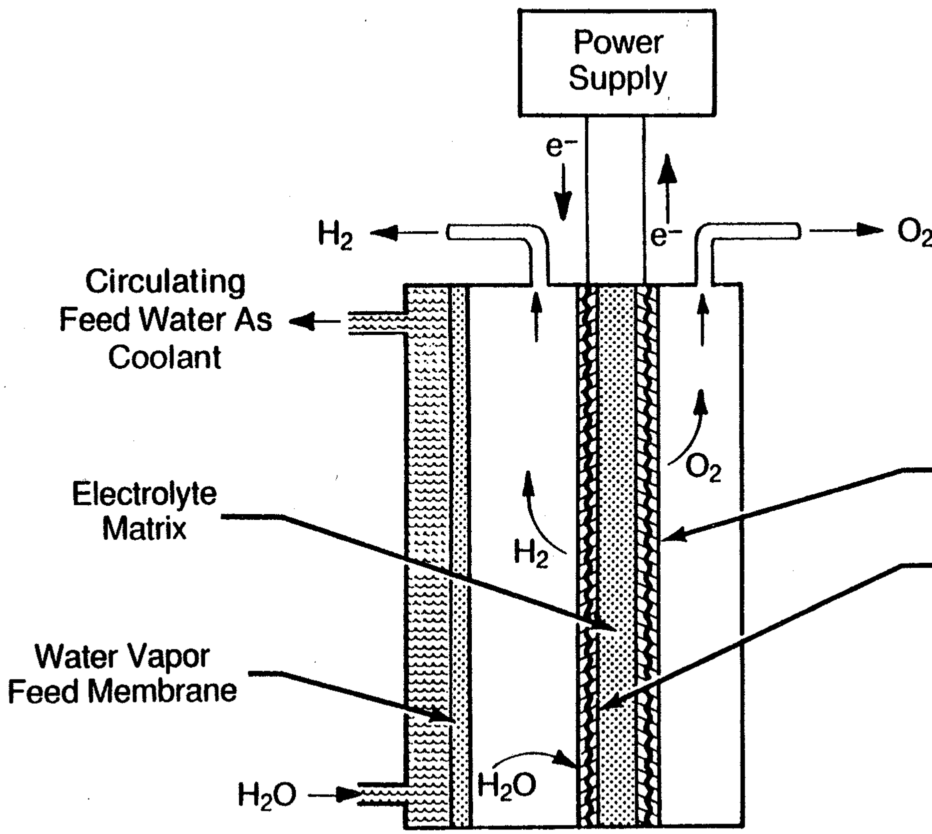
(a) Not required for SFE-IV

3150

TYPICAL SFE PROCESS SCHEMATIC



ELECTROLYZER CELL SCHEMATIC AND REACTIONS



Reactions

Anode (+): $4OH^- \rightarrow O_2 + 2H_2O + 4e^-$

Cathode (-): $4H_2O + 4e^- \rightarrow 2H_2 + 4OH^-$

Overall: $2H_2O + \text{Power} \rightarrow 2H_2 + O_2$

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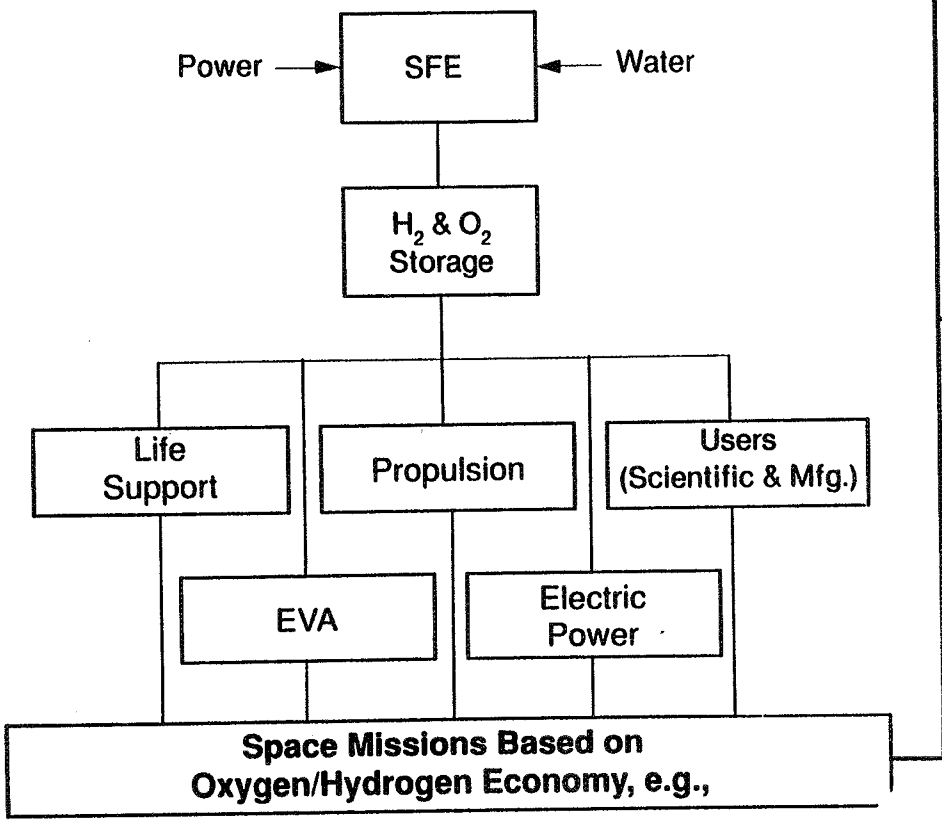
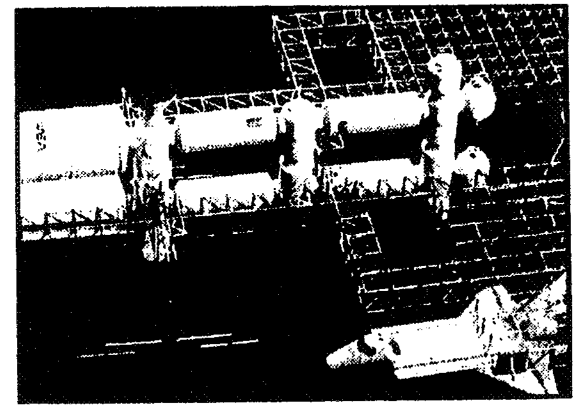
The Static Feed Electrolysis (SFE) Concept Was Developed For Space Application

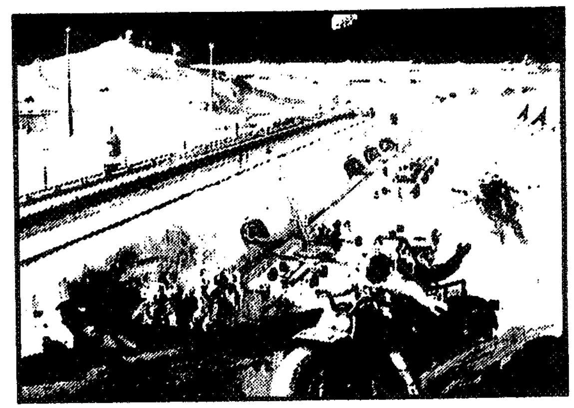
SFE APPLICATIONS MEET NASA MISSION NEEDS/GOALS



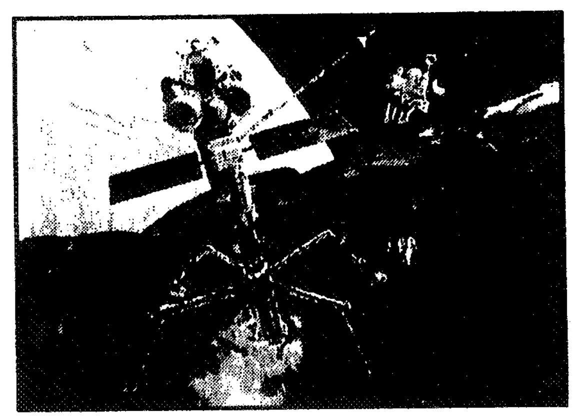
Space Station Freedom
- Propulsion
- Life Support
- EVA



Lunar Base
- Propellant Production
- Life Support
- Energy Storage
- EVA



Phobos Propellant Plant
- Propellant Production
- Life Support
- Energy Storage
- EVA



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Water Electrolysis Will Play An Ever Increasing Role In Space Missions

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PRESENTATION OVERVIEW

- Water Electrolysis: An ever increasing need/role for space missions
- Static Feed Electrolysis (SFE) Technology: A concept developed for space applications
- Experiment Objectives: Why test in microgravity environment
- Experiment Description: Approach, hardware description, test sequence and schedule
- Summary: Successfully completed Phase B, ready for Phase C/D

N93-28738

RISK-BASED SPACECRAFT FIRE SAFETY EXPERIMENTS

S39-18
159244
P-11

G. Apostolakis, I. Catton, F. Issacci,
T. Paulos, S. Jones, K. Paxton and M. Paul

Mechanical, Aerospace and Nuclear Engineering Department
University of California, Los Angeles

Flight Experiments Technical Interchange Meeting
Monterey, California, October 5-9, 1992

Sponsored by NASA Lewis Research Center

PROBABILISTIC RISK ASSESSMENT APPLIED TO FIRE SAFETY

Spacecraft fire risk can never be reduced to a zero probability.

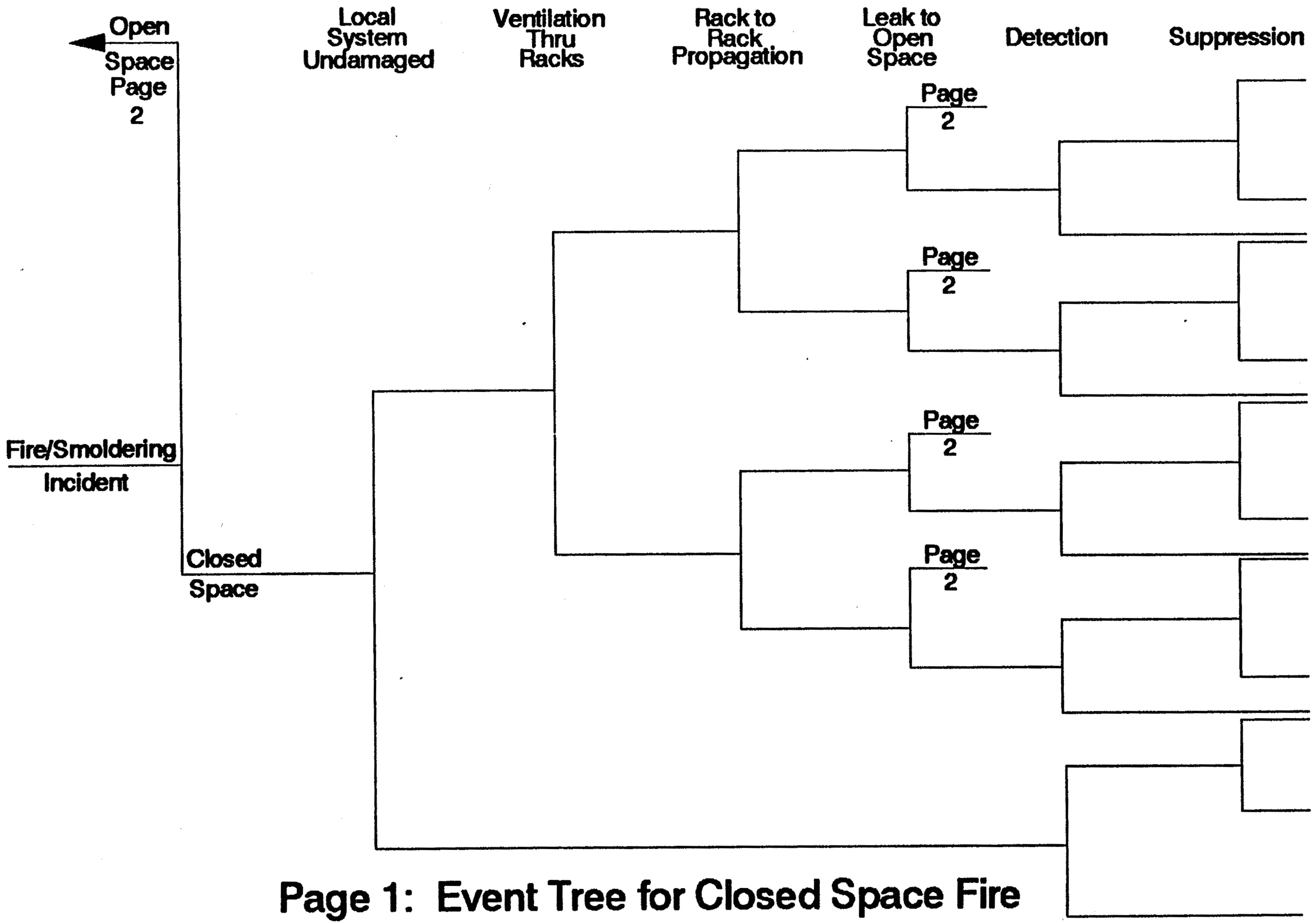
Probabilistic risk assessment is a tool to reduce risk to an acceptable level.

MAJOR STEPS:

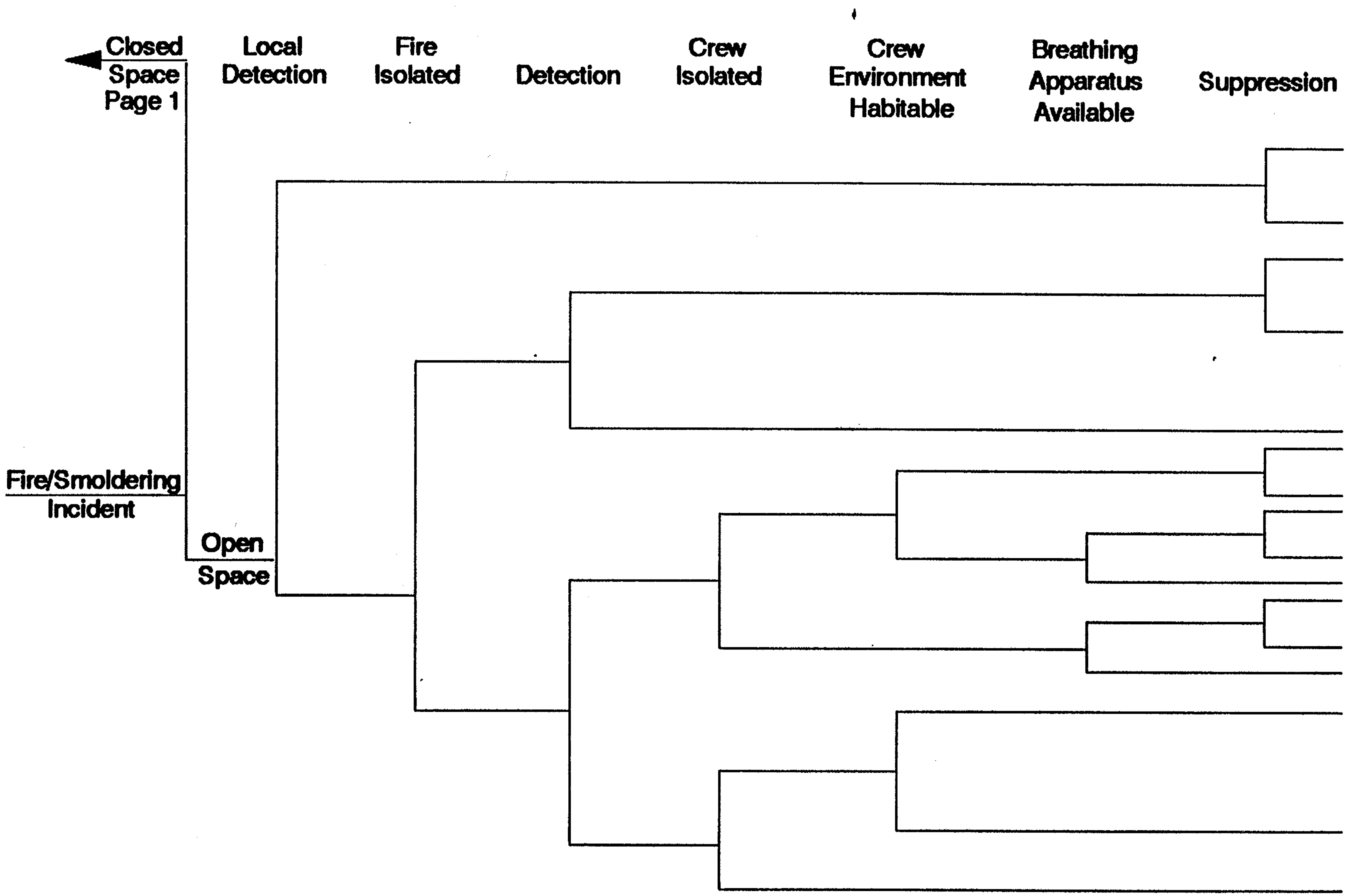
1. Identification of "critical" locations and the assessment of the frequency of fires: overheating, spills, smoldering, ignition, etc.

2. Estimation of the fraction of fires that can lead to damage of specified components: fire growth time and the competing detection and suppression times

3. Estimation of the fraction of fires that can lead to mission damage



Page 1: Event Tree for Closed Space Fire



Page 2: Event Tree for Open Space Fire

$$\lambda_{loss} = \sum \lambda_j \, Q_{d/j,k} \, Q_{loss\ d/j,k}$$

λ_{loss}	frequency lost
λ_j	frequency of class j fires
$Q_{d/j,k}$	fraction of class j fires that lead to damage of the k^{th} critical system
$Q_{loss\ d/j,k}$	fraction of class j fires leading to damage of the k^{th} system that cause the loss of the spacecraft

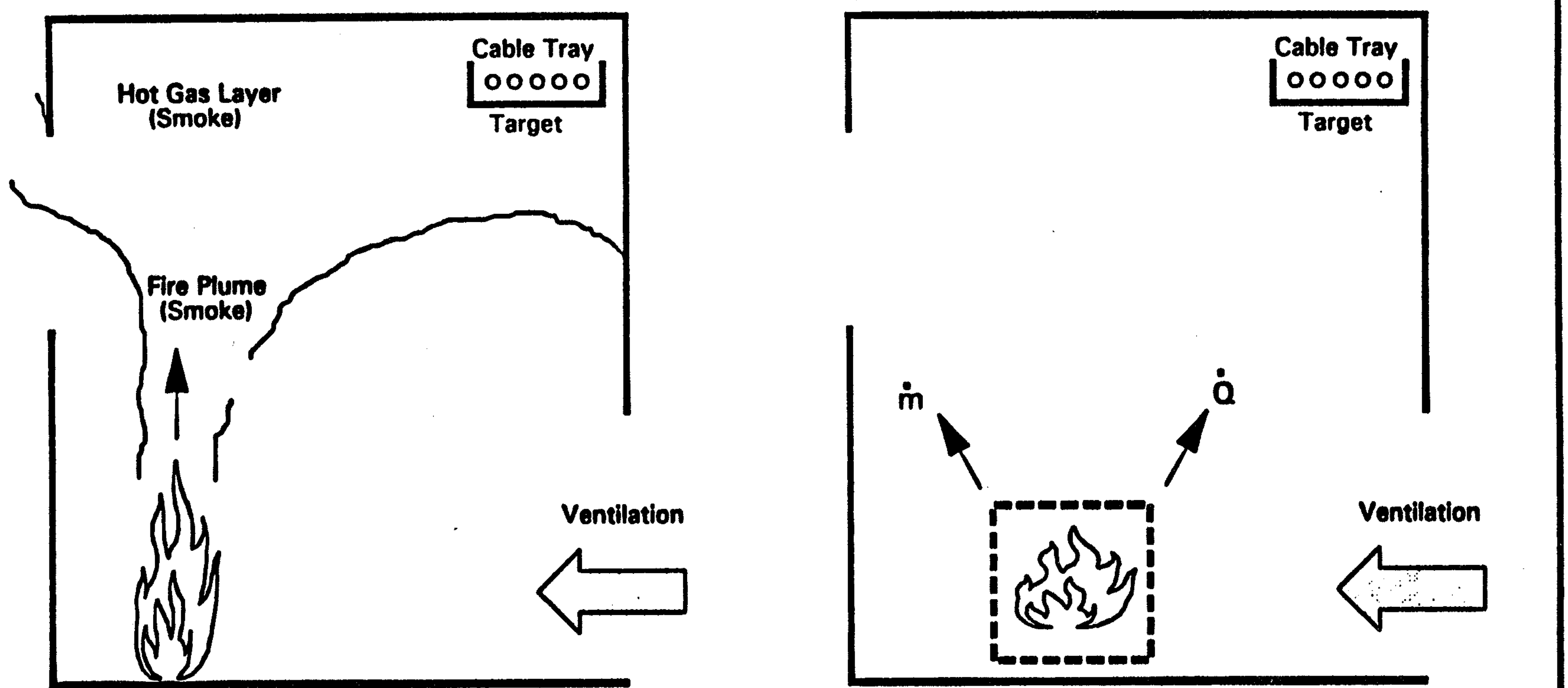
$$Q_{d/j} = Fr\,[T_G < T_H \,/\, fire]$$

T_G	growth time
T_H	hazard time

$$T_H = T_f + T_d + T_s$$

T_f	time to detection
T_d	detector response time
T_s	suppression time

Source - Transport - Deposition



Terrestrial Microgravity

UCLA μg

Fire Safety Assessment

Damage Time

Detection & Suppression Time

Target Identification

Crew
Station System

Modes Identification

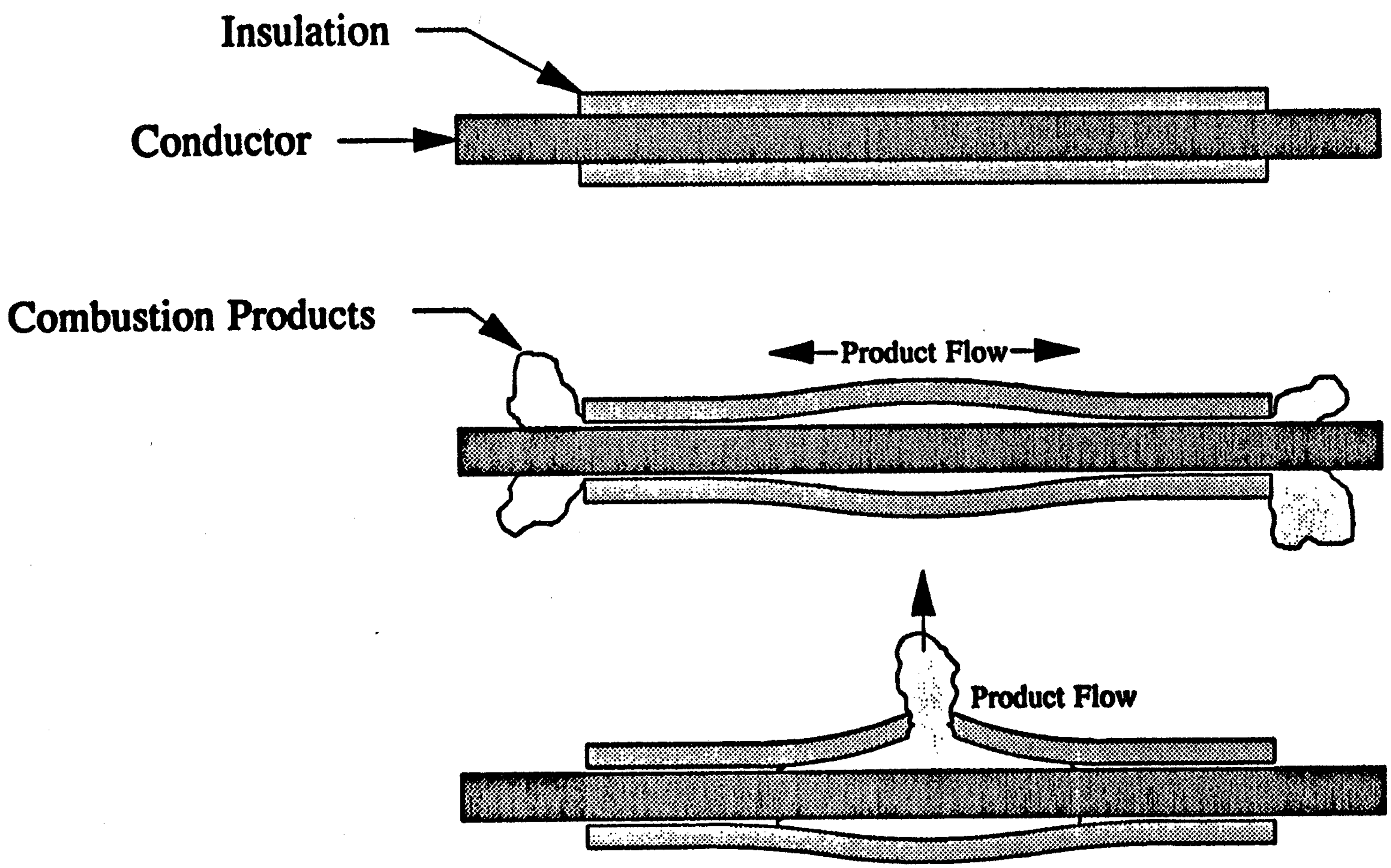
Heat
Smoke
Toxins

Event Description

Source
Transport
Deposition

Wire Overload Phenomena

UCLA μg



UCLA μg

	Damage Modes	Tests	Models
Source	Heat Release	temperature measurements	f(T)
	Smoke Release	obscuration, TEM grids	f(T)
	Toxin Release	IR/Mass spec.(White Sands), sampling	f(T)
Transport	Heat Transport	temperature measurements	fluid flow, temp., etc
	Smoke Transport	TEM grids/visualization	fluid flow, temp., etc
	Toxin Transport		fluid flow, temp., etc
Deposition	Adjacent Wire Damage	pairs, bundles	heat release, qualitative
	Particulate Deposition	TEM grids	TBD
	Corrosivity	*thin copper target plate*	*qualitative*

NASA Lewis 2.2 sec Drop Tower

UCLA μg

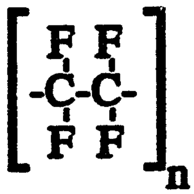
Sample Materials

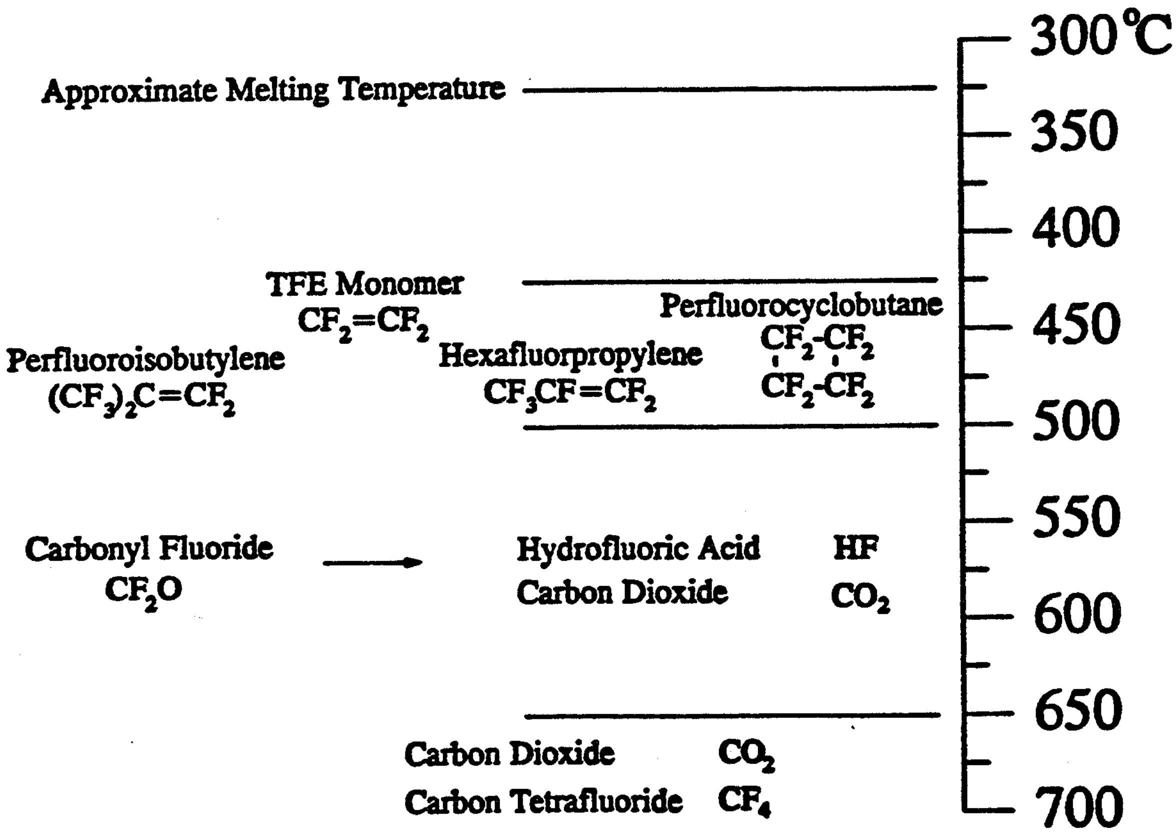
PTFE - Teflon $[-CF_2-CF_2-]$	Interior wiring	Smoke Production Toxic Production Acidic Production Combustible Production
ETFE - Tefzel $[-CF_2-CH_2-]$	Exterior wiring	
Bundles	Interior/Exterior	+ Adjacent Wire Damage
Twisted Pairs	Interior/Exterior	

UCLA
μg

PTFE

Polytetrafluorethylene
Thermal Degradation Products





ARMSTRONG LABORATORY SPACE VISUAL FUNCTION TESTER PROGRAM

LT COL MELVIN R. O'NEAL, O.D., Ph.D.

H. LEE TASK, Ph.D.

MAJ GERALD A. GLEASON, O.D., Ph.D.

Visual Display Systems Branch

Human Engineering Division

Crew Systems Directorate

Armstrong Laboratory

AL/CFHV, Wright-Patterson AFB, Ohio 45433-6573

SPACE VISION LOGO

MEL "BROOKS" O'NEAL PRESENTS:

SpaCE BALLS

(SPACE CALIBRATED EYEBALLS)

Mensura Precisa Visionis Oculi

INTRODUCTION

Many astronauts and cosmonauts have commented on apparent changes in their vision while on-orbit. Comments have included descriptions of earth features and objects that would suggest enhanced distance visual acuity. In contrast, some cosmonaut observations suggest a slight loss in their object discrimination during initial space flight. Astronauts have also mentioned a decreased near vision capability that did not recover to normal until return to earth.

DUNTLEY SPACE VISION EXPERIMENT

VISUAL ACUITY

- Hand-held device
  - -- Square wave bar gratings
  - -- High and low contrast
  - -- Tested at optical infinity (distance vision)
- Also used ground targets

RESULTS

- Gemini V and VII
  - -- No significant change in acuity

USSR SPACE VISION EXPERIMENTS

VISUAL ACUITY

- Square wave gratings
- High (94%) and low (13%) modulation contrast
- Tested at 30 cm (near vision)

RESULTS

- Voskhod
  - -- Two subjects
  - -- 5 - 10% drop in high contrast acuity
- Soyuz 4 & 5
  - -- Four subjects
  - -- Three showed ~ 10% drop in both high and low contrast acuity
  - -- One showed ~ 20% improvement in high contrast acuity
- Soyuz 9
  - -- One subject
  - -- 18% drop in high contrast visual acuity
  - -- 4% drop in low contrast visual acuity

VISUAL FUNCTION TESTERS

- Model 1 (VFT-1) : Multi-Visual Functions
- Model 2 (VFT-2) : Visual Contrast Threshold
- Model 4 (VFT-4) : Visual Near Point/Facility

EFFECT OF MICROGRAVITY ON SEVERAL VISUAL FUNCTIONS DURING STS SHUTTLE MISSIONS

VISUAL FUNCTION TESTER - MODEL 1 (VFT-1)

LT COL MELVIN R. O'NEAL, O.D., Ph.D.
H. LEE TASK, Ph.D.
COL LOUIS V. GENCO, O.D., M.S.

N93- 28740

PURPOSE (VFT-1)

- Previous visual acuity studies at different test distances and may be affected by age and lighting
- Determine effect of microgravity on distance visual acuity over mission duration
- Use high contrast acuity targets in small size increments under set lighting conditions
- Expand assessment to several other visual functions

METHODS (VFT-1)

SUBJECTS

- 26 STS Astronauts
  - -- 5 subjects with only 1 pre- and 1 on-orbit eliminated
  - -- 1 Toric-SCL with on-orbit problem eliminated
  - -- n = 20; 1 HGP CL, 1 SCL, 1 Toric-SCL included
  - -- Repeat data on 2 subjects

APPARATUS

- Visual Function Tester - Model 1 (VFT-1)
  - -- Small, hand-held, battery powered
  - -- Seven vision tests:
    - Acuity in small steps to 20/7.7
    - Stereopsis to 10 sec-of-arc
    - Lateral phoria, Vertical phoria, Cyclophoria
    - Critical flicker fusion
    - Retinal rivalry

METHODS

PROCEDURE

- Pre-mission briefing and tester familiarization
- Vision assessed
  - -- 2x pre-flight at 14 days (L-14) and 7 days (L-7)
  - -- Daily after wake-up on-orbit
  - -- 3x post flight at landing, 3 days (L+3) and 7 days (L+7)

DATA ANALYSIS

- Calculated difference between mean of two pre-flight sessions (taken as baseline) and each subsequent measurement for each subject
- Non-parametric statistical analysis (Wilcoxon signed-rank)

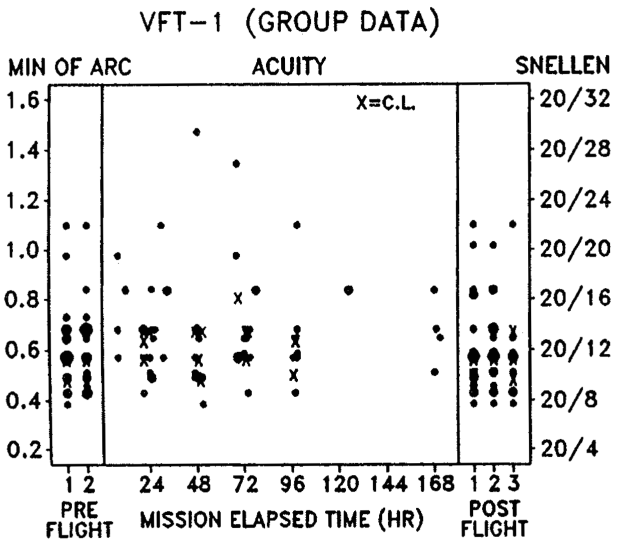
RESULTS

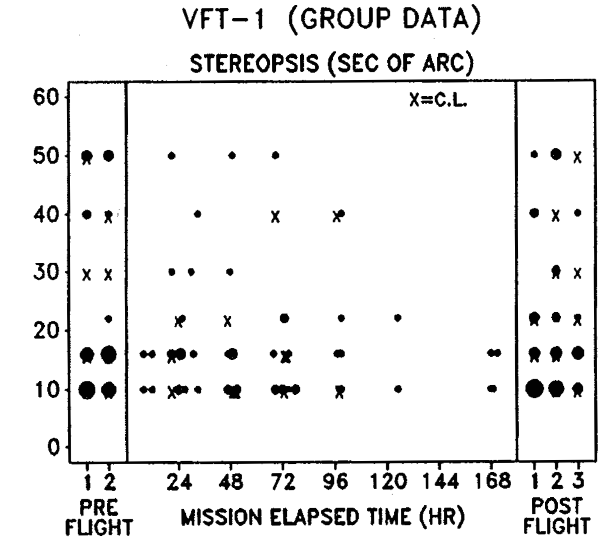
GROUP DATA

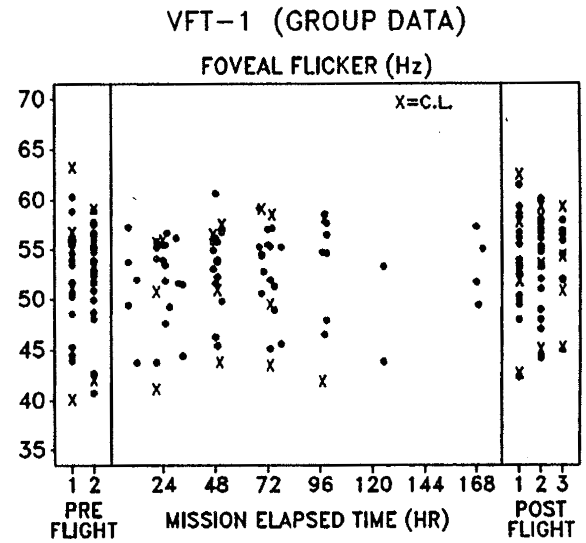
- Corresponding data days are:
  - -- L-14 days = Pre-flight 1
  - -- L-7 days = Pre-flight 2
  - -- On-orbit = Hours of mission elapsed time (MET)
  - -- Landing = Post-Flight 1
  - -- L+3 days = Post-flight 2
  - -- L+7 days = Post-flight 3

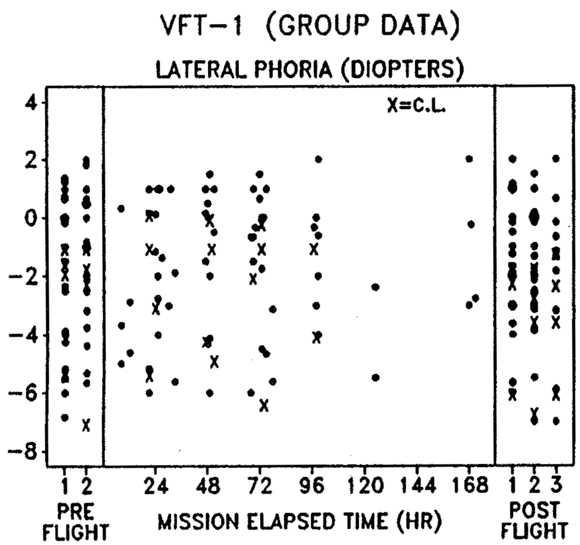
- Size of dots represent number of subjects with same performance

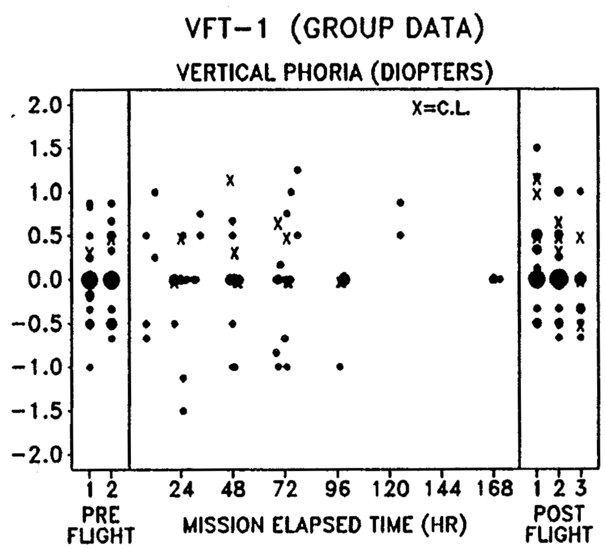
- Variability between subjects in baseline pre-flight data is typical of psychophysical vision data

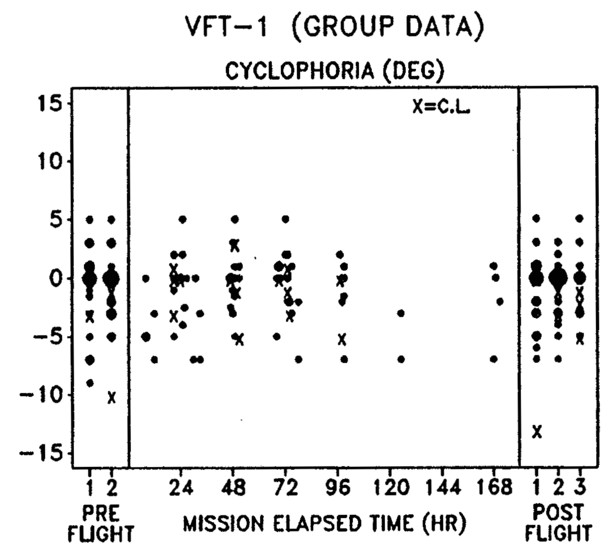

VFT-1 (GROUP DATA)
MIN OF ARC
ACUITY
SNELLEN
X=C.L.
1.6
1.4
1.2
1.0
0.8
0.6
0.4
0.2
20/32
20/28
20/24
20/20
20/16
20/12
20/8
20/4
1 2
24 48 72 96 120 144 168
1 2 3
PRE FLIGHT
MISSION ELAPSED TIME (HR)
POST FLIGHT


VFT-1 (GROUP DATA)
STEREOPSIS (SEC OF ARC)
X=C.L.
60
50
40
30
20
10
0
1 2
24 48 72 96 120 144 168
1 2 3
PRE FLIGHT
MISSION ELAPSED TIME (HR)
POST FLIGHT


VFT-1 (GROUP DATA)
FOVEAL FLICKER (Hz)
X=C.L.
70
65
60
55
50
45
40
35
1 2
24 48 72 96 120 144 168
1 2 3
PRE FLIGHT
MISSION ELAPSED TIME (HR)
POST FLIGHT


VFT-1 (GROUP DATA)
LATERAL PHORIA (DIOPTERS)
X=C.L.
4
2
0
-2
-4
-6
-8
1 2
24 48 72 96 120 144 168
1 2 3
PRE FLIGHT
MISSION ELAPSED TIME (HR)
POST FLIGHT


VFT-1 (GROUP DATA)
VERTICAL PHORIA (DIOPTERS)
X=C.L.
2.0
1.5
1.0
0.5
0.0
-0.5
-1.0
-1.5
-2.0
1 2
24 48 72 96 120 144 168
1 2 3
PRE FLIGHT
MISSION ELAPSED TIME (HR)
POST FLIGHT


VFT-1 (GROUP DATA)
CYCLOPHORIA (DEG)
X=C.L.
15
10
5
0
-5
-10
-15
1 2
24 48 72 96 120 144 168
1 2 3
PRE FLIGHT
MISSION ELAPSED TIME (HR)
POST FLIGHT

VFT-1 GROUP DATA

	MEAN PRE-FLIGHT	MEAN CHANGE
VISUAL ACUITY	0.61 min arc (20/12.2)	+0.06 min arc (to 20/13.4)
STEREOPSIS	19.8 arc sec	-4.9 arc sec
LATERAL PHORIA	-2.08 $^{\Delta}$ (ESO)	+0.36 $^{\Delta}$
VERTICAL PHORIA	0.04 $^{\Delta}$	-0.07 $^{\Delta}$
CYCLOPHORIA	-1.14 (ENCYCLO)	-0.02
FOVEAL FLICKER	52.43 Hz	-0.06 Hz

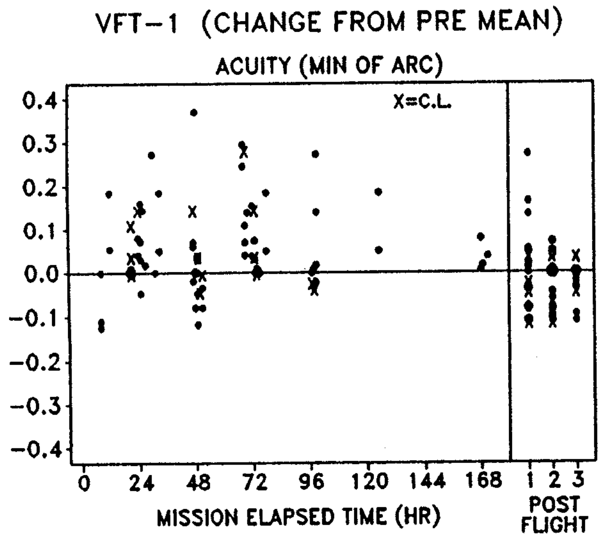
RESULTS

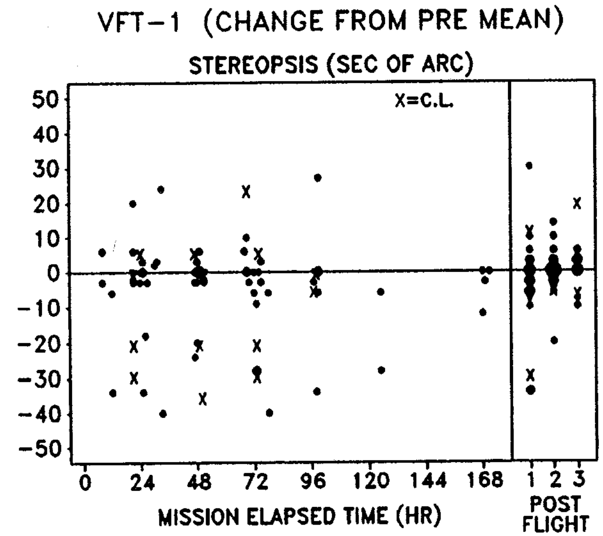
CHANGE DATA

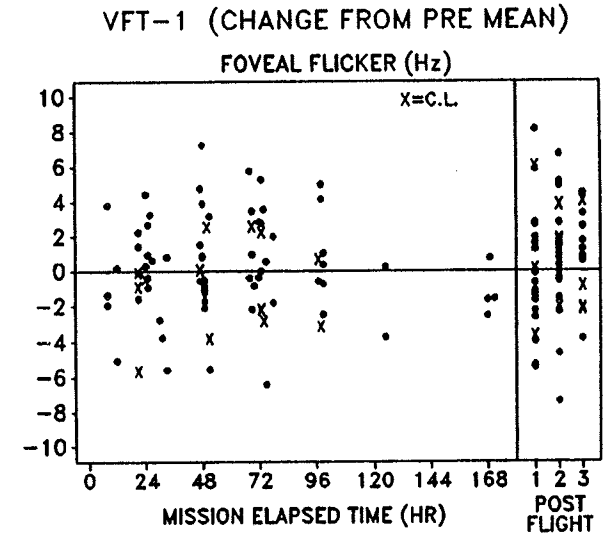
- Difference between mean of two pre-flight sessions (baseline) and each subsequent measurement for each subject was calculated

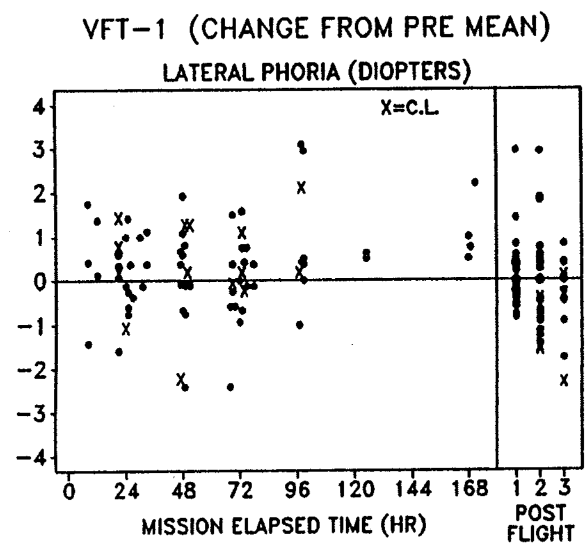
- Size of dots represent number of subjects with same amount of change

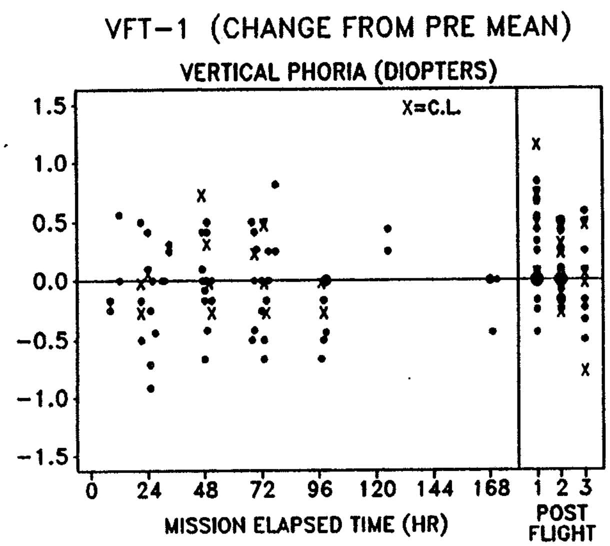
- No apparent trend in change for lateral and vertical phorias, cyclophoria, and critical flicker fusion; nor retinal rivalry (no figure)

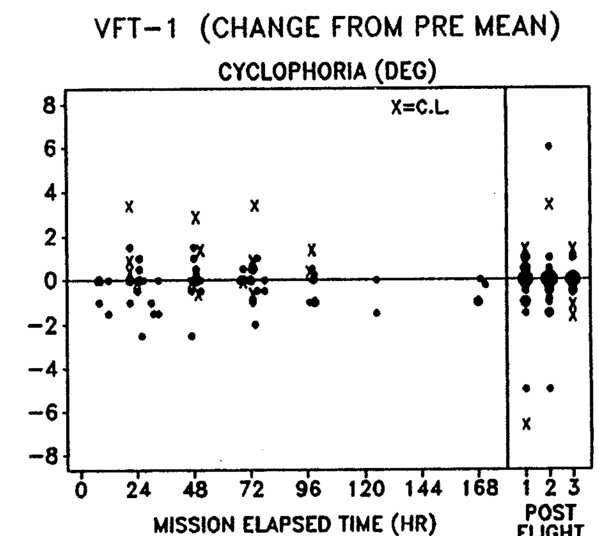

VFT-1 (CHANGE FROM PRE MEAN)
ACUITY (MIN OF ARC)
X=C.L.
0.4
0.3
0.2
0.1
0.0
-0.1
-0.2
-0.3
-0.4
0 24 48 72 96 120 144 168
1 2 3
POST FLIGHT
MISSION ELAPSED TIME (HR)


VFT-1 (CHANGE FROM PRE MEAN)
STEREOPSIS (SEC OF ARC)
X=C.L.
50
40
30
20
10
0
-10
-20
-30
-40
-50
0 24 48 72 96 120 144 168
1 2 3
POST FLIGHT
MISSION ELAPSED TIME (HR)


VFT-1 (CHANGE FROM PRE MEAN)
FOVEAL FLICKER (Hz)
X=C.L.
10
8
6
4
2
0
-2
-4
-6
-8
-10
0 24 48 72 96 120 144 168
1 2 3
POST FLIGHT
MISSION ELAPSED TIME (HR)


VFT-1 (CHANGE FROM PRE MEAN)
LATERAL PHORIA (DIOPTERS)
X=C.L.
4
3
2
1
0
-1
-2
-3
-4
0 24 48 72 96 120 144 168
1 2 3
POST FLIGHT
MISSION ELAPSED TIME (HR)


VFT-1 (CHANGE FROM PRE MEAN)
VERTICAL PHORIA (DIOPTERS)
X=C.L.
1.5
1.0
0.5
0.0
-0.5
-1.0
-1.5
0 24 48 72 96 120 144 168
1 2 3
POST FLIGHT
MISSION ELAPSED TIME (HR)


VFT-1 (CHANGE FROM PRE MEAN)
CYCLOPHORIA (DEG)
X=C.L.
8
6
4
2
0
-2
-4
-6
-8
0 24 48 72 96 120 144 168
1 2 3
POST FLIGHT
MISSION ELAPSED TIME (HR)

2

RESULTS

STEREOPSIS CHANGE

- Slight trend toward smaller sec-of-arc stereopsis on-orbit (i.e., improvement), not apparent at landing or after

- On-orbit change from pre-flight baseline
  -- Mean change at subject's first and last data = -5.0 arc sec

- Mean group change in stereopsis on-orbit was -4.9 arc sec from baseline; nearly significant ($p = 0.07$)

- Post-flight, change was only -0.8 arc sec at landing and was +1.1 arc sec by second post-flight (L+3 days) session

RESULTS

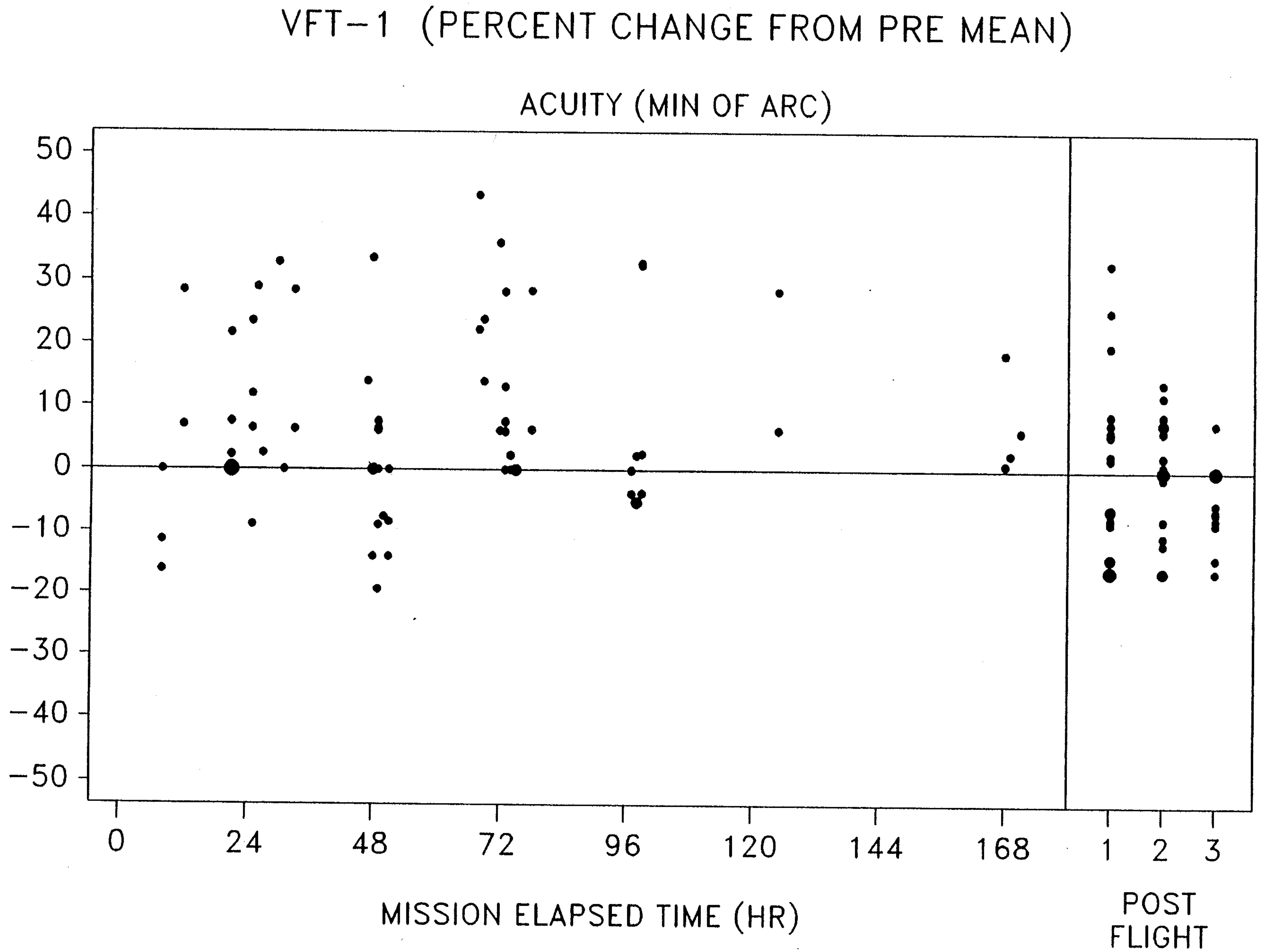
VISUAL ACUITY CHANGE

- Definite trend toward larger min-of-arc resolution on-orbit (i.e., decreased acuity), not apparent at landing or after

- On-orbit change from pre-flight baseline
  - -- Mean change at first on-orbit data = +0.04 min arc ($p = 0.13$)
  - -- Mean change at last on-orbit data = +0.07 min arc ($p = 0.001$)
  - -- No significant difference between first and last data ($p = 0.15$)

- Significant mean group change of 0.06 min arc in visual acuity on-orbit from baseline ($p = 0.005$)

- No change from pre-flight baseline at landing or after ($p=0.90$)


VFT-1 (PERCENT CHANGE FROM PRE MEAN)
ACUITY (MIN OF ARC)
50
40
30
20
10
0
-10
-20
-30
-40
-50
0
24
48
72
96
120
144
168
1
2
3
MISSION ELAPSED TIME (HR)
POST FLIGHT

DISCUSSION

- No group changes on-orbit in lateral and vertical phorias, cyclophoria, critical flicker fusion, and retinal rivalry
- Mean group visual acuity loss on-orbit of only +0.06 min arc; corresponds to only slight change in Snellen acuity from 20/12.2 at baseline to 20/13.4 on-orbit
- Mean percent loss in acuity on-orbit = 7.5%; single data points ranged from 40% loss to 20% improvement

DISCUSSION (Con't)

- Mean group stereopsis improvement on-orbit of only 4.9 arc sec. Some subjects with marked improvement

- Two repeat subjects, in general, confirmed their initial results. Both subjects had large improvements in stereopsis on-orbit. Also found at the second mission (although one on-orbit data point varied for each)

EFFECT OF MICROGRAVITY ON VISUAL CONTRAST THRESHOLD DURING STS SHUTTLE MISSIONS

VISUAL FUNCTION TESTER - MODEL 2 (VFT-2)

LT COL MELVIN R. O'NEAL, O.D., Ph.D.
H. LEE TASK, Ph.D.
COL LOUIS V. GENCO, O.D., M.S.

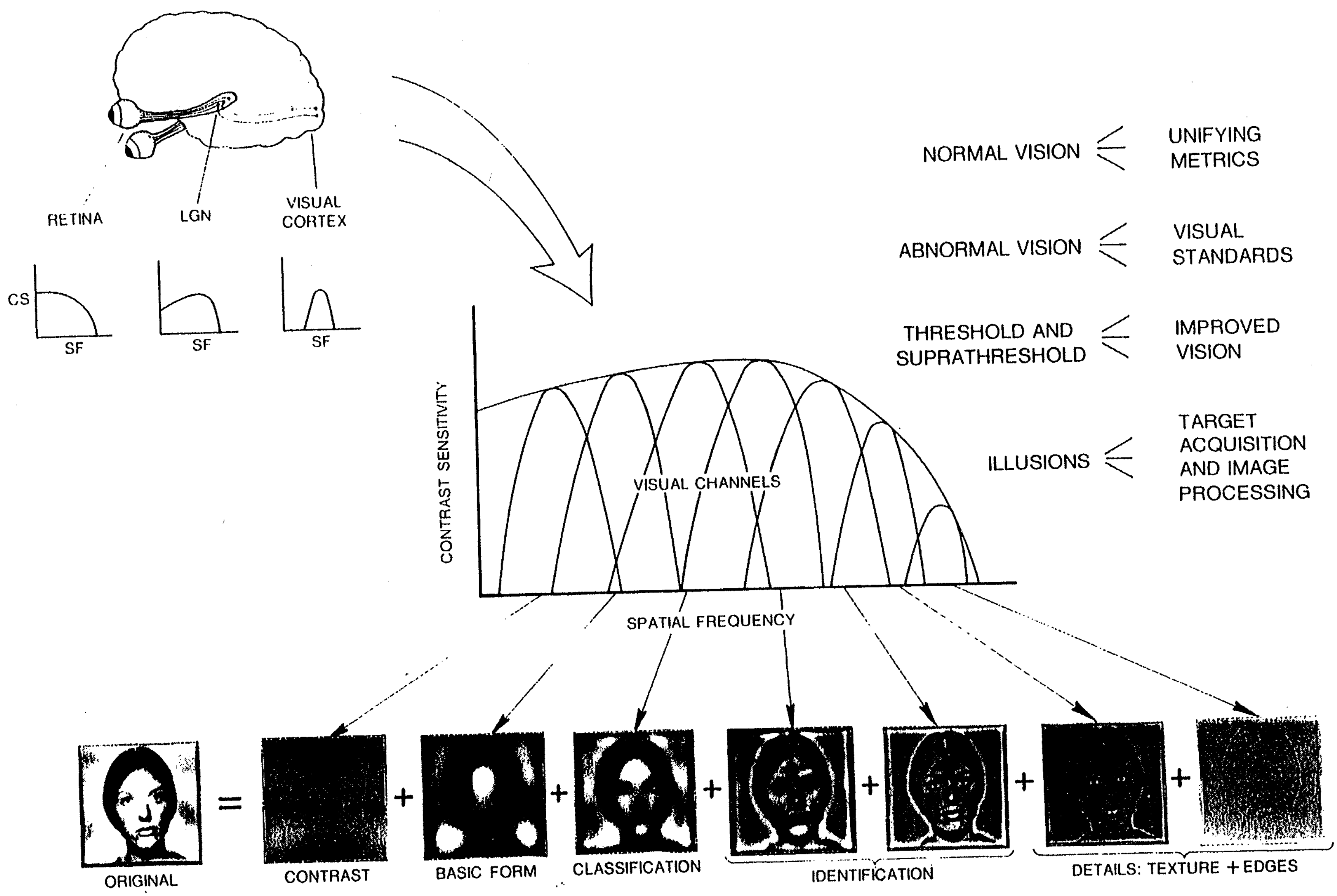
PURPOSE (VFT-2)

- Previous contrast threshold studies, both U.S. and Soviet, at different test distances and may be affected by age, lighting, and method of target presentation

- Determine effect of microgravity on distance visual contrast threshold over mission duration

- Use variable contrast adjustment device under controlled lighting condition to obtain more precise threshold measurement

- Test at multiple spatial frequencies and with additional target types to more completely evaluate



RETINA
LGN
VISUAL CORTEX
CS
SF
SF
SF
NORMAL VISION
UNIFYING METRICS
ABNORMAL VISION
VISUAL STANDARDS
THRESHOLD AND SUPRATHRESHOLD
IMPROVED VISION
ILLUSIONS
TARGET ACQUISITION AND IMAGE PROCESSING
CONTRAST SENSITIVITY
VISUAL CHANNELS
SPATIAL FREQUENCY
ORIGINAL
CONTRAST
BASIC FORM
CLASSIFICATION
IDENTIFICATION
DETAILS: TEXTURE + EDGES

HE-79-3-2

METHODS (VFT-2)

SUBJECTS

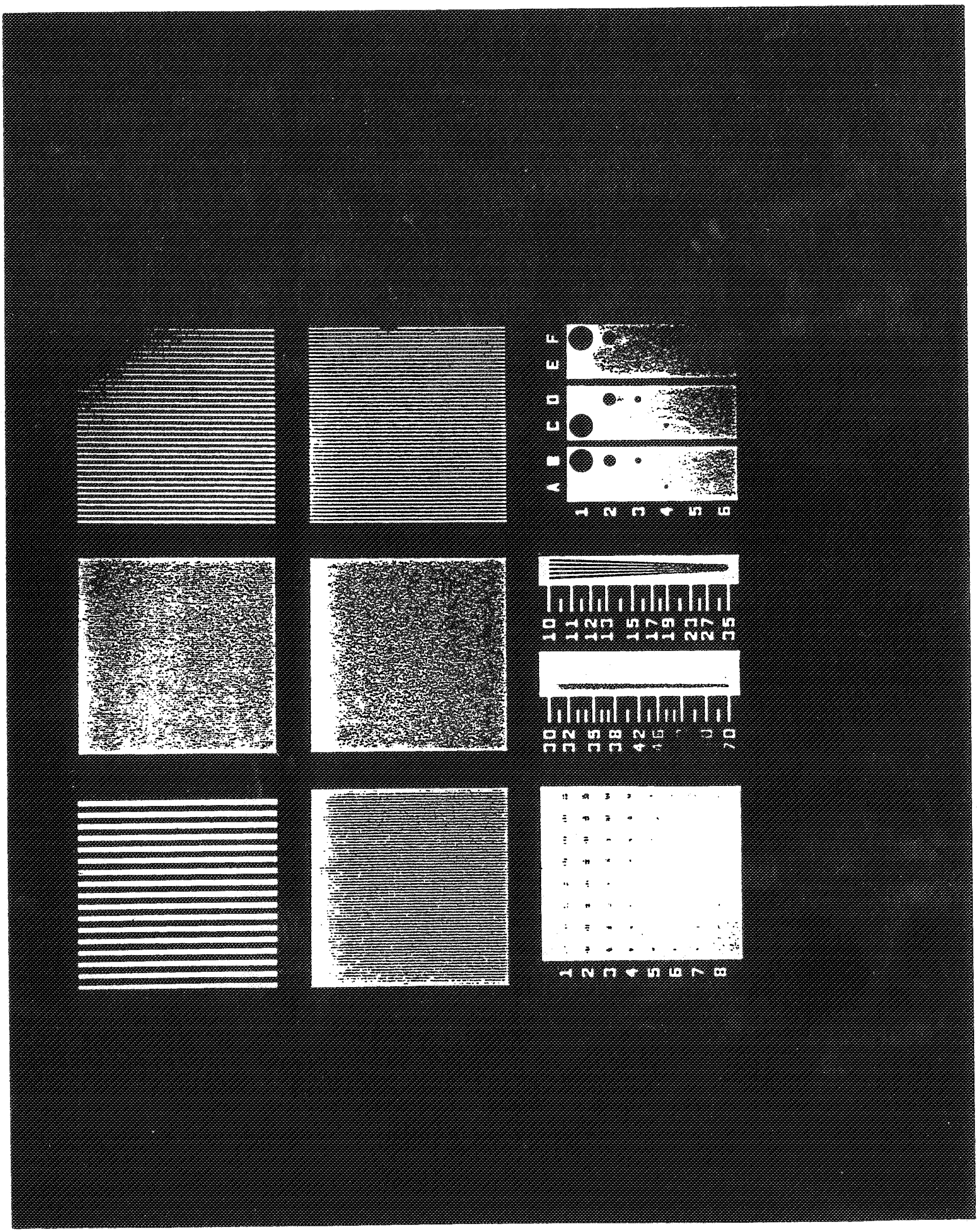
- 5 Flights, 12 STS Astronauts
  - -- 3 subjects with no post flight data
  - -- 1 uncorrected (no glasses)
  - -- 1 SCL, 1 Toric-SCL

APPARATUS

- Visual Function Tester - Model 2 (VFT-2)
  - -- Small, hand-held, battery powered
  - -- Three target types:
    - Square-wave gratings (detection task)
    - Disks (detection task)
    - Tribars (orientation task)

PROCEDURE

- SAME AS VFT-1



A B C D E F
1 2 3 4 5 6
10 11 12 13 15 17 19 23 27 35
1 2 3 4 5 6 7 8

RESULTS

VISUAL CONTRAST THRESHOLD

- Insufficient number of subjects for report at this time
- VFT-2 manifested on STS-53 (2 astronauts) scheduled to fly Dec 92
- Preliminary reporting of visual psychophysical study may affect subsequent data and should be avoided

EFFECT OF MICROGRAVITY ON THE VISUAL NEAR POINT

Visual Function Tester - Model 4 (VFT-4)

Maj GERALD GLEASON OD, PhD
H. LEE TASK PhD
Lt Col MELVIN R O'NEAL OD, PhD

Visual Display Systems Branch
Human Engineering Division
Armstrong Laboratory

AL / CFHV
Wright-Patterson AFB
OH, 45433

BACKGROUND

- Duntley Study (GEMINI)
  - No Significant Changes in (Far) Vision
  - EXCEPT Complaints by 2 Astronauts of Blurred Vision While Reading (Anecdotal)
- Reports of Blurred Near Vision Continue
  - 25 - 40% of Shuttle Astronauts
    - CR Gibson, FK Manuel (NASA-Contracted Optometrists)
    - MP Caputo (KRUG Life Sciences)

BACKGROUND

- Potpourri of Blurred Near Vision Reports
  - No Concurrent Blurred Far Vision (Duntley Studies, VFT1, VFT2)
  - Immediate Upon Orbit
  - Attenuates After Several Days
  - Normal Upon Return To Earth
  - 40 and Older

PROBLEM

- Anecdotal Reports Without Supporting Data
- Possible Etiology
  - Loss of Accommodation
    - Parasympathetic Nervous System
  - Shift Towards Hyperopia
    - Physiological Cause (Choroidal Swelling)
    - Vegetative Cause (Water Balloon)
      - Flatter in Gravity
      - More Circular in Space
- Course Unknown

OBJECTIVES

- Near Term
  Verify Astronaut's Complaints
- Determine Category of Cause
- Provide Data Base Concerning Extent and Range of Effects
- Long Term

  Determine Specific Cause

  Develop a Quantitative Model that Predicts On-Orbit Visual Changes

APPROACH

- VISION FUNCTION TESTER - Model 4 (VFT4)
  - Investigate Ground - Space Differences:
    - Far Point of Vision
    - Near Point of Vision
    - Speed (Facility) of Accommodation

BENEFITS

- Gain Knowledge Concerning MicroGravity Effects on Vision
- Better Prescribe Prophylactic Spectacles In Support of Space and Hypersonic Crew Stations
- Identify Crew Members Who Could Benefit From Vision Training Prior to Flight

STATUS

- Software - Completed
- Hardware
  - 3 Units
    - 2 Units are Fully Functional & Validated Optically
    - 1 Unit Near Completion
  - All Units Will Be Converted from Battery Power to Orbiter Power
  - DOS Computer with RS-232 Serial Port (NASA-Supplied PGSC)
  - Half-Locker (VFT4 Unit & 2 Floppy Disks)

THE EMULSION CHAMBER TECHNOLOGY EXPERIMENT

N93-28743

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Huntsville, Alabama 35899-2900
(205) 895-6028

097-32
174988

ABSTRACT

Photographic emulsion has the unique property of recording tracks of ionizing particles with a spatial precision of 1μm, while also being capable of deployment over detector areas of square meters or 10's of square meters. ~~Since the~~ detectors are passive, their cost to fly in Space is a fraction of that of ~~electronic~~ instruments of similar collecting ~~power~~ A major problem in their continued use has been the labor intensiveness of data retrieval by traditional microscope methods. Two factors ~~are radically~~ changing the acceptability of emulsion technology in Space.or ~~First is~~ the astronomical costs of flying large electronic instruments such as ionization calorimeters in Space.and ~~Secondly~~, the power and low cost of computers ~~has allo~~wed a small revolution in the ~~automation of~~ laboratory microscope data-taking.

~~Given th~~is situation, much needs to be done to bring the technology up to ~~date from~~ the Space launch and performance aspects. Our group at UAH, and ~~our colleagues at MSFC,~~ have made measurements of the high energy composition and spectra of cosmic rays. The Marshall group has also specialized in space radiation dosimetry. We have developed ionization calorimeters, using alternating layers of lead and photographic emulsion, to measure particle energies up to 10^{15}eV. We have performed 10 balloon flights with them. No such calorimeters have ever flown in orbit.

In the ECT program, a small emulsion chamber has been developed and will be flown on the Shuttle mission OAST-2 to resolve the principal technological questions concerning space exposures. These include assessments of: (1) Pre-flight and orbital exposure to background radiation, including both self-shielding and secondary particle generation. The practical limit to exposure time in space can then be determined. (2) Dynamics of stack to optimize design for launch and weightlessness, and (3) Thermal and vacuum constraints on emulsion performance. All these effects are cumulative and affect our ability to perform scientific measurements but cannot be adequately predicted by available methods.

The emulsion chamber is contained in an hermetically sealed aluminum box. Temperature is maintained within ±2°C by passive cooling and active heating.

All materials acquisition is complete and fabrication of qualification and flight units is 70% complete. We will shortly begin qualification and acceptance testing at MSFC and GSFC.

NASA/DOD Flight Experiments Technical Interchange Meeting

Sponsored by the Space Technology Interdependency Group
Flight Experiments Committee

October 5-9, 1992
Monterey, California

~~S46-35~~
~~159247~~
P. 16

DEVELOPMENT OF EMULSION CHAMBER TECHNOLOGY

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EMULSION CHAMBER TECHNOLOGY
UAH

◊ EXPERIMENT OBJECTIVE:

DEVELOP SPACE BORNE EMULSION CHAMBER TECHNOLOGY SO THAT COSMIC RAYS AND NUCLEAR INTERACTIONS MAY SUBSEQUENTLY BE STUDIED AT EXTREMELY HIGH ENERGIES WITH LONG EXPOSURES

◊ DESCRIPTION/APPROACH:

- **A SMALL EMULSION CHAMBER WILL BE DEVELOPED AND FLOWN ON AN STS FLIGHT TO RESOLVE THE PRINCIPAL TECHNOLOGICAL QUESTIONS CONCERNING SPACE EXPOSURES**
  - PREFLIGHT AND ORBITAL EXPOSURE TO BACKGROUND RADIATION INCLUDING BOTH SELF-SHIELDING AND SECONDARY PARTICLE GENERATION FROM ENERGETIC BACKGROUND RADIATION. THE PRACTICAL UPPER LIMIT TO EXPOSURE TIME IN SPACE CAN THEN BE DETERMINED.
  - ASSESS DYNAMICS OF STACK: PACKAGING AND ALIGNMENT OF THE 300 LAYERS OF MATERIAL FOR LAUNCH AND WEIGHTLESSNESS.
  - THERMAL AND VACUUM CONSTRAINTS AND LIMITS.
  - ASSESS ABILITY TO PERFORM VARIOUS MEASUREMENTS ON INTERACTIONS OBSERVED IN THE TEST FLIGHT CHAMBER WITH KNOWN BACKGROUNDS
- **DETECTOR PLATES INCLUDE 300 EMULSION PLATES AND X-RAY FILMS OF DIFFERENT TYPES, AND CR39 ETCHABLE TRACK DETECTORS. THIS STACK (ALSO INCLUDING INACTIVE MATERIAL SUCH AS LUCITE AND LEAD) MUST BE PROTECTED FROM LIGHT, HEAT, HUMIDITY AND VIBRATION DAMAGE.**
  - THE EMULSION CHAMBER IS CONTAINED IN A HERMETIC ALUMINUM BOX WHICH IS PARTIALLY EVACUATED
  - PASSIVE COOLING, HEATERS, THERMISTORS AND TEMPERATURE DATA RECORDER
  - EMULSION PLATE SURFACES PROTECTED BY SURFACE COATING AND SOFT FILM

EMULSION CHAMBER TECHNOLOGY
UAH

EXPERIMENT DESCRIPTION

- DETECTOR PLATES INCLUDE NUCLEAR TRACK EMULSIONS OF DIFFERENT TYPES, X-RAY FILM AND CR-39 ETCHABLE TRACK DETECTORS. THIS STACK (ALSO INCLUDING INACTIVE MATERIALS SUCH AS LUCITE AND LEAD) MUST BE PROTECTED FROM LIGHT, HEAT, HUMIDITY AND VIBRATION DAMAGE.

- THE STACK (OR EMULSION CHAMBER) IS CONTAINED IN A HERMETIC ALUMINUM BOX WHICH IS PARTIALLY EVACUATED. THE BOX HAS A HONEYCOMB LID TO REDUCE NUCLEAR INTERACTIONS OF INCOMING COSMIC RAYS.

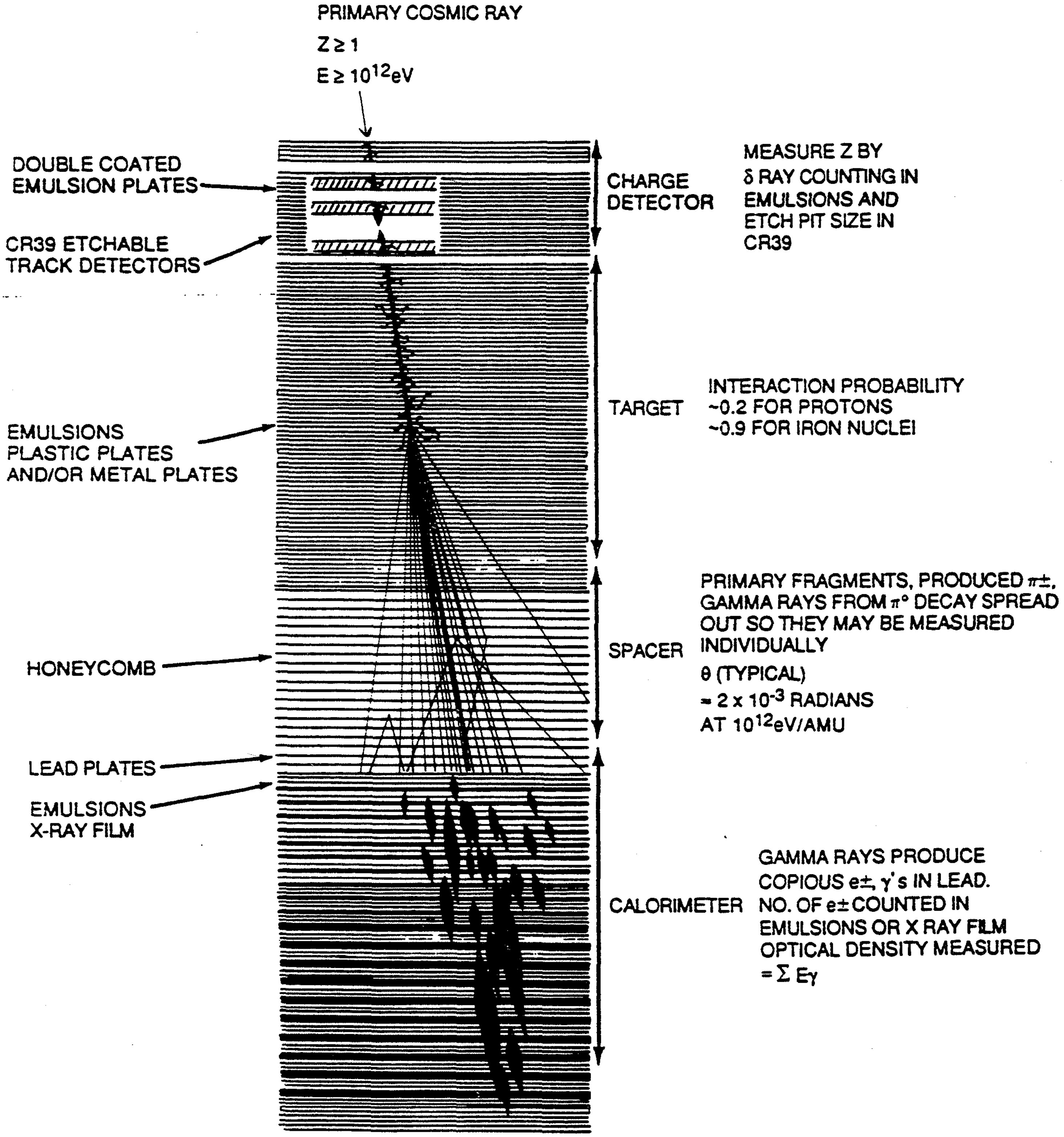
- DIMENSIONS: 50 CM X 60 CM X 40 CM

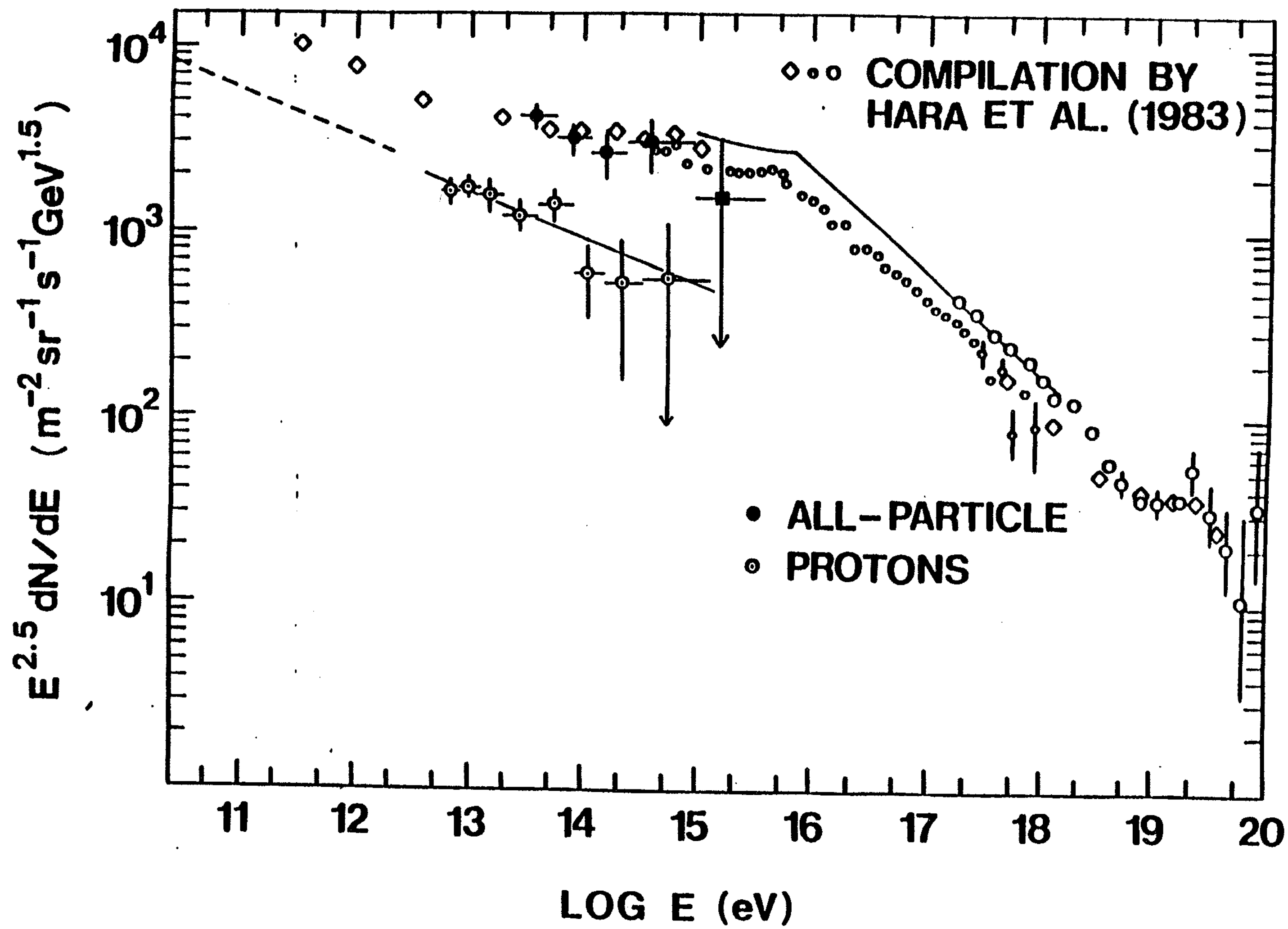
- WEIGHT: 180 KG

- TEMPERATURE: 20 $\pm$ 2°C;

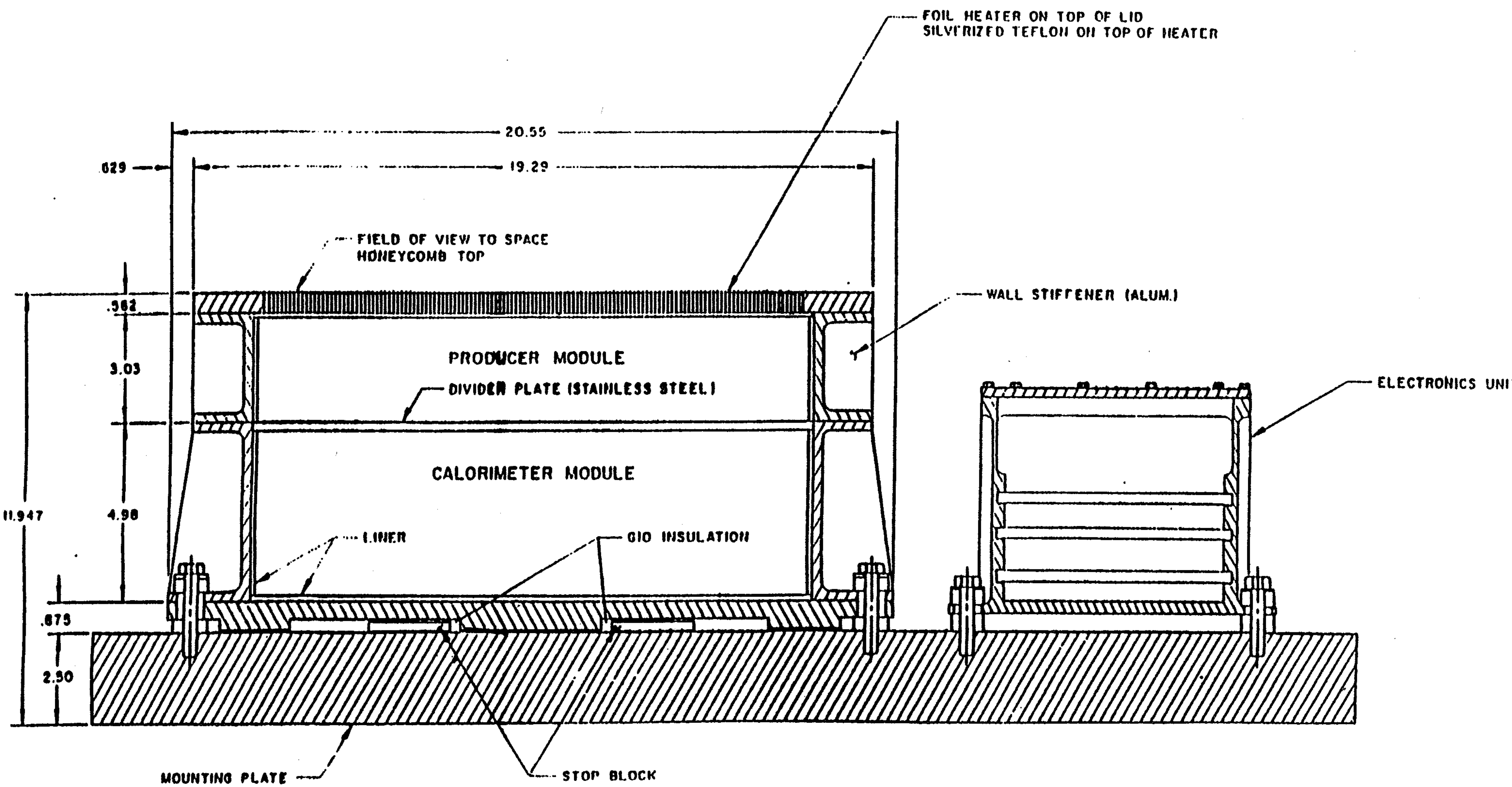
- ORBITAL REQUIREMENTS: $\leq$ 57 DEG; $\leq$ 400 KM; 5 - 10 DAYS

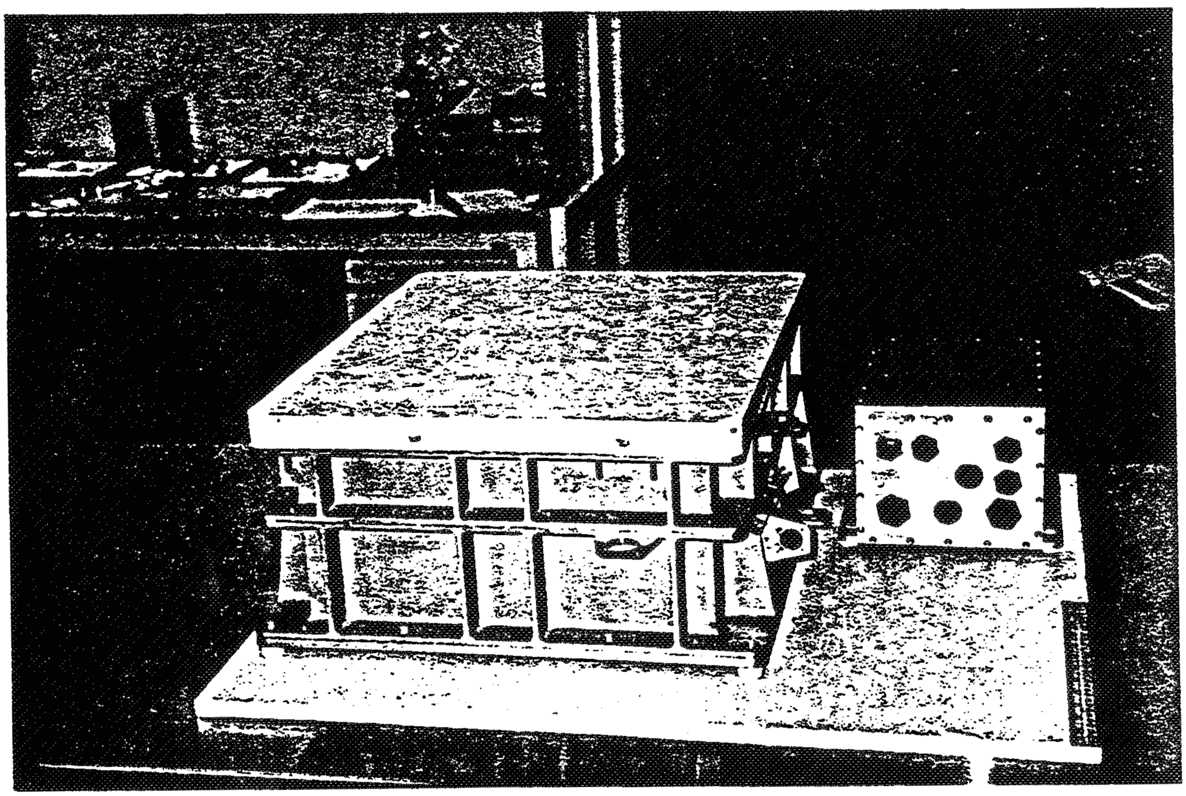
EMULSION CHAMBERS





THE FIGURE SHOWS THE CURRENT KNOWLEDGE OF THE SPECTRUM AND COMPOSITION OF HIGH ENERGY COSMIC RAYS AT THE HIGHEST ENERGIES. THE UPPER DATA SETS ARE "ALL-PARTICLE" SPECTRA (PARTICLES CANNOT BE DIFFERENTIATED BY CHARGE OR MASS). THE ONLY ELEMENTAL SPECTRA ABOVE 10^{13} ELECTRON VOLTS EXISTING (PROTON SPECTRUM SHOWN FOR EXAMPLE) WERE OBTAINED BY THE JACEE COLLABORATION (UAH, MSFC AND OTHERS) USING EMULSION CALORIMETERS ON BALLOONS.


FOIL HEATER ON TOP OF LID
SILVERIZED TEFLON ON TOP OF HEATER
20.55
.029
19.29
FIELD OF VIEW TO SPACE
HONEYCOMB TOP
WALL STIFFENER (ALUM.)
.962
PRODUCER MODULE
3.03
DIVIDER PLATE (STAINLESS STEEL)
ELECTRONICS UNIT
CALORIMETER MODULE
11.947
4.98
LINER
G10 INSULATION
.875
2.50
MOUNTING PLATE
STOP BLOCK



THE DEVELOPMENT UNIT OF THE ECT SHOWING THE EMULSION CHAMBER AND ELECTRONICS UNIT. THE SCALE AT LOWER RIGHT IS ONE FOOT.

EMULSION CHAMBER TECHNOLOGY
UAH

COSMIC RAY CALORIMETRY

◊ VALUE OF EMULSION CHAMBER TECHNOLOGY:

EMULSION CHAMBERS EMPLOY IONIZATION CALORIMETRY

- **THE ONLY TECHNIQUE THAT CAN BE USED TO MEASURE PARTICLE ENERGY ABOVE 10^{14} EV**

TYPES OF IONIZATION CALORIMETERS

- **ELECTRONIC SCINTILLATION COUNTERS**
  (ONE DIMENSIONAL, TOTAL ABSORPTION REQUIRED, HEAVY WEIGHT, TYPICALLY 60-100 TONS OR 1,000 g/cm^2 REQUIRED FOR A SMALL GEOMETRY FACTOR: 0.1 sr)
- **AIR SHOWERS**
  (OBSERVATIONS MADE OVER 40 YEARS BY MORE THAN 30 GROUND-BASED STATIONS, NO DIRECT INFORMATION ON PRIMARY PARTICLES)

ADVANTAGES OF EMULSION CHAMBERS

- **VERY LARGE GEOMETRY FACTOR FOR WEIGHT**
- **COMPLETE INFORMATION ABOUT EACH EVENT**
- **LOW COST TO ASSEMBLE**

LIMITATION OF EMULSION CHAMBERS

- **BACKGROUND PROVIDES UNCERTAIN TIME LIMIT TO EXPERIMENTS**
  (EXPECTATION: ~2-24 MONTHS, NEEDS INVESTIGATION)

EMULSION CHAMBER TECHNOLOGY
UAH

COSMIC RAY CALORIMETRY CONT'D

◊ JUSTIFICATION FOR SPACE FLIGHT:

- **SPACE EXPOSURE OF EMULSION CHAMBERS OFFER A LARGE INCREASE IN EXPOSURE FACTOR TO COSMIC RAY NUCLEI AND THUS EXTENSION TO HIGHER ENERGY REGIMES.**

- **EFFECT OF SPACE ENVIRONMENT ON EMULSION CHAMBER MEASUREMENTS CANNOT PRESENTLY BE ADEQUATELY SIMULATED OR CALCULATED TO PREDICT OPTIMUM EXPOSURE TIME.**

◊ BENEFITS/PAYOFFS:

- **EMULSION CHAMBERS IN SPACE COULD POTENTIALLY EXTEND THE KNOWLEDGE OF COSMIC RAY COMPOSITION AND NUCLEAR INTERACTION CHARACTERISTICS TWO ORDERS OF MAGNITUDE IN ENERGY.**

- **NUCLEUS-NUCLEUS COLLISIONS COULD BE STUDIED WELL ABOVE ACCELERATOR ENERGIES.**

- **EFFECTS OF LARGE SHIELDING TO SPACE RADIATION CAN BE MEASURED AND MODELLED FOR APPLICATION TO LATER MISSIONS.**

SPACE RADIATION ENVIRONMENT

◊ SPACE RADIATION

◊ MAJOR PRIMARY COMPONENTS

- GALACTIC COSMIC RAYS
- GEOMAGNETICALLY TRAPPED PROTONS
- GEOMAGNETICALLY TRAPPED ELECTRONS
- SOLAR FLARE PARTICLES

◊ MAJOR SECONDARY COMPONENTS

- PROJECTILE FRAGMENTS
- SHOWER PARTICLES (MESONS, ELECTRONS, GAMMA RAYS)
- TARGET FRAGMENTS
- SECONDARY NEUTRONS AND PROTONS
- BREMSTRAHLUNG (X-RAYS) FROM ELECTRONS
- RECOIL NUCLEI

◊ GROUND BACKGROUND

- MU-MESONS, PI-MESONS, GAMMA RAYS AND ELECTRONS FROM COSMIC RAY SHOWERS
- GAMMA RAYS FROM RADIOACTIVITY IN SOIL, CONCRETE, AIR

EMULSION CHAMBER TECHNOLOGY
UAH

LONG DURATION SPACE RADIATION TECHNOLOGY ISSUES

1. HUMAN EXPOSURE CONSIDERATIONS

• **ALLOWED DOSE TO PUBLIC (vs)**		**0.5 REM/YEAR**
• **1 YEAR GCR DOSE (Outside Earth's Field)**		**50 REM/YEAR**
• **SOLAR EVENTS**	**UP TO**	**1000 REM**

2. ELECTRONICS

   - **SOFT FAULTS OR LATCH-UPS CAUSED BY IONIZATION AS CHARGED PARTICLE TRANSITS DEVICE**

   - **EFFECT IS RELATED TO ENERGY DEPOSITED BY THE PASSING PARTICLE AND IS RELATED TO ITS CHARGE AND ENERGY**

   - **FOR A PARTICULAR DEVICE TECHNOLOGY THE THRESHOLD DEPOSITION IS CALLED Q (crit)**

3. LONG DURATION MANNED FLIGHTS, AND SPECIAL ELECTRONICS WILL REQUIRE HEAVY SHIELDING.

4. MOST CODES USED DO NOT INCLUDE COMPONENTS DUE TO RECOIL, SPALLATION, NUCLEAR EVAPORATION ETC., WHICH BECOME VERY IMPORTANT UNDER SHIELDING.

EMULSION CHAMBER TECHNOLOGY
UAH

RADIATION TRANSPORT AND DOSE CALCULATIONS

- **PARTICLE FLUXES AT A SPACECRAFT ARE GENERALLY ANISOTROPIC AND DEPENDENT ON ORBITAL INCLINATION AND ALTITUDE AS WELL AS POSITION IN THE ORBIT**

- **VARIOUS PROGRAMS THEN CALCULATE ENERGY DEPOSITION IN PLANAR OR SPHERICAL TARGETS, E.G. CREME USES PRIMARY COSMIC RAY FLUXES AND SPECTRA MODIFIED BY THE EARTH'S FIELD AND TRANSPORTS THE RADIATION THROUGH A TARGET ALLOWING SLOWING AND NUCLEAR INTERACTION (NO FRAGMENT PRODUCTION)**

- **HETC ALLOWS PRODUCTION OF SECONDARIES (SEC. N'S AND P'S, EVAPORATION PARTICLES AND HEAVY PROJECTILE FRAGMENTS)**

- **NONE OF THESE PROGRAMS APPLIED TO SPACE CURRENTLY INCLUDES HIGH ENERGY NUCLEAR-ELECTROMAGNETIC CASCADE SHOWER PRODUCTION**

- **PREDICTION OF DOSE UNDER HEAVY SHIELDING WHERE THE RADIATION IS LARGELY SECONDARY NEEDS VERIFICATION**

- **WHEN THE CODES TO PROVIDE ALL DOSE INFORMATION ARE FULLY ASSEMBLED, THERE IS NO WAY TO VERIFY THEM ON THE GROUND**

SPACE RADIATION ENVIRONMENT

◊ ECT APPROACH TO RADIATION BACKGROUND ASSESSMENT

- **AUGMENT EMULSION CHAMBER DETECTORS WITH STANDARD PASSIVE DOSIMETERS**
  - THERMOLUMINESCENT DOSIMETERS (TOTAL DOSE)
  - LINEAR ENERGY TRANSFER DETECTORS
  - NEUTRON DETECTORS
- **CALCULATE FLIGHT TRACK DENSITIES WITH STANDARD ENVIRONMENT AND TRANSPORT METHODS**
  - TRAPPED BELT MODELS, INCLUDING DIRECTIONAL ASSESSMENT, AND TRANSPORT MODELS
  - COSMIC RAY ENVIRONMENT MODEL AND TRANSPORT MODELS
  - HIGH ENERGY CASCADE MODELS, INCLUDING ELECTROMAGNETIC SHOWER BUILD-UP
- **COMPARE CALCULATIONS AND FLIGHT EXPOSED EMULSION AND X-RAY FILM TO DETERMINE MODEL DEFICIENCIES, MODIFY MODELS FOR LATER APPLICATION**
- **PERFORM MEASUREMENTS ON COSMIC RAY EVENTS AND ASSESS MEASUREMENT EFFICIENCY**
- **ASSESS EFFICIENCY OF LONGER EXPOSURES BY MODELING**
- **MONITOR GROUND BACKGROUND WITH ELECTRONIC COUNTERS AND GROUND PROOF EMULSIONS**

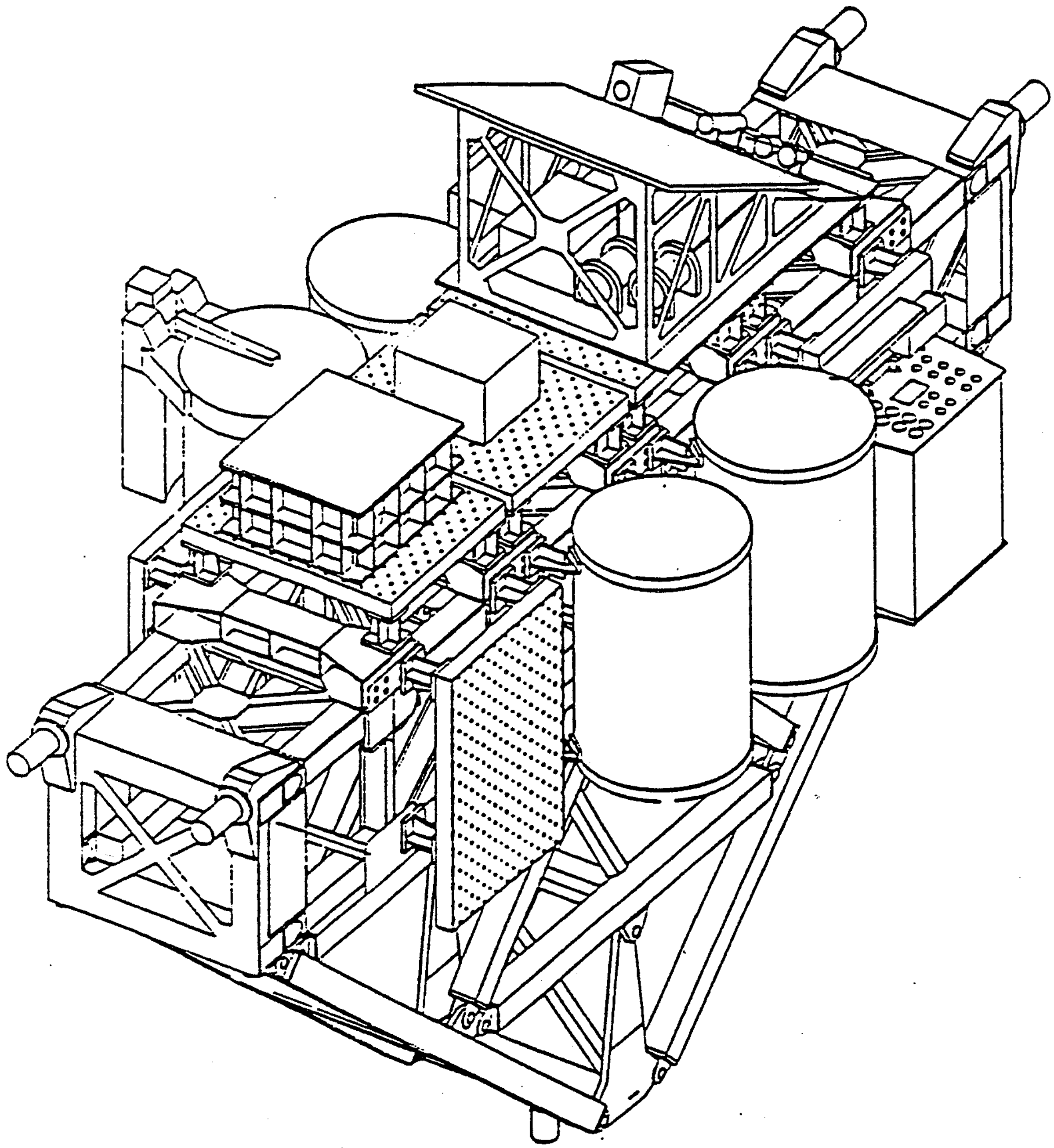
EMULSION CHAMBER TECHNOLOGY
UAH

SUMMARY

- AN EMULSION CHAMBER IN SPACE CAN ESTABLISH AN EXTREMELY EFFICIENT, ON-ORBIT TECHNOLOGY TO MEASURE ENERGETIC COSMIC RAYS FAR BEYOND THE REACH OF ANY EXISTING SPACE FLIGHT EXPERIMENTS

- EMULSION CHAMBER TECHNOLOGY EXPERIMENT CAN VERIFY ABILITY OF EMULSION CALORIMETER FOR LONG EXPOSURE EXPERIMENTS SUCH AS ON SPACE STATION

- MEASUREMENTS OF PARTICLE TRACKS AND RADIATION DOSE IN EMULSION CALORIMETER WILL REVEAL, FOR THE FIRST TIME, EFFECTS OF SELF-SHIELDING OF THICK ABSORBER MATERIAL; WHICH SERVE TO VERIFY A LONG EXPOSURE CAPABILITY OF EMULSION CHAMBERS, AND TO PROVIDE A UNIQUE DATA BASE OF SHIELDING AGAINST SPACE RADIATION FOR FUTURE SPACE MISSIONS



INSTRUMENT COMPLEMENT FOR OAST-2 SHOWING THE ECT EMULSION CHAMBER AND ELECTRONICS BOX MOUNTED ON THE TOP SURFACE OF THE CROSS-BAY MPESS CARRIER. THE OTHER LARGER INSTRUMENT MOUNTED ON THE TOP IS THE LOCKHEED SPACECRAFT GLOW EXPERIMENT.